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This report is made up of the combined reports of 35 separate teams of students who entered the 2002 Design, Build & Fly Competition. The objectives of the Design, Build & Fly Competition were to have students teams design, build and fly unmanned remote control electric aircraft designed for two specific missions: a short-field cargo sortie and a cruise/ferry sortie. A "fly-off" took place on the Cessna Flight Test Field near Wichita, KS, in April 2002. Winners of the contest: 1st place, University of CA-San Diego; 2nd, University of Southern CA; 3rd, West Virginia University. The Design, Build & Fly Competition was supported by Cessna, the Office of Naval Research and the AIAA Foundation.

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CESSNA / ONR / STUDENT DESIGN/BUILD/FLY COMPETITION

An AIAA Student Activity

The 2002 Cessna/ONR Student Design/Build/Fly competition was held at the Cessna flight test center in Wichita Kansas over the weekend of 26-28 April. Twenty eight teams from the United States and three foreign countries; Canada, Italy and Turkey, competed under sometimes challenging weather conditions to see whose aircraft would complete all three sorties over the prescribed course in the minimum time. Of the 28 teams attending the fly-off competition, 27 made at least one scoring flight attempt, with many teams making multiple flights during the two days of competition. Only one team, La Sapienza from Italy, made 5 successful scoring flights (the maximum allowed)

The competition spanned two days of flying, with the flight queue filled with aircraft waiting for their turn to make a competition flight. During the second day of the competition a total of 69 multiple sortie flights were flown.

The total score for each team was comprised of their flight performance on their best three flights, their score on a written report documenting their aircraft design and selection, and a "Rated Aircraft Cost" representing the complexity and manufacturing costs of their design.

The final results showed a close battle between teams from the University of California at San Diego, University of Southern California, and West Virginia University. In the end, the University of Southern California team was not able to match the top team from their neighbor to the south the University of California at San Diego. The highest score obtained on the written report portion of the competition was from Oklahoma State University, Team Orange with 94 of a possible 100 points awarded.

The final positions and scores for all of the competing teams are listed in the table below.

More details on the 2002 competition objectives and rules can be found at the contest web site at http://amber.aae.uiuc.edu/~aiaadbf

School	Team	Paper	RAC	Flight	Score	Position
Univ. of Calif at San Diego #2	TLAR 3	79.0	9.81	16.45	132.52	1
USC	SCrewball	92.5	13.11	16.32	115.17	2
West Virgina Univ. #2	Phastball	86.5	12.49	15.22	105.43	3
Univ. of Illinois, Champain	Illiniwek 1	88.0	13.31	12.99	85.87	4
La Sapienza, Italy	Flying Centurions	65.8	13.52	15.06	73.27	5
San Diego State University	Monty's Revenge	81.0	13.95	8.82	51.22	6
VA Polytechnic and State Univ.	Marooned	82.8	13.07	7.80	49.37	7
Calif. Polytechnic, San Luis Obispo	Third Base	86.3	8.41	4.18	42.84	8
USA Naval Academy #2	USNA Gold	82.3	13.57	4.87	29.54	9
Mississippi State Univ. #1	Fast Pitch	88.5	16.76	4.25	22.46	10
USA Naval Academy #1	USNA Blue (Tango)	79.4	15.49	2.97	15.20	11

Istanbul Technical University	ATA4	69.5	14.52	1.54	7.38	12
University of Arizona #2	AirCat2002	39.3	13.18	2.19	6.52	13
Case Western Reserve Univ.	Learning Curve	41.3	17.09	1.81	4.38	14
West Virgina Univ. #1	Daedalus	18.7	15.65	2.50	2.99	15
Oklahoma State Univ. #1	OSU Black	91.9	9.49	0.25	2.45	16
Oklahoma State Univ. #2	OSU Orange	94.0	10.09	0.11	1.07	17
University of Maryland	Terp Flyer	75.2	12.97	0.18	1.02	18
Georgia Institute of Technology	Buzz Light	87.3	13.33	0.14	0.91	19
Queen's University, Canada	Negative Margin	85.2	13.47	0.12	0.79	20
USA Military Academy #1	Team 1	83.4	16.26	0.11	0.57	21
Univ. of Calif at San Diego #1	TLAR 3.5	77.5	9.32	0.01	0.08	22
Wichita State Univ.	Swallow	92.6	11.31	0.01	0.08	23
University of Arizona #1	Evolution	66.0	12.71	0.01	0.05	24
Clarkson University	Knight Hawk	73.0	14.15	0.01	0.05	25
City College of NY #1	Falcon I	70.7	14.03	0.01	0.05	26
Syracuse University	~(Tilda)	70.3	16.24	0.01	0.04	27
Univ. of Texas at Arlington	Black Magic	70.6	17.15	0.01	0.04	28
Univ. of Texas at Austin	Better is the Enemy of Good	89.1	100.00	0.01	0.01	29
Middle East Technical University	Ruzgar Sultan	86.3	100.00	0.01	0.01	30
Mississippi State Univ. #2	Milk Run	85.9	100.00	0.01	0.01	31
Cleveland State University	The Flying Viking	76.3	100.00	0.01	0.01	32
Turkish Air Force Academy	Conqueror	69.8	100.00	0.01	0.01	33
Miami University, Ohio	It's Close Enough	69.3	100.00	0.01	0.01	34
University of Central Florida	Butter Duck	66.2	100.00	0.01	0.01	35

The success of the competition required the efforts of many individuals. A special thanks goes to the judges who assisted in the operation, technical inspections and scoring of the flight competition; and to the many judges who evaluated and scored the teams written proposal reports. Thanks also go to the Applied Aerodynamics, Aircraft Design, Design Engineering, and Flight Test Technical committees of the AIAA who organized and manage the competition, and the AIAA Foundation for their administrative support. Special thanks is due to the competitions corporate supporters, the Cessna Aircraft Company and the Office of Naval Research. And the final and biggest thank you must go to our hosts for the weekend, Cessna Aircraft.

Overall the 2002 Cessna/ONR Student Design/Build/Fly competition marked another very successful event, allowing the participating students to mix a highly enlightening educational experience with a good dose of fun. Congratulations to all the teams who participated for your great enthusiasm and achievement.

See you next year - Greg Page: Contest Director

2001-2002 Design, Build, Fly Competition

Place: Wichita, Kansas—Cessna Air Field

Ranking by Total Score

Order Within This Document

School	Team	Paper	RAC	Flight	*Score*	Position
Univ. of Calif. at San Diego #2	TLAR 3	79.0	9.81	16.45	***132.52***	1
USC	SCrewball	92.5	13.11	16.32	***115.17***	2
West Virginia Univ. #2	Phastball	86.5	12.49	15.22	***105.43***	3
Univ. of Illinois, Champaign	Illiniwek 1	88.0	13.31	12.99	***85.87***	4
La Sapienza, Italy	Flying Centurions	65.8	13.52	15.06	***73.27***	5
San Diego State University	Monty's Revenge	81.0	13.95	8.82	***51.22***	6
VA Polytechnic and State Univ.	Marooned	82.8	13.07	7.80	***49.37***	7
Calif. Polytechnic, San Luis Obispo	Third Base	86.3	8.41	4.18	***42.84***	8
USA Naval Academy #2	USNA Gold	82.3	13.57	4.87	***29.54***	9
Mississippi State Univ. #1	Fast Pitch	88.5	16.76	4.25	***22.46***	10
USA Naval Academy #1	USNA Blue (Tango)	79.4	15.49	2.97	***15.20***	11
Istanbul Technical University	ATA4	69.5	14.52	1.54	***7.38***	12
University of Arizona #2	AirCat2002	39.3	13.18	2.19	***6.52***	13
Case Western Reserve Univ.	Learning Curve	41.3	17.09	1.81	***4.38***	14
West Virginia Univ. #1	Daedalus	18.7	15.65	2.50	***2.99***	15
Oklahoma State Univ. #1	OSU Black	91.9	9.49	0.25	***2.45***	16
Oklahoma State Univ. #2	OSU Orange	94.0	10.09	0.11	***1.07***	17
University of Maryland	Terp Flyer	75.2	12.97	0.18	***1.02***	18
Georgia Institute of Technology	Buzz Light	87.3	13.33	0.14	***0.91***	19
Queen's University, Canada	Negative Margin	85.2	13.47	0.12	***0.79***	20
USA Military Academy #1	Team 1	83.4	16.26	0.11	***0.57***	21
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Univ. of Texas at Austin	Better is the Enemy of Good	89.1	100.00	0.01	***0.01***	29
Middle East Technical University	Ruzgar Sultan	86.3	100.00	0.01	***0.01***	30
Mississippi State Univ. #2	Milk Run	85.9	100.00	0.01	***0.01***	31
Cleveland State University	The Flying Viking	76.3	100.00	0.01	***0.01***	32
Turkish Air Force Academy	Conqueror	69.8	100.00	0.01	***0.01***	33
Miami University, Ohio	It's Close Enough	69.3	100.00	0.01	***0.01***	34
University of Central Florida	Butter Duck	66.2	100.00	0.01	***0.01***	35

2001-2002 Design, Build, Fly Competition

Place: Wichita, Kansas—Cessna Air Field

University of Alabama	* Withdrew *	0.0	100.00	0.01	***0.00***	36
MIT	*Withdrew*	0.0	100.00	0.01	***0.00***	37
USA Military Academy #2	*Withdrew*	0.0	100.00	0.01	***0.00***	38
State Univ. of NY, Buffalo	* DQ*	0.0	100.00	0.01	***0.00***	39
City College of NY #2	*DQ*	0.0	100.00	0.01	***0.00***	40
East Stroudsburg University	*Withdrew*	0.0	100.00	0.01	***0.00***	41
Embry-Riddle Aeronautical Univ.	*DQ*	0.0	100.00	0.01	***0.00***	42
UCLA	*DQ*	0.0	100.00	0.01	***0.00***	43

2001-2002 Design Build, Fly Competition

Place: Wichita, Kansas—Cessna Air Field

Ranking by Paper Score

School	Team	*Paper*	RAC	Flight	Score	Position
Oklahoma State Univ. #2	OSU Orange	***94.0***	10.09	0.11	1.07	17
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Oklahoma State Univ. #1	OSU Black	***91.9***	9.49	0.25	2.45	16
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West Virginia Univ. #1	Daedalus	***18.7***	15.65	2.50	2.99	15

2001-2002 Design Build, Fly Competition

Place: Wichita, Kansas—Cessna Air Field

University of Alabama	* Withdrew *	***0.0***	100.00	0.01	0.00	36
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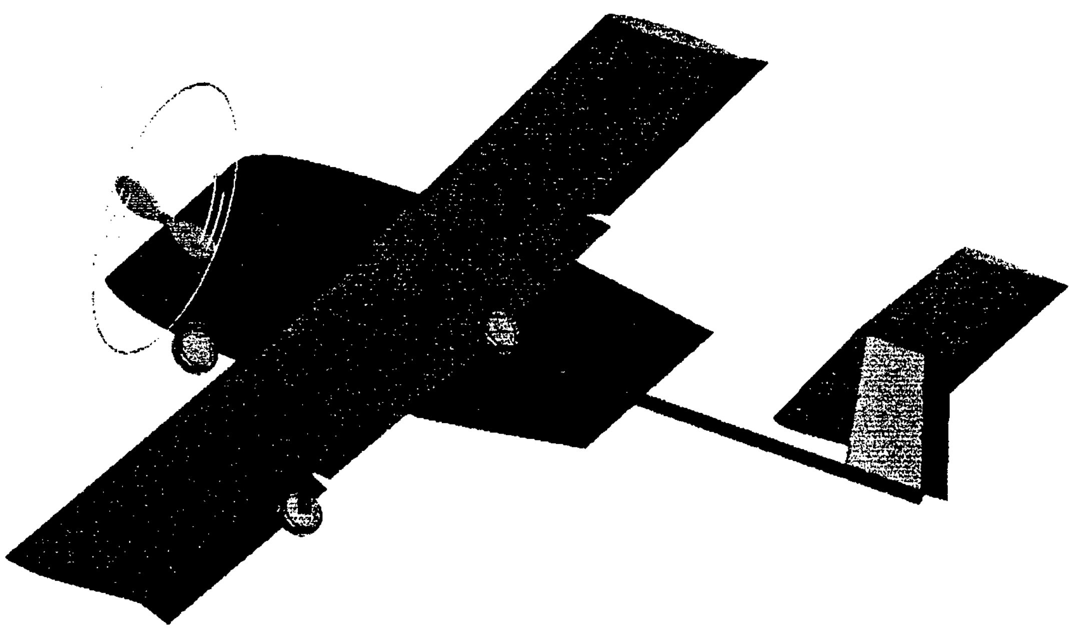
State Univ. of NY, Buffalo	* DQ*	***0.0***	100.00	0.01	0.00	39
City College of NY #2	*DQ*	***0.0***	100.00	0.01	0.00	40
East Stroudsburg University	*Withdrew*	***0.0***	100.00	0.01	0.00	41
Embry-Riddle Aeronautical Univ.	*DQ*	***0.0***	100.00	0.01	0.00	42
UCLA	*DQ*	***0.0***	100.00	0.01	0.00	43

UNIVERSITY OF CALIFORNIA, SAN DIEGO

Proposal Phase Design Report

Submitted March 12, 2002



TLAR III

TLAR III

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1 Executive Summary

Student members of the American Institute of Aeronautics and Astronautics (AIAA) at the University of California, San Diego designed and built a remote-controlled aircraft for the 2002 AIAA Design/Build/Fly Competition to apply and broaden their knowledge of aircraft design and manufacturing.

The AIAA Design/Build/Fly Competition emphasizes the importance of design, construction, and the actual flight performance of the final plane. Based on one of the Mission Tasks requiring the aircraft to carry between 10 to 24 softballs, the team designed the aircraft to carry a high volume payload of 24 softballs each weighing approximately 180 grams with a take-off distance less than 200 ft. The wingspan is approximately 84 in. in length, and the empty weight is 14.9 lbs.

1.1 Objectives

The main objective of the team is to earn the highest score possible. The Rated Aircraft Cost (RAC) must be as low as possible; the Written Report must be as high as possible; and during the actual flight competition, the flight speed must be high while the pit stops must be quick. The lowest RAC was achieved through trial calculations for all of the design phases. To achieve the highest Written Report score, the team edited the Written Report based on the reviews of the faculty and mentor advisors during all paper preparation phase. To achieve the highest flight speed, the team selected the airfoil and propulsion system for TLAR III, and to achieve the quickest pit stop, the team ran pit stop coordination practices.

The second objective is to allow experienced members of the UCSD team to pass knowledge onto novice members. This year, most of the team members are new, being involved in the project for the first time and/or as first year students entering college. The competition is meant to convey the importance of responsible leadership and teamwork.

1.2 Analytical Tools

Several analytical tools were used throughout the entire design process. The most important was Microsoft Excel, which was used for everything from calculations of weight-and-balance, the quickest time to complete task, climb rate, take-off distance, and turn radius, to financial analysis and Rated Aircraft Cost (RAC). General aviation design principles and equations were used in determining exact dimensions and specifications. The mathematical toolbox, Matlab 6.0, was used to perform numerical analysis of flight performance of the chosen wing configuration. Finally, the programs AutoCAD 2000 and PRO/E 2000i were crucial tools used to visualize and present aircraft design ideas.

FOMs and Design Parameters were chosen to provide requirements to ensure that the design would meet the functional requirements dictated by the contest administrators, as well as the design goals as defined by the UCSD AIAA team. They were applied to each stage within the entire design process to analyze the strengths and weaknesses of each component as well as those of the overall

aircraft configurations. The design process used to determine the final configuration began with the Conceptual Design Phase (CDP). Next was the Preliminary Design Phase (PDP) followed by the Detail Design Phase (DDP), from which emerged the final configuration of the aircraft.

1.3 Conceptual Design Phase Outline

The CDP consists of two stages, Stage I and Stage II. Stage I incorporates a range of ideas for each component that was considered. The FOMs of Stage I were geared towards eliminating ideas that were intuitively and/or obviously impractical. These FOMs centered on material availability, handling characteristics, RAC, ease of transportation and complexity of manufacture in relation to time and skill level required. Stage I analysis resulted in component concepts that were determined to be viable. These components were assembled into three different configurations that initially met the design parameters and FOMs. The three configurations were then progressed into Stage II.

The goals of applying Stage II FOMs were to maximize the overall flight score, minimize the RAC, and develop a preliminary aircraft design. Stage II FOMs focus on an in depth analysis of the components, with specific attention to payload access, construction feasibility, propulsion and maximizing strength to weight ratios, operator and component safety, difficulty of repairs, component and interfacing strength, and payload capacity. The analysis of Stage II resulted in a single overall design configuration, at which point, the question was raised of whether it met the functional and design goal requirements. If it was determined that the configuration met these requirements, it continued into the PDP. Otherwise, it was reiterated through Stage II of the CDP in an attempt to optimize the configuration with alternate component designs.

1.4 Preliminary Design Phase Outline

Once an overall design configuration was decided upon, the next step was to conduct a more thorough and detailed analysis of the configuration using the PDP FOMs. The main purpose of the PDP was to optimize sortie performance, RAC, and structural integrity of each component individually, and as part of the entire configuration. Specific design parameters, such as sizing, materials, connection interfaces, and exact location of each part within the assembly were also determined. These, combined with optimization of power requirements, led to a complete detailed aircraft configuration.

1.5 Detail Design Phase Outline

The DDP was less focused on actual aircraft configuration design as compared to the previous two phases. The focus in this section was on determining the flight performance of the detail configuration to discern whether or not the design met the functional requirements. To complete these calculations the team made use of performance analysis techniques suited to propeller driven general aviation and radio controlled aircraft. These techniques were validated by the previous year's design, "TLAR II" and used to determine characteristics such as takeoff distance, turn radius, stall speed and sortie times.

Though manufacturing related FOMs had been applied to eliminate ideas throughout the both the Conceptual and Preliminary Design Phases, in the Manufacturing Phase, each component underwent a

rigorous investigation as to the best method of construction possible. Specific FOMs were developed to take into consideration the restrictions of skill level, availability of materials and machinery as well as time constraints and cost.

The final product of the design process was a configuration in which each component satisfied the contest functional requirements, design goals as well as the FOMs of each category. The resulting design was conventional, single engine, single fuselage, low mono-wing aircraft with a 84 in. wingspan and a 54 in. nose-to-tail length. The specific details of the aircraft are given in Table 1.1.

Table 1-1 Final Design Specifications

Design Parameter	Specification
Manufacturer's Empty Weight	14.9 lbs
Payload Capacity/Weight	24 Softballs/9.6 lbs.
Main Wing Configuration	Low Mono-Wing
Main Wing Airfoil	Eppler-212
Wing Span	84 in.
Length	54 in.
Motor	Graupner Ultra 3300/7
Rated Aircraft Cost	**9.29**
Final Flight Score	6.2

2 Management Summary

2.1 Architecture of the Design Team

The 2002 UCSD Design/Build/Fly Project team consists of seventeen undergraduate students from various engineering disciplines. At the first project meeting it was decided that the most efficient way to design and build the plane would be to group members together with assigned areas to work on. The

project manager, Maziar Sefidan, assigned each group with one of the following focuses: wing, fuselage, propulsion, landing gear or the tail. All members were responsible for fundraising. The specific team member profiles and assignment areas are shown in the table below.

Table 20-1 Archtecture of Team Members

Name	Major	Year	CAD Programs			Matlab	Technical Writing	Machining	Assignment Area
			Pro-Engineer	AutoCAD	SolidWorks				
Maziar Sefidan	ME	SR	X	X		X	X	X	Wings
Josh Adams	ME	SR	X	X		X	X	X	Fuselage
Chad Valenzuela	AE	JR		X			X	X	Propulsion System
Mark Shtayerman	AE	JR	X	X		X	X	X	Landing Gear
Brooke Mosley	AE	FR		X				X	Manufacturing
Steve Wong	ME	SR	X	X	X				Drawing Package
Aaron Pebley	AE	SR			X				Systems Architecture
Hamarz Argafar	AE	SO		X				X	Manufacturing
Justin Smith	AE	SR	X	X					Power Management
Guy Watanabe	AE	SR	X	X		X	X	X	Finite Element Analysis
Matthew Napoli	AE	FR					X		Tail
Brian Berg	AE	FR		X			X		Manufacturing
Carrie Nishimura	AE	FR							Systems Architecture
Lynn Chouw	AE	FR					X		Manufacturing
Jillian Allan	BE	JR			X			X	Graphic Design
Mann Chau	ME	SR	X		X		X	X	Fundraising
Ceazar Javallana	AE	JR	X	X				X	Fundraising

Weekly meetings were set where each sub-group discussed their design ideas, chose designs that were useable, then reported it to the entire team. The end of each meeting was designated as a time to assign topics for which ideas would be needed for the next week's meeting. The product of these brainstorming meetings commenced in the conceptual design of our plane. This management structure allowed the plane to have a safe, stable, and fast design while considering the contributions and ideas of all members of the team. In addition, this format was conductive to a timely completion of the airplane's configuration.

2.1.1 Task Scheduling

In early September, the group decided upon a schedule of completion dates. Each subgroup was expected to complete tasks by a certain deadline. The chart below (Figure 2.1.2) depicts the planned and actual dates of completion (D.O.C.) of each major event. Problems that were encountered completing

these tasks were quickly resolved through teamwork and subgroup collaboration. The subgroup dependencies were as follows, each task's completion vital to starting the next:

Assembly of Design Team: The returning DBF members from 2001 met two weeks before the start of school to decide how many new members the project would need to compete in this years competition. A meeting was held the first week of classes, and the team began to form. Only a month from competition, the team has solidified with 17 members.

Notice of Intent to Compete: This notice was sent by the project manager, Maziar Sefidan, on October 19, 2001, binding our participation in the work ahead of us.

Obtain Funding: While our search for project funding has been continuous over the past 11 months, our saving grace has come in the form of grants from General Atomics (Aeronautical Systems), Jacobs School of Engineering (UCSD), Hitec RCD, Corland Co., San Diego Silent Fliers, and Diversity Model Aircraft .

Concepts and Preliminary Design FOMs: The figures of merit were applied to all conceptual and preliminary designs in order to either eliminate or accept designs based on the structural, weight, safety, or financial needs of the plane.

Conceptual Design: Having found designs that met all of our necessities, a final conceptual design could be assembled and taken to be critiqued by our mentors.

Preliminary Design: The conceptual design having been finalized, a preliminary design could be made and analyzed.

Data Design: Taking our preliminary design to the computers, we digitally took apart our design analyzing each piece to ensure the best flight characteristics possible under design parameters.

Acquire Materials and Parts: The ongoing task of replacing materials that have been used in the manufacture of the plane, and acquiring needed manufactured parts as the project calls for them.

Proposal Phase: Having a large part of the manufacturing complete and all of the data collected, the design proposal phase report was able to be written for submission to competition judges.

Aircraft Construction: The longest and most important process of the entire project, having started in December, will commence with the test flights of our plane in late March.

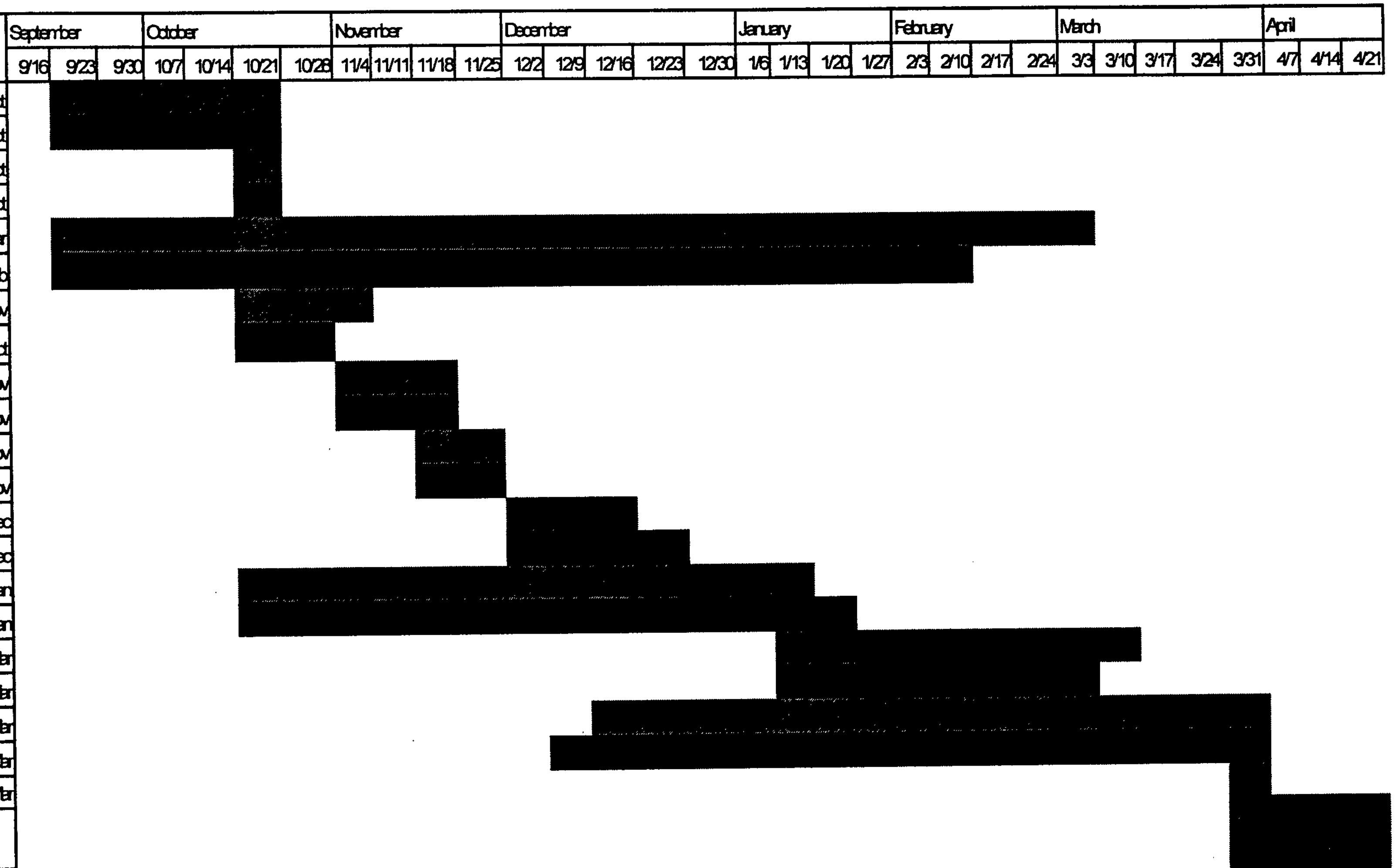
Test Flight(s): Having completed construction, test flights are vital to ensure flyability, structural integrity under true loads, and good component interaction, as well as serving as practice for the pilot.

Pit Crew Selection and Coordination Drills: Since the competition will be timed, having a good "Pit Crew" that is coordinated in thoughts and procedures in different scenarios to unload our cargo will be imperative to scoring well and winning the contest.

2.1.2 M[illegible]stone Chart

Figure 2.1.2 Mi[illegible]one Chart

ID	Task Name	Start	Finish
1	Assembly of Design Team	21-Sep	17-Oct
		21-Sep	17-Oct
2	Notice of Intent to Complete	19-Oct	19-Oct
		19-Oct	19-Oct
3	Obtain Funding	22-Oct	4-Mar
		22-Oct	10-Feb
4	Concepts and Preliminary Design FOMS	20-Oct	2-Nov
		20-Oct	28-Oct
5	Conceptual Design	2-Nov	14-Nov
		2-Nov	14-Nov
6	Preliminary Design	15-Nov	29-Nov
		15-Nov	29-Nov
7	Data Design	30-Nov	19-Dec
		30-Nov	23-Dec
8	Aquire Materials and Parts	22-Oct	14-Jan
		22-Oct	20-Jan
9	Proposal Phase	19-Jan	9-Mar
		19-Jan	3-Mar
10	Aircraft Construction	16-Dec	30-Mar
		3-Dec	30-Mar
11	First Test Flight	31-Mar	31-Mar
12	Plt Crew Selection/ Coordination Drills/ Test Flights	31-Mar	21-Apr



Key: Grey = **Proposed duration of task,** Blue = **Actual duration of task**

3 Conceptual Phase

3.1 Stage I – Feasibility Analysis

Table 3-1 Design Parameter FOM analysis – Stage I

(-) - Not Applicable
(r) - Rejected
(K) - Kept

Design Parameters – Stage I		Figures of Merit – Stage I (x weighting)										
		Ease of Manufacture (x2)	Material Availability (x1)	Strength to Weight Ratio (x2)	Handling Characteristics (x1)	Rated Aircraft Cost (x1)	Cost (x1)	Motor Power Limitations (x1)	Thrust Ratings (x2)	Motor Installation Flexibility (x1)	Sum of Ratings	Decision
Wing Configuration	Low	1	-	-	0	0	0	-	-	-	2	K
	Mid	-1	-	-	0	0	0	-	-	-	-2	r
	High	1	-	-	0	0	0	-	-	-	2	K
	Bi-plane	0	-	-	0	-1	0	-	-	-	-1	r
Wing Planform	Delta	-1	0	-1	-1	-1	0	-	-	-	-6	r
	Elliptical	-1	0	1	1	1	0	-	-	-	2	r
	Tapered	1	0	1	1	0	0	-	-	-	5	K
	Rectangular	1	0	1	1	0	0	-	-	-	5	K
Airfoil	E214	0	0	-	-1	0	0	-	-	-	-1	r
	E211	0	0	-	-1	0	0	-	-	-	-1	r
	E212	0	0	-	1	0	0	-	-	-	1	K
Structural System	One Piece	-1	-	0	-	-	-	-	-	-	-2	r
	Two Piece	0	-	0	-	-	-	-	-	-	0	K
	Three Piece	0	-	0	-	-	-	-	-	-	0	K
	Strut Braced	-1	-	0	-	-	-	-	-	-	-2	r
	Bending Beam	0	-	1	-	-	-	-	-	-	2	K
	Ring Frame	-1	-	1	-	-	-	-	-	-	0	K
Tail Structure	Conventional	1	-	-	1	0	-1	-	-	-	2	r
	Canard	-1	-	-	0	-1	-1	-	-	-	-4	r
	T-tail	1	-	-	1	0	0	-	-	-	3	K
	Cruciform	0	-	-	0	0	0	-	-	-	0	r
	V-tail	1	-	-	-1	0	0	-	-	-	1	r
	Inverted V-tail	1	-	-	1	0	0	-	-	-	3	K
	H-tail	-1	-	-	1	-1	-1	-	-	-	-3	K
Fuselage Structure	Trusses	-1	-	0	1	-	1	-	-	-	0	r
	Monocoque	1	-	-1	-1	-	0	-	-	-	0	r
	Semi-monocoque	0	-	1	1	-	0	-	-	-	3	K
Landing Gear	Tail-Dragger	1	-	-	1	-	-	-	-	-	3	K
	Tricycle	0	-	-	1	-	-	-	-	-	1	K
	Bicycle	-1	-	-	0	-	-	-	-	-	-2	r
	Quadracycle	0	-	-	-1	-	-	-	-	-	-1	K
Power Plant	1 motor 19 cells	-	-	-	-	-	-	1	-1	1	0	r
	1 motor 22 cells	-	-	-	-	-	-	0	1	1	3	K
	2 motors/20 cells	-	-	-	-	-	-	1	0	0	1	K
	2 motors/22 cells	-	-	-	-	-	-	0	0	0	0	r

The purpose of Stage I was to apply FOMs to eliminate and analyze concepts for aircraft components that were proposed during brainstorming sessions at the beginning of the design process. The goal was to produce a set of desirable components to incorporate into three conceptual aircraft designs. The Stage I FOMs consisted of five primary areas. First, material availability required that any material not commercially available or unreasonably expensive be ruled out. Second, handling characteristics were used to eliminate concepts that were initially thought to have poor ground or flight performance. Third, designs that did not promote ease of transportation and/or manufacture were dismissed from consideration. The fourth consideration for Stage I was simplicity and tooling. Simple structures are easier to analyze and build without necessarily losing functionality or performance characteristics. These guidelines established a solid set of conceptual components to consider for the aircraft.

3.1.1 Airfoil

When choosing an airfoil, the lift to drag ratio was the most important feature to be considered. For the final design, an aircraft constrained by take-off distance, an airfoil that maximized lift while minimizing drag was necessary. Three airfoils (Eppler 211, 212, and 214) were analyzed in order to obtain their respective coefficients of lift and drag. We chose the airfoil Eppler 212 over the two others due to its desirable lift and drag coefficients. When calculating the C_L and C_D coefficients, a Reynolds number of 200,000-300,000 was considered, which was determined by an average cruise velocity of 65 MPH.

The maximum lift and drag coefficients calculated for airfoil Eppler 212 were 0.79 and 0.0142, respectively. The maximum lift and drag coefficients calculated were 0.78 and 0.012, respectively, for airfoil Eppler 214. For the E212, the maximum lift and drag coefficients were found to be 1.544 and 0.023104, respectively. While the other Eppler airfoils are low drag sections that would be better than the E212 for cruise performance, they are predicted to generate insufficient lift to achieve takeoff in 200 ft given power limitations imposed by contest rules. The lift generated by E212 section was determined to be sufficient for takeoff within 200 feet distance limit and was chosen as the plane's airfoil.

3.1.2 Wings

The wings are a primary defining characteristic of an aircraft. In this year's design, they were optimized to the role of the aircraft's flight objectives. Other primary, secondary and tertiary structures were designed around the general wing structure. In this design process, there were limitations imposed by contest rules, flight conditions, and structural feasibility. The initial design parameters being investigated were wing configuration and wing planform. The available wing configurations included a high and low placement mono-wing and a bi-plane configuration. The considered wing types were delta, elliptical, tapered, and rectangular. The intent of the Stage I wing analysis was to eliminate design parameters based on FOMs to determine their applicability to the design goals. FOMs that were used consisted of handling characteristics, RAC, ease of transportation, and complexity of manufacture.

3.1.2.1 Wing Configuration

The wing configurations presented as options (low and high mono-wing and bi-plane) represented the traditional configurations that were well documented for planes of our size. One of the basic limitations was that the payload was to be carried within the fuselage. Bi-plane configuration would have caused an array of problems with payload access and would have increased the planes RAC thus was ruled out of consideration. The high and low wing configurations would not pose a payload problem and were therefore left for further analysis.

3.1.2.2 Wing Planform

Delta: A delta wing was eliminated immediately as a design possibility due to the fact that it is optimized for transonic and supersonic flight.

Elliptical: Elliptical wings, which feature a constantly varying chord length, were eliminated due to their difficulty of manufacture and increase of the RAC.

Rectangular: The rectangular wing satisfied the initial set of FOMs and was selected for further analysis in Stage II.

Tapered: The tapered wing also satisfied Stage 1 FOMs and will be further analyzed in Stage II.

3.1.3 Empennage

The tail surfaces of TLAR III will be referred to as empennage, and are an essential part of an aircraft's control and stability.

Existing empennage configurations were selected from documented styles (stage 1 – feasibility analysis). Each was analyzed based on the experience of the team members gained in past year's competitions and through application of basic FOMs. The goal of the first stage of the CDP was to complete the analysis of proven empennage configurations so as to select three designs to investigate further in the second stage.

The purpose of empennage is to provide stability and pitch/yaw control. The three main FOMs were handling characteristics, RAC, and ease of manufacture. Other FOMs, such as material availability, ease of transportation, and reparability were not considered explicitly in this section, as the materials of all designs were understood to be the same, and the types of tails were assumed to be equally transportable and repairable.

3.1.3.1 Tail Configuration

Canard: A canard configuration would reduce the required control surfaces at the rear of the aircraft to a single vertical stabilizer. The benefit of a reduction in control surfaces at the rear, however, would be negated by the complexity of both pitch and roll capabilities integrated into the aileron control. While this configuration had benefits that included main wing stall prevention and less control surface actuators, the design, analysis and fabrication was complex and was eliminated for these reasons.

Conventional: Major benefits of a conventional design were ease of construction and lightweight design. However, the low placement of the horizontal stabilizer usually puts it in the wake of the main wing "washing out" the airflow over the horizontal stabilizer and rendering elevator control ineffective. This configuration was eliminated due to its poor efficiency, susceptibility to damage and loss of control.

"T": The most important benefit of the "T"-Tail design is that the horizontal stabilizer is above the turbulence from the body and main wing, thus allowing the horizontal control area to be reduced. It also acts as an endplate for any control deflections caused by the vertical fin, which increases the vertical fin's efficiency and allows a reduction in the fin's area. This allowed a large reduction in weight and the "T" style tail was deemed the first design to be considered in Stage II.

Cruciform: The fourth concept examined was the cruciform design, a hybrid of the Conventional and the "T"-Tail. It was determined that this style shared several benefits of both, but was difficult to construct and the end plate effect from the "T" was lost. For those reasons, this design was eliminated.

"V": The next configuration investigated was the "V"-Tail design, which was efficient because it eliminates one wing tip vortex, decreasing induced drag. The design was discarded due to its many disadvantages, including high control surface complexity, and an induced opposite roll when given rudder input.

Inverted "V": The inverted "V", by contrast, produces favorable yaw-roll coupling. Elevator control was less affected by turbulent wake from the fuselage and main wing, and the use of only two control surfaces reduced the number of servos needed for both rudder and elevator control. This design had bad ground clearance and complex control surface design. However, it was kept for further analysis because of the good turning characteristics and the reduced drag of one less wing tip vortex.

"H": This was deemed a good design, as turbulence from the fuselage does not flow around the vertical fins. The vertical fins also prevent spillover of the elevator with an end plate effect, therefore eliminating drag from tip vortices induced by elevator input. Although this design is complex, it was kept for further analysis as it worked well with a twin-body design.

3.1.4 Fuselage

The fuselage serves as a primary structure of the aircraft and acts as a critical load path for the aircraft to transmit forces between the tail and wings, in addition to housing flight equipment and payload. Design of the fuselage is somewhat dependent on the design of the wing and tail because it links the components. The FOMs that were used to optimize the design parameter of structural reinforcement were ease of manufacture, strength to weight ratio, handling characteristics, and cost. The four types of fuselage structural reinforcement investigated were trusses, monocoque, semi-monocoque, and box-beam.

3.1.4.1 Structural Reinforcement

Truss: While a truss structure in the fuselage would have a relatively good strength to weight ratio, it may interfere with the payload volume. Trusses are very difficult to manufacture, so it was quickly dismissed as a means of reinforcement in the fuselage.

Monocoque: A monocoque design has good torsional strength, but it is weak in bending, which is the type of load transferred by the tail to the fuselage. The required skin thickness needed to support these bending loads would increase the weight significantly and therefore it was eliminated as a design parameter.

Semi-Monocoque: A semi-monocoque structure combines the torsional strength of the skin with bending strength that can be customized for the aircraft by means of stringers and ribs. This type of

configuration allows the strength to be customized for specific loads. However, due to the positioning of the cargo, a semi-monocoque structure would be too large and not feasible.

Box Beam: A box-beam structure utilizes the fuselage's skeletal frame for structural strength. This frame would be made of a carbon/foam/carbon sandwiched material. This material would act much like steel "I" beams of buildings act. This box structure could sufficiently accommodate the torsional forces induced during flight thus was chosen for further analysis.

3.1.5 Landing Gear

The functional requirement of the main landing gear is to absorb and dissipate landing loads effectively. The design parameters considered included: tail dragger, quadracycle, bicycle and tricycle. During Stage I of the CDP various landing gear configurations were screened according to the established FOMs, such as ease of manufacture and ground handling characteristics.

3.1.5.1 Landing Gear Configurations

Bicycle: The bicycle landing gear configuration consisted of two in-line wheels straddling the CG of the fuselage and another wheel on each wing for balance. Due to complexity of manufacture and poor ground handling characteristics, the bicycle design was immediately eliminated.

Tricycle: The tricycle landing gear has two main wheels aft of the CG and an auxiliary wheel forward of the CG. The advantage of this arrangement is that it is stable and the aircraft can be landed at a large "crab" angle. Also, the tricycle landing gear allows for a level fuselage which is helpful when loading the softballs. The tricycle will be kept for further analysis.

Tail dragger: This design would place two main wheels forward of the CG, and a small auxiliary wheel attached to the boom directly below the empennage. This configuration would be sufficiently stable on the ground while giving the propeller extra ground clearance. The tail dragger will be kept for further analysis.

Quadracycle: This configuration would be used in the event that we choose the dual fuselage body structure. It would be very stable on the ground due to it's two wheels forward the CG along with one wheel placed under each of vertical stabilizers on the "H" style empennage. The quadracycle will also be kept for further analysis.

3.1.6 Propulsion and Power System

The propulsion design began with little knowledge about the overall configuration of the aircraft, so the initial decisions related to propulsion choices were based on experience gained from last year's project.

The amount of power available to propel the aircraft is dependent upon the total power of the system and the efficiency of the entire propulsion system. To maximize this number, given that the battery power was limited by weight restrictions, a maximally efficient engine/propeller combination was chosen. Optimizing the efficiency of the motor as well as motor configurations while lowering the REP was the goal of the propulsion design portion of the conceptual phase.

The battery weight restriction of 5 lbs. limited the number of battery cells to 38, which amounted to approximately 40 volts. For standard radio controlled propulsion systems, 40 volts exceeded the rated

power of many motor possibilities. While a motor's power is related to the input voltage, they can normally withstand higher voltages. Problems experienced in the competition two years ago showed that this was not a desirable arrangement, yet it was determined that a single motor design was still a valid concept.

Single-Motor: Initially, due to concerns with overpowering the single motor, the single motor configuration was developed with a maximum of 19 cells connected to the motor. However, due to the lower thrust of this configuration, the single motor connected to 22 cells was considered instead. This design was left for analysis in Stage II.

3.1.7 Structural System

The structural system was required to provide efficient "load paths" which resolve forces experienced during landing and flight sequences. The spar structure, which reinforces the wings, transmits the bending loads through the fuselage, is the major component of the structural system. When considering viable structural configurations it was noted that several options placed constraints on other component design solutions. These affected components included the wing platform (swept and tapered), wing configuration (biplane, delta and monoplane) and fuselage parameters. Several of these concepts were analyzed during Stage I of the CDP to effectively meet the various requirements of the structural system. Various spar cross-sectional geometric shapes were also considered for each of the competing component designs. These included the box beam, hollow cylindrical, solid rectangular and I-beam cross-sections.

The primary objective of the Stage I screening process was to eliminate design concepts that were complex, raised the RAC, and/or were difficult to repair or replace. The one structural system design idea that remained following the application of the Stage I FOMs follows:

Spar Structure: Initial concepts that were considered were the one and two piece spars. The one-piece spar was chosen.

3.1.8 Conceptual Design Phase – Stage I Summary

The CDP Stage I was focused on the development of initial aircraft designs that feature various component concepts that were determined to have merits applicable to the mission goal. By the end of the first stage, concepts that were immediately discernable as inadequate had been eliminated. Any remaining concepts were combined into several comparable configurations. These configurations would be evaluated further in the second stage of the CDP as to their individual component interactions and total system performance.

The decisions made when combining the components into complete configurations were somewhat arbitrary. However, each component that survived the first stage was examined to determine its ability to interact with other components. For each component, this process was done to piece compatible components together and develop three aircraft configurations.

First configuration, featured a single fuselage mounted over a low mono-wing. The tail dragger concept was considered for this model with an advantage of propeller clearance. However, the unstable nature of the tail dragger design made the team skeptical of this configuration.

Second configuration employed a single fuselage with a high mono-wing concept. The high wing necessitated the use of the tricycle landing gear concept and allowed, due to the height of the fuselage, an inverted "V" tail mounted to a tail boom. The single motor mounted to the front of the fuselage also required that the payload access be from the rear of the fuselage.

Other designs consisting of "hybrids" of the above mentioned configurations will most likely yield more and better designs that will incorporate the best aspects of each into the most efficient, lowest cost, best performing design desired. These concepts will be taken as the configurations examined in the second stage of the CDP. There, they are to be analyzed with regards to component interaction and initial flight characteristics while further determination of RAC and ease of manufacture will be made.

3.2 Stage II – Generalized Analysis

Table 3-2 Design Parameter FOM Analysis – Stage II

(-) - Not Applicable

(r) - Rejected

(K) - Kept

Design Parameters – Stage II		Ease of Construction (x2)	Component Safety (x1)	Component Safety (x1)	Component Interactions (x2)	Sortie Performance (x2)	Difficulty of Repairs (x1)	Rated Aircraft Cost (x1)	Cost (x1)	Motor Aerodynamic Flow (x2)	Motor Access/Mounting Ease (x1)	Sum of Ratings	Decision
		Figures of Merit – Stage II (x weighting)											
Wing Types	Low Mono-wing	1	-	-	1	0	0	0	0	-	-	4	K
	Bi-plane	-1	-	-	-1	0	0	0	0	-	-	-4	R
	High Mono-wing	1	-	-	0	0	0	0	0	-	-	2	R
Wing Structure and Materials	Rib Structure	-1	0	-1	-1	0	-1	-	1	-	-	-5	R
	Foam Core	1	0	0	0	0	-1	-	0	-	-	1	K
Structural System	One Piece	1	-	1	1	-	0	0	-	-	-	5	K
	Three Piece	0	-	0	-1	-	0	0	-	-	-	-2	R
	Bending Beam	1	-	-	1	-	0	0	-	-	-	4	K
	Ring Frame	-1	-	-	-1	-	0	0	-	-	-	-4	R
Tail Structure	Twin Body H	-1	0	0	-1	0	-1	0	0	-	-	-5	R
	Inverted V-tail	0	0	-1	0	0	0	0	-1	-	-	-2	R
	T-tail	1	0	0	1	1	0	0	0	-	-	6	K
Fuselage Configuration	Flying Wing	-1	-	-	-1	0	-1	1	0	-	-	-4	R
	Lifting-Body	0	-	-	1	1	1	-1	0	-	-	4	K
Fuselage Materials	Fiberglass	0	-	-1	0	1	1	0	0	-	-	2	R
	Aircraft Grade Plywood	-1	-	-1	0	1	0	0	1	-	-	0	R
	Kevlar	0	-	1	0	-1	1	0	-1	-	-	-1	R
	Carbon Fiber Reinforced Kevlar	-1	-	1	1	1	1	0	-1	-	-	3	K
Landing Gear	Tail-Dragger	0	-	1	0	0	0	0	0	-	-	1	R
	Tricycle	0	-	1	1	0	0	0	0	-	-	3	K
	Quadracycle	-1	-	-1	1	0	-1	-1	-1	-	-	-4	R
Power Plant	1 motor 22 cells	-	-	-	1	-	-	1	1	0	1	6	K
	2 motors 11 cells each	-	-	-	0	-	-	-1	-1	0	-1	-2	R

Stage II of the CDP was necessary to obtain our final aircraft characteristics. This stage rendered the designs that promoted a finished product, which in turn satisfied all of the design parameters. FOMs for Stage II included ease of construction, payload access and capacity, strength to weight ratio, difficulty of repair, component and interfacing strength, and component/flight safety. The application of these FOMs to specific design parameters resulted in a final set of components that, when assembled, produced an aircraft design with all of the desired characteristics.

3.2.1 Wings

Wing FOMs in Stage II focused mainly on structural complexity and construction. The FOMs were applied to the determination of the best wing type.

3.2.1.1 Wing Type

Bi-wing: The bi-wing implemented failed to satisfy the ease of manufacture FOM as it required the fuselage to be reinforced in multiple places and had the possible necessity of external bracers. The increased difficulty of manufacturing, increased cost, and additional induced drag thus eliminated it as a design concept.

High Mono-wing: High mono-wing design carried the benefit of large ground clearances for propellers (for wing mounted motors). It was however a major challenge in terms of designing fuselage mounted landing gear, which would not interfere with the payload. This was therefore eliminated due to the structural stipulations.

Low Mono-Wing: Low mono-wing configuration demonstrated to be effective in both TLAR I and TLAR II. Landing gear length is minimized, allowing for lighter and/or stronger struts. Torsion load transfer can be accomplished with limited interference with fuselage layout. The possibility of low propeller clearance was shown to be inconsequential due to the nose mounted motor which yielded several extra inches of clearance. This configuration was chosen for further analysis and development.

3.2.2 Empennage

The goal for the second stage of the CDP of empennage was to analyze the interaction of the surviving Stage I conceptual designs as combined with the rest of the aircraft. The primary focus areas were the structural and aerodynamic dependencies and interactions with other components in the design under the application of further FOMs. The sorties that TLAR III will fly consist of steep climbs, tight turns and short take-off and landings.

"Twin Body H": In the conceptual design "Twin Body H" configuration was implemented. While the construction was viewed as reasonably simple and sturdy, several disadvantages were immediately apparent. This design required having two fuselages and an additional servo, both of which significantly increase the RAC and add to the aircraft weight. While the structural and aerodynamic properties of this design were favorable, the added cost to the overall weight and RAC of the aircraft was too much to consider it further for the final design.

Inverted "V": A tricycle landing gear and high tail boom, provided the opportunity to apply the "Inverted V" configuration due to the fact that it would have enough ground clearance. Aerodynamically, the concept was free of major concerns related to interaction with the fuselage and main wing. While the aerodynamic and structural constraints were satisfied by this configuration, there were distinct drawbacks related to the complexity of the control design and the ease of manufacture. This increased complexity eliminated the possibility of an Inverted "V" configuration.

"T" Tail: T-tail configuration prevents tail surface washout and related loss of stability, the high placement of the horizontal stabilizer was essential. The fact that this configuration has been proven effective in the flight competition in past years while still being simple and efficient, led to the decision that it would again be the tail configuration of choice for TLAR III.

3.2.3 Fuselage

Stage II fuselage design involved the evaluation of the fuselage design possibilities within the various configurations. The FOMs used to ensure that the design goals were met included ease of construction, payload access, RAC and component and interfacing strength. The two aircraft configurations considered Blended Wing and Lifting Body. An analysis of the characteristics of these three styles was necessary to decide which will best satisfy the FOMs.

Blended Wing: first design featured a fuselage blended into a wing thus providing a good drag characteristics. Also there is no empenage thus provides a better RAC because of time saved in not making those components. One major concern arose in this design that lack of tail empenage provides a poor stability thus can lead to increased risk of crashing the plane.

Lifting Body: second design style featured a single fuselage that would house the payload and flight equipment. This configuration was considered because it provides good airflow around the fuselage, maintains interfacing strength and would not affect the RAC drastically. In addition, efficient access to the payload is granted via a hatch located at the end of the fuselage. This design meets all the FOMs and was thus considered a viable design option.

3.2.4 Landing Gear

In Stage II of the CDP, advantages and disadvantages of each remaining landing gear configuration were explored as they related to specific design choices made for other components of the aircraft. The three major considerations focused on were; ground handling, plausibility of adding brakes, and minimization of weight and drag during flight.

3.2.4.1 Landing Gear Configuration

Tricycle: Tricycle arrangement was implemented because of the additional stability due to the center of gravity is ahead of the main (rear) wheels, the aircraft would be inherently more stable on the ground and can be landed at a fairly large "crab" angle. While this configuration required one extra gear block and steering servo for the nose landing gear, which added more weight to the design and increased the RAC, the increased ground stability and ability to add brakes to the rear wheels was a desirable perk. Having brakes would enable shorter landing roll time thus bettering the flight scores. This concept therefore satisfied the Stage II FOMs and was chosen as the landing gear configuration for TLAR III.

Quadra-cycle: This configuration has good ground handling stability and an improved strength to weight ratio due to the lack of struts. However, two wheels require two servos for steering ability, thereby increasing the RAC greatly. Due to these unnecessary RAC increases, the Quadracycle was eliminated as a viable configuration.

Tail Dragger: The tail dragger landing gear configuration provided ample propeller clearance by angling the nose off the ground. In addition, its simple configuration of connecting the tail wheel to the same servo that controls the rudder would make the aircraft more easily controlled on the ground. While this dual use for a servo provided an excellent reduction in RAC, the design had structural strength concerns due to the thin rear wheel strut being under increased plane and payload weight. Therefore the tail dragger design was eliminated as being a configuration for further analysis.

3.2.5 Propulsion and Power System

The three configurations developed in Stage I of the CDP provide a basis for the propulsion requirements that needed to be met. At this stage however, thrust or specific efficiency requirements were not determined. Instead, FOMs such as power limitations of the motor and a low REP were applied to the design parameters such as number of motors and number of batteries per motor. This was similar to the FOM/design parameter process applied during Stage I of the conceptual phase, except that here it was applied in parallel, with analysis of the interaction with the other components in the design. This was accomplished by applying FOMs relating to aerodynamic flow of the thrusting air, ease of construction and motor/controller cooling to determining the final rating of each design.

Single Motor: The single engine configuration was the only configuration seriously considered for the design of the aircraft. Even though multiple motors result in better trust and power, the penalty incurred for this option is extremely high. This penalty would also factor into the total number of batteries being used to power the aircraft motors. For this reason the total weight of the battery pack must be kept to the bare minimum.

Summary: In order to minimize the RAC of the aircraft the only acceptable form of power system would have to be a single motor configuration. This configuration would also have to be powered via a single battery pack with minimum weight. The propellors, will have to be optimized for cruise speed, as this will be crucial in reducing the mission completion time. Further analysis of all these configurations will be required to conclude if a gearbox will be necessary.

3.2.6 Structural System

The purpose of Stage II for the structural system was to evaluate the individual strength as well as the interface strength within the various configurations. The Stage II FOMs that were pertinent to the structural system were RAC, component interaction and ease of construction. Through the application of these FOMs the individual structural components remaining from Stage I are evaluated. The structural system components were evaluated as follows:

3.2.6.1 Spar Structure

The shear and bending stress increases approaching the fuselage interface, therefore in the interest of weight conservation, the size of the spar structure could be made to vary. In all CDP aircraft configurations, it was also determined that torsional loads were a major concern, needing to be addressed by the design of the spar structure.

Two piece: The two piece spar model consisted of two separate spar lengths, one from the wing tip to half of the wing span, and the other continuing to the wing's root. The exact size would be determined by the existing stresses due to the shear and moment. However, the required torsional strength was not sufficient enough to consider this design further.

One piece: The one piece model is a solid, single-piece spar running the length of the wing. The size, and therefore weight, of the spar could be further reduced in respect to multiple piece spars. An additional benefit of the One Piece was its ability to accurately maintain the wing angle of attack. This model satisfied the Stage II FOMs and thus was considered a feasible spar design.

3.2.7 Conceptual Design Phase – Stage II Summary

The combined Conceptual Stage I and Stage II had the goal of determining a final configuration for the aircraft design. For each possible component of the configuration: airfoil; wing; tail; fuselage; landing gear and power supply, design parameters were defined and then eliminated through the application of FOM to determine the best final choices for the preliminary aircraft design.

The result of this process was the aircraft designated as "TLAR 2.9", a design very close to the final design, "TLAR III", though lacking final sizing and optimization of the components. "TLAR 2.9" features a single aerodynamically faired cylindrical fuselage riding over a low mono-wing. The payload access, through the nose of the fuselage, was facilitated by the symmetric mounting of two motors to the leading edges of the wings. Similar to last year's design was the implementation of a "T" tail configuration on a tail boom extending from the center of rear of the fuselage. This lengthened design allowed the use of the tail dragger landing gear to increase the lateral stability of the aircraft on the ground. Overall the preliminary design was simple and efficient and reflects the great effort put forth in the conceptual design to determine the best choice for each concept.

4 Preliminary Design Phase

As the design of the aircraft progresses, estimates for actual design specifications need to be made. The PDP of the design focuses on completing the initial calculation and estimation of general sizing, performance, and configuration with regards to the final aircraft design created in the second stage CDP.

During this process, techniques developed for the design of propeller driven general aviation aircraft were used to a large extent. Standard aerodynamic theories, such as Prandtl's lifting line method, were also implemented in computer algorithms to analyze the aerodynamic properties of the wing. While these techniques are not expected to be exact, they were implemented with confidence as TLAR II validated their accuracy in last year's competition. Overall, these proven methods provided excellent analysis results while directly exposing the design team to the fundamental principles of aircraft design.

As the CDP was highly focused on determining design parameters and FOMs to apply to the design process, so was the PDP. Here, however, the considerations that need to be made are more related to the optimization of flight performance and aircraft component interaction, than with the broader concerns examined in the previous sections. For each area of the preliminary phase, design parameters and FOMs will be applied as suit that particular area, and will be described specifically. Overall, we are still concerned with component and personnel safety, structurally and aerodynamically favorable component interaction, ease of manufacture and construction, and a low RAC.

4.1 Wings

The wings PDP deals with the general sizing of the wing and associated components as well as materials selection. Wing geometry was chiefly governed by necessary flight characteristics and optimized by RAC concerns.

Materials: Strength to weight ratio was a primary FOM in this stage, as was cost. The goal was to maximize the strength to weight ratio, while also considering the implications to construction. Aircraft grade plywood is relatively cheap and lightweight. Manufacturing a wing with this material, however, is a complicated task. The wing would have to be made out of a series of ribs, each of which had to be individually assembled and shaped. The difficulty of this fails to satisfy the ease of construction FOM. A composite wing holds several advantages over the plywood wing. It was easier to construct and was considerably stronger while being comparable in weight. The structure and procedure were established by TLAR II in the 2001 competition. Composite wings are somewhat more expensive than plywood, but the increased ease of construction resulted in their selection for the final wing structure.

Wing Geometry: The rectangular wing planform selected out of the CDP required optimization to reduce unnecessary drag and maximize lift and stability. Many geometric characteristics either had an adverse impact on performance or simply were not applicable to a model aircraft. Adaptations such as sweep and wing twist were not considered due to complexity of design and manufacture.

The wing sizing was begun with a desired aspect ratio of at least 8. Since there is no wingspan limitations, it was necessary to design the wing to not just perform well, but also reduce RAC as much as possible. In short, it is easy to over-design a wing that will produce lots of lift, but it might not be fully

optimized. After analysis of the power system and an estimate of the fully loaded aircraft, and 84" wingspan with a 13.5" chord length was decided upon. It was also decided that no taper would be introduced in the design of the wing. Although a taper tends to reduce the wing tip angle of attack during high G maneuvers, thus unloading the wing, it also reduces the wing's area, therefore reducing the lift attained from the wing. Because of the way the RAC of the aircraft is calculated, it would be more advantages to have a wing without any taper. Finally, the wings are connected to each other. This significantly makes construction easier by eliminating the need for a wing joiner.

Stability: Documented performance characteristics of the chosen platform and low wing configuration led to some concern over roll axis stability. The low wing configuration creates a pressure buildup during a banked turn due to side-slipping (the fuselage interferes with airflow over the upwind wing). This causes a tendency for the aircraft to roll into a turn as the leading wing loses lift. To compensate for this, a dihedral angle was introduced to roll the aircraft back to its stable flight position. Due to the lack of a simple method to calculate the appropriate dihedral angle, empirical data was used to determine that, for radio controlled aircraft design, a 3° dihedral is most effective.

Control Surfaces: The type and location of control surfaces were dependent, in part, upon the rest of the primary aircraft structures. The available options included flaps, ailerons, or a hybrid control surface. Achieving the necessary level of control was the principal concern, followed by the impact to the RAC and building complexity.

Flaps were considered to increase lift during takeoff in order to minimize the takeoff distance. However, the inclusion of flaps would add significantly to the RAC. Spoilerons were chosen in place of ailerons to ensure control of the rolling/turning of the aircraft at low airspeeds. They are hybrid structures combining spoiler effects with aileron control surfaces. They help the flow stay attached to the wing during slow, high AOA flight, ensuring that the ailerons do not stall before the main part of the wing. They can also be used after touch down to reduce the lift of the main wing and ensure that the aircraft stays on the ground. In using spoilerons, the RAC penalty of the flaperons is avoided. Typical sizing of an aileron structure is approximately 25% of the span, extending to the outer edge of the wing. Thus our initial control surfaces were each 21 inches long.

4.2 Empennage

The preliminary phase of the Tail design focused on the initial determination of the sizes of the vertical and horizontal members, the size of the control surfaces, and the placement of the tail surfaces away from the aircraft's CG. Other design parameters considered at that point in the design were material choices, and manufacturing techniques. The main FOMs that were applied to these design parameters were ease of construction and stability.

General Sizing: The Tail sizing calculations used were a combination of general aviation and R/C aviation empirical data relations. Specifically to design the volume coefficient of both the horizontal and vertical members as well as the size of the elevator and rudder control surfaces.

The sizes of the tail components are functions of their distance from the CG of the aircraft and main wing size. Empirical data gives a function for tail volume coefficient, which is the standard method of approximating initial tail size, dependant on these parameters (equations shown below). Modifications to this empirical relationship for a T-tail allowed the vertical coefficient to be reduced by approximately 5%

due to end plate effect of the horizontal member. Also, with the engines mounted on the leading edge of the fuselage, the tail arm was given to be roughly 50% of the fuselage length as measured from the mean aerodynamic chord of the main wing.

Typical values used for the volume coefficients depend on the type of aircraft and, from experience for homebuilt aircraft, are approximated as 0.5 and 0.03 for the horizontal and vertical respectively. With these available parameters, and the size of the aircraft configuration at this stage giving an arm of 31 in, the initial areas of the tail sections were determined to be 237 in^2 for the horizontal and 96 in^2 for the vertical.

$$Area_{HorizontalStabilizer} = \left(Coefficient_{HorizontalVolume}\right)\left(MAC_{MainWing}\right)\left(Area_{MainWing}\right)/\left(Arm_{HorizontalStabilizer}\right)$$

$$Area_{VerticalStabilizer} = \left(Coefficient_{VerticalVolume}\right)\left(Span_{MainWing}\right)\left(Area_{MainWing}\right)/\left(Arm_{VerticalStabilizer}\right)$$

An empirical approach was utilized in the preliminary sizing of the tail control surfaces. Guidelines derived from empirical data suggest that the rudders and elevators be approximately 25% of the tail chord, while extending from the tail boom to about 80-90% of the tail span. Given these areas and control surface relationships, as well as an Aspect Ratio and Taper Ratio of 3.4 and 0.92 proven successful for TLAR II, the horizontal stabilizer was determined to need a span, root chord, tip chord and elevator chord of 26.0, 9.5, 8.75, and 2.0 inches respectively. With the Aspect Ratio and Taper Ratio of 1.59 and 0.76, again from TLAR II, the vertical stabilizer was determined to need a span, root chord, tip chord and elevator chord of 12.0 in, 8.50 in, 7.0 in, 2.5 in. respectively.

Manufacturability: Similar to the main wings, FOMs such as ease of construction and strength to weight ratio must be applied to all material choices. One technique considered was a rib/spar structure of aircraft ply, which is lightweight and strong, but again very complex and time consuming to manufacture. The materials that were finally chosen fitted well with the most simple of available manufacturing techniques which was foam core, carbon spar, and fiberglass skin structure. These materials provide excellent strength to weight ratios and the ability to manufacture the components quickly with tools already available from the construction of TLAR II.

4.3 Fuselage

Optimization of the fuselage included determining a viable support structure and the necessary dimensions for maximum payload capacity. In order to determine the necessary dimensions for the fuselage a packing structure for the softballs was needed. The FOMs that were considered pertinent during this design phase were interfacing and component strength, weight and payload access.

A satisfactory packing arrangement was found following several attempts to find an arrangement that was not too long or too wide. This arrangement consisted of a rectangular organization of 4 softballs wide and 6 softballs long giving a total of 24 softballs (See Drawing Package). With the softballs packed as tightly as possible this resulted in a cargo loading area of 16x24 inches. However this rectangular box sitting on top of the wing would create much undesirable drag. In order to mitigate this problem, a symmetric airfoil (NACA 0013) would be designed to overlay the cargo ball array.

In order to avoid payload access problems and to accommodate the batteries and flight equipment, the fuselage dimensions would have to be slightly larger. This was not a problem, as the

thickest part of the symmetric airfoil allowed for ample room for these accessories. The room created also accommodates a support structure for the wing and boom interfaces. Working closely with the structural system group allowed for mutually beneficial solutions.

Box Beam Structure: This arrangement describes a box type structure designed to be able to withstand shear and torsional stresses. Since the top skin of the fuselage would be breached to access the payload, the fuselage needs to have an internal structure designed to withstand the transfer of forces and moments from the tail and wing. Similar to a drawer from a cabinet, in essence the structure of the fuselage is an open-topped box that is able to resist shear and torsional forces. This torque box consists of ball-floor, airfoil-shaped sides, lower skin, and wing. The walls of this box would have to be very strong, yet very light. This introduced the idea of "Sandwich Material" for this design. "Sandwich Material" consists of two very stiff but thin layers of carbon, which surround a slightly thicker layer of anisotropic (stronger in one dimension than the other) foam or balsa. This creates an I-beam effect, resulting in great strength-to-weight ratio.

Final Sizing: The final size of the fuselage was determined to be 39.5" in length, 16.25" in width and 5.5" in height at its thickest point. The box structure itself (entirely made of "Sandwich Material") consists of two side walls in the shape of a symmetrical airfoil. These two side walls are then connected together with a 39.5"x16.25" bottom with circular cutouts for where softballs can be placed. This structure is then reinforced with three bulkheads placed 4.25, 27.5, and 31.5 inches from the leading edge of the side walls. The wing sits directly beneath the latter 2/3 of the softballs, as there is an airfoil cutout for which the wing fits into. Hard-points on both wing and fuselage allow for the attachment of these two components to each other.

4.4 Landing Gear

Due to the previous year's bad experience with failed landing gears this portion of the aircraft was given much attention. It was necessary to optimize the landing gear configuration as well as specific components associated with the landing gear including gear struts, wheels, and gear blocks.

Landing Configuration: At this point, the preliminary calculations and design of a tricycle landing configuration could be completed. To prevent the aircraft from overturning, the main wheels should be laterally separated beyond 25 degrees angle off the center of gravity as measured from the rear in a nose-down attitude. This was evaluated to be a distance of approximately 15 in. from the aircraft centerline at the spar location. The nose landing gear should be placed as close to the front of the aircraft as possible to prevent the propellers from hitting the ground on a hard landing.

Gear Struts: A number of materials were considered for the gear strut, which was designed to interface with the wing spar, including steel, aluminum, and carbon. Carbon and steel would provide the most strength, while aluminum and carbon would be the lightest. Although light, aluminum would not provide the strength required for harsh landings. Due to the previous year's experience with a steel gear strut, the team was confronted with two options; Design a thicker strut from steel, or design a strut from Carbon. It was calculated that steel would be far heavier than a carbon strut with similar strength. The only downside would be the difficulty of manufacturing associated with the composite strut. Despite this, the team decided that composite strut would much greatly improve performance and reliability.

Wheels: Although construction of a lightweight aluminum wheel was a possibility, previous year's experience allowed the team to eliminate this option. It was difficult to attain transverse skid resistance from aluminum wheels, and gluing of rubber o-rings to the aluminum proved a poor tactic as the thin rubber quickly disintegrated upon landing impact. The alternative would be rubber wheels, which are heavier than the custom-made aluminum wheels. To remedy this problem, in-line skating wheels will be purchased and machined to reduce their weight.

4.5 Propulsion and Power System

The propulsion choices that were made in the conceptual phase of the paper were based on general power management, motor life, and RAC concerns. In the PDP, there was a lot more known about the aircraft configuration and more precise propulsion system design could be conducted.

In limiting the motor type and manufacturer, the contest organizers made the determination of motor type a fairly narrow search. Due to recommendations from a number of people with previous experience with such motors, the Astro line of motors was eliminated. We also had our Graupner Ultra 3300 7-wind series motor from the 2001 competition, which was still in great condition and will satisfy this plane's needs and the competition requirements. This motor is designed to run off of 20 volts and turn a direct drive propeller. Assuming a maximum current draw of 35 Amps, safely below the 40 Amp limit, the 7 wind series motor gave an efficiency, output wattage and RPM of 83.5%, 650 Watts, and 8300 RPM, respectively

At this point, using a setup built by the team mentor Steve Neu, a preliminary experiment was performed to qualitatively test how the motor performs under a load. The testing platform was setup precisely for this purpose, as the RPM, Output Voltage, and Input Current were measured. This test also applied a load onto the motor (using hysteresis brakes) and measured the torque applied (using a strain gage) by the motor. From this information, the motor efficiency was calculated. A plot of this experiment can be seen in the following figure. As can be seen from the following figure, the efficiency of the motor is maximized between 12,500 and 13,500 RPM. Because of this and the choice of propeller diameter and pitch, a 2:1 gear ratio will be used to ensure that the motor RPM's are kept at the optimum range.

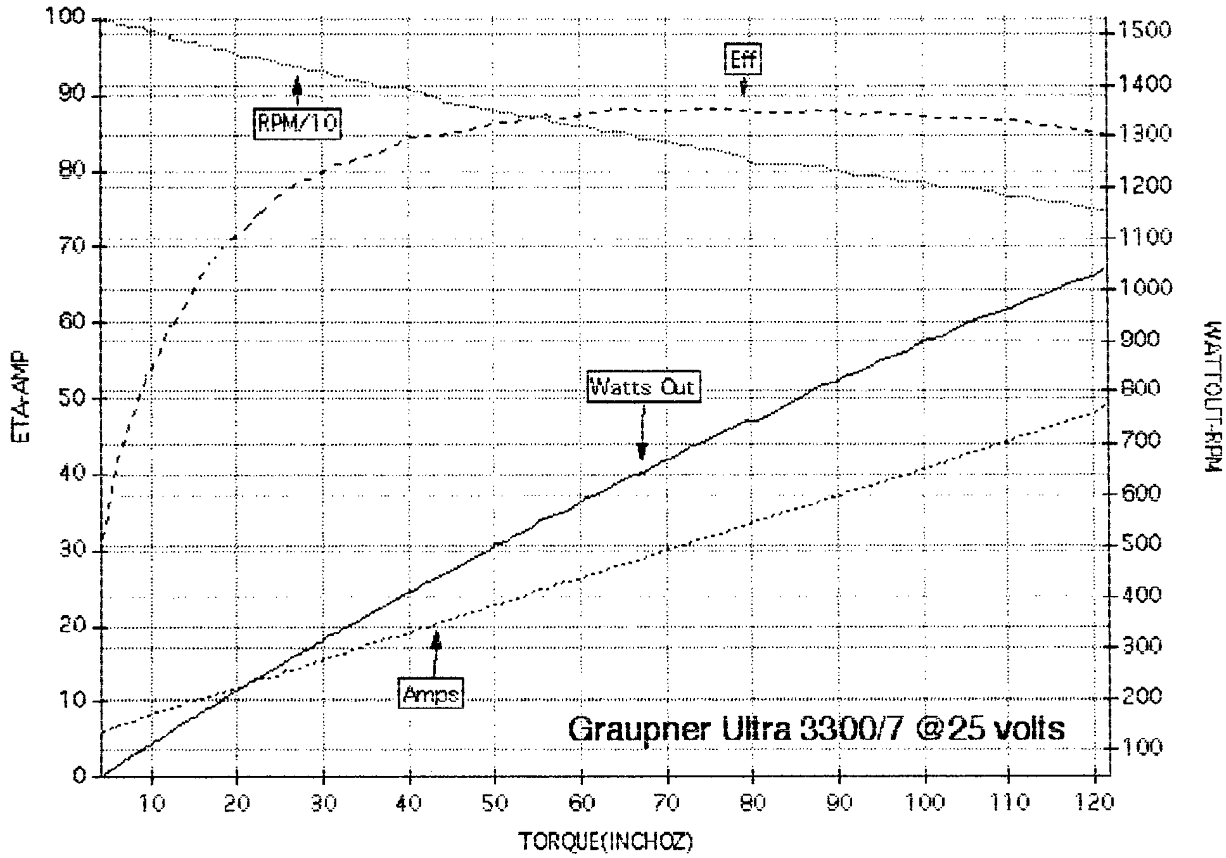


Figure 4.1

Initial testing on the Graupner Ultra 3300/7 motors showing vital statistics.

Courtesy of Steve Neu

The propeller choice was made from a decision for a cruise speed of 65 mph, at a current-draw setting below maximum. The Graupner Ultra 3300/7 motor operates most efficiently at an input voltage of between 23 and 29 volts. At this setting and RPM, a 20" diameter x 20" pitch propeller gives a speed of approximately 65 mph. This factor of safety will be verified during flight testing to determine if an alternate propeller will be needed.

The final step in determining the power system was the choice of batteries. The weight, voltage and current limit was dictated in the rules. The batteries were chosen based on durability, availability and cost. The Sanyo 2400 batteries that were used in last year's competition performed very well under raised current levels and fast discharging, so their reliability had been proven. This fact, combined with their increased availability and reduced cost this year, led to them being selected.

The final configuration for the power plant of TLAR III featured a single Graupner 3300/7 motor, powered by 22 Sanyo 2400 batteries through a 2:1 gear ratio and a commercially purchased propeller of 20" diameter and 20" pitch.

4.6 Structural System

The principal objectives of the PDP for the structural system were to maximize the strength to weight ratio without adversely affecting the payload capacity and access. Focus was placed on structural integrity, to avoid failure under flight and landing loads and component interfacing, as most systems fail at joints and interfaces.

Application of theory from solid mechanics and statics provided the basis upon which necessary design adjustments were made. These adjustments were the variation of the cross-section of the spar structure and the fuselage/wing interface method. In order to determine the best cross-section for the spar structure, stress and static analyses were conducted via the use of Microsoft Excel. Optimization of the fuselage/wing interface involved the determining the maximum stresses, for which it was necessary to determine the maximum load experienced (during banked turns). The following equations were utilized to determine the necessary values of maximum moment and bending loads:

Newton's Law: $\sum F = ma$

Moment Equation: $M = Fd$

Maximum Bending Stress: $\sigma = \frac{Mc}{I}$

Moments of Inertia: $I_x = \int_A y^2 dA$ & $I_y = \int_A x^2 dA$

4.6.1 Spar Structure Optimization:

The wing's thickest point was chosen as the location for the spar structure so as to allow the furthest distance, 'c', from the neutral axis for bending. At the wing root and tip the width of the carbon were 2.25" and 1.5", respectively. At specific points along the span, the values of 'c' and 'M' were fixed thus allowing for variation of the moment of inertia by varying the cross-sectional area. The above equations were used, along with values for a 27 lb. aircraft travelling at 65 mph, to calculate the following load values:

Maximum G Force (with safety factor): 9.7

Maximum Load (per wing panel): 2118.5 oz.

Maximum Moment: 18345.6 in.-oz.

Moments of Inertia: 0.3869 in^4 (box beam)

Maximum Bending Stress: 1,002,000 psi (box beam)

Although the maximum bending stress can be taken by both the box-beam and I-beam sections, the box-beam cross-section was chosen due to the additional strength that would result.

4.7 Preliminary Design Phase Summary

The PDP analyzed the preliminary configuration as developed in the CDP with regard to the determination of exact sizing and component interaction. Design parameters that were related to the purpose of sizing the components were defined locally. FOMs such as component safety, structurally and aerodynamically favorable component interaction, ease of construction, and a low RAC were the

primary factors determining much of the component design. Most general sizing calculations that were done were based upon those used in designing a propeller-driven general aviation and radio controlled aircraft. Precise calculations on flight performance will be completed during the DDP now that the detail configuration of TLAR II has been established.

5 Detail Design

The goal of the DDP is to optimize the sizing and design parameters that were developed in the PDP with regard to flight and takeoff performance. Similar to the PDP, the primary performance analysis techniques used were those applied to propeller driven general aviation aircraft. The calculation of basic characteristics, such as lift and thrust, allowed for the computation of more specific flight characteristics such as takeoff distance, rate of climb and endurance. In each section of the PDP, individual component weights were estimated.

5.1 General Sizing

5.1.1 Weight Estimate

Estimates based on experience gained from last year's design were assembled into a preliminary weight estimate of 20 lbs. Upon completion of the DDP and construction of the aircraft, the final value for aircraft weight was estimated at 24.5 lbs. While heavier than first estimated, it was still well within the competition rules and design limitations of the aircraft.

5.1.2 Wing Performance

The detailed performance calculations of the main wing were based upon Prandtl's classic lifting line theory. Prandtl's theory was implemented numerically in Matlab code and run with the sizing parameters developed in the PDP given inputs for wing dimensions of span, root chord length, aspect ratio, $\alpha_{CL=0}$. These values were reduced from the theoretical infinite wing data by approximately 26%, which was relatively efficient despite the low aspect ratio. However, by altering the values of the taper ratio, it was possible to achieve a 20% reduction. To minimize RAC, we had to use rectangular wing because the advantages of the tapered wing were negated by the cost factor. The final values for $C_{L\alpha=0}$ and C_{Lmax} 0.562 and 1.544 respectively and for $Cd_{\alpha=0}$ and $Cd_{\alpha max}$ .007055 and .023104 respectively, were realized with a root chord of 13.5 in. This gave the final wing an aspect ratio of 6.22, taper ratio of 1.00, platform area of 1134 in^2. An important quantity resulting from these refined sizing and performance characteristics was a wing loading, W/S, of 2.74lbf/ft^2.

5.1.3 Fuselage Layout

The final configuration of 4x6 softballs in the fuselage brought a peculiar problem of increased drag over the very wide (16.25 in) fuselage. To reduce the drag a more streamlined fuselage was suggested and implemented. NACA 0013 airfoil shape was used for this purpose. Additionally, there will be a benefit of the lift produced at angle of attack, which will decrease the take-off distance and increase the rate of climb. Consideration of the lift generated by the fuselage shall be limited to replacing the wing area displaced by the fuselage. Since a hatch will be cut into the fuselage skin for loading the payload, torsional strength of fuselage will be lost. Torsional stiffness comes from the torque box made up of the fuse sides, bulkheads, ball floor, wing and bottom skin.

5.1.4 Motor Thrust Available

While the static thrust of an engine was a necessary calculation related to the determination of performance characteristics such as takeoff distance and rate of climb. Analysis of power supplied being reduced by the inefficiency of the motor and propeller was used to indicate the amount of power available to actually propel the aircraft. Having determined the number of cells to power each motor in the preliminary phase, the total power available to the motor was calculated using the equation $P = VI$, giving an available power of 1017 Watts. With a combined efficiency of .78, the thrust developed per engine was 6.18 lbs. This gave a final thrust-to-weight ratio of 0.286.

5.2 Flight Performance Calculations

In optimizing sortie performance, it was necessary to evaluate takeoff, rate of climb and range/endurance characteristics as well as static stability. While dynamic stability was also a concern, for a fairly conventional design as TLAR III, dynamic instability has proven to be highly transient in modes that are easily controlled by a skilled pilot.

5.2.1 Takeoff Performance

The takeoff profile for TLAR III was essential, as it was a competition constraint. It was determined for takeoff that the rotate/stall speed needed was 42.1 ft/s, while the actual takeoff speed was 46.3 ft/s.

For the Reynolds's number (~200,000 to 300,000 based on M.A.C.) at which the takeoff was performed, the induced drag of the aircraft was small enough to be approximated to zero. This enables the approximation of integrating Newton's second law over the ground run of the aircraft. This integration of the acceleration can be reduced to the following equations from reference 1:

$$\left(\frac{1}{2gK_A}\right)\ln\left(\frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}\right) \quad \text{with} \quad K_T = \left(\frac{T}{W}\right) - \mu\,, \quad \text{and} \quad \frac{\rho}{2(W/S)}\left(\mu\, C_L - C_{D_o} - KC_L^2\right)$$

A ground roll of 166 ft was calculated for a rolling resistance factor (μ) of 0.05 and a total takeoff distance of 183.5 ft. While this seems close to the distance limit, the likelihood of winds being present during the competition will shorten the actual takeoff distance. If this is not the case, and taking off within the limit proves difficult, reducing the payload may be considered.

5.2.2 Rate of Climb

The rate of climb (R/C) is the vertical component of the aircraft velocity. It can be expressed in terms of weight (W), perpendicular component of drag (W_D) and velocity (V). W_D is a function of thrust (T) and drag (D), therefore yielding:

$$\frac{R}{C} = \frac{(T - D) \times V}{W}$$

The best climb rate will occur at the velocity for maximum lift to drag ratio, L/D_{max}. Although the thrust produced by the propeller and corresponding power available changes with the airspeed, an

approximation was made to show that the maximum lift to drag depends only on the values for the parasite drag coefficient, C_D and the wing's aspect ratio, AR.

$$\frac{L}{D_{\max}} = 0.886\sqrt{\frac{AR}{C_D}}$$

Using these formulas, the rate of climb for the aircraft was calculated to be 597 ft/min.

5.2.3 Turning Radius

The turning radius gives an indication of how much distance, time and power are needed to perform a turn and is necessary because it affects sortie flight time and power management.

The turn radius calculations began with the assumption that the turn will be made at the cruise speed of 65-MPH. The equation

$$n = \frac{\left(V_{cruise}\right)^2\left(C_{L_{\max}}\right)}{2\left(W_{aircraft}\right)}\left(Area_{wing}\right)$$

gave a load factor of 4.87 for the turn. From this, and geometry, a bank angle of 78.1 degrees and turn radius value of 59.3 ft was calculated.

5.2.4 Range and Endurance

Range and endurance characteristics are essential performance indicators for an aircraft designed to participate in timed, limited fuel flight profiles. The range figure of TLAR III was based on the power available and the power that the missions will require, so that the time can be estimated.

Initially, this was accomplished by examining TLAR III in steady flight. The power available is the number of cells multiplied by the power rating for each cell. This equated to 1 battery pack multiplied by 2400 milliAmp-hours for at total of 2.4 Amp-hours. This allows 6.3 minutes of flight time at the current setting of 22.8 Amps, which subsequently gave a cruise range of 6.8 miles and 3.6 minutes and maximum thrust.

While this flight time was less than the maximum flight period, an Excel spreadsheet was used to determine a flight plan. It calculated using average speed and current setting breakdown for the entire mission profiles, with gliding and powered back cruises; the flight time was under five minutes which is far less then the maximum time given.

5.2.5 Static Stability

The ability of the aircraft to return to trimmed flight, if it was somehow deflected, was an important consideration. Due to the symmetry of the aircraft about the centerline of the fuselage, it was possible to achieve some roll stability of the design, with the selection of 3° dihedral in the preliminary phase. The focus was placed on the pitch stability (the directional stability was already addressed in the sizing of the vertical tail stabilizer). A well-designed aircraft that has good static stability, historically will also behave well dynamically.

The pitch stability is related to how the main-wing and horizontal stabilizer work together to correct pitch deflection. For TLAR III, this was especially critical due to the choice of a symmetrical airfoil for the tail. To prevent excessive corrective elevator deflection during steady flight, the pitching moment

of the airfoil needs to be balanced by the placement of the center of gravity relative to the center of pressure of the wing. Assuming that any pitch deflections would be small, the pitch static stability is directly dependent on the horizontal stabilizer. The calculations focused on determining the exact placement of aircraft CG for a desired Stability Margin of 0.3. This margin is the geometrical distance between the location of the CG and the location of the neutral point. The neutral point was determined with the following equation along with design parameters defined earlier in the paper.

$$x_{np} = x_{AerodynamicCenter} + V_t\left[1-\left(\frac{4}{AR_{MainWing}}\right)\right]$$

This allowed the placement of the wing at the optimum position for stability. The main wing leading edge was located at 11.375 in from the nose of the aircraft, which places the CG almost directly on the spar location. Additional pitch stability was achieved through decalage (angle between the horizontal stabilizer and the main wing), which was initially set at 1° and will be finalized during flight-testing.

5.2.6 Systems Architecture

The Systems Architecture describes the electrical components used to fly the aircraft including the batteries, servos, receivers, and transmitter. The number, location, and type of motors and batteries were determined in the propulsion section of the PDP. Determining the types of servos remained.

Given the values obtained for the torque required by each servo, two different servos (HS-225MG and HS-545BB) manufactured by Hitec RCD Inc. were chosen for the rudder, elevator, and aileron controls. The HS-225MG, capable of 55-67 $^{oz}/_{in}$ of torque was used to control the elevator. The HS-225MG was also used for the rudder, despite the rudder having less control surface area than the elevator, because the same servo controls both the rudder and the nose wheel. The spoilerons however require more torque to control due to higher surface area, therefore a more powerful servo, HS-545BB, capable of 62-73 $^{oz}/_{in}$ of torque was used.

(Tables on Next Page)

5.2.7 Detail Design Summary

Table 5-1 General Sizing

Design Parameter	Size
Aircraft Dry Weight	12 lbs.
Manufacturers Empty Weight	14.9 lbs.
Full Capacity Weight	24.5lbs. (with 24 Softballs 9.5 lbs.)
Battery Pack Weight	2.9 lbs. (22-2.4 Amp Cells)
Main Wing Configuration	Low Mono-Wing
Main Wing Airfoil	E212
Main Wing Area	1134 in^2
Taper Ratio	1
Aspect Ratio	6.22
Number of Motors	1
Propeller Size	20" x 20"
Total Thrust	5.31 lbs.
Horizontal Stabilizer Area	226 in^2
Vertical Stabilizer Area	125 in^2
Tail Airfoil	NACA 0009
Fuselage Height	5.25 in
Fuselage Length	39.5 in
Fuselage Width	16.25 in
Fuselage Foil	Modified NACA 0013

Table 5-2 Systems Architecture

Component	Description (Amount)
Motors	Graupner Ultra 3300/7
Servos	HS-225MG (2), HS-545BB (2)
Batteries	Sanyo 2400 (22 cells)
Receiver	HPD-07RB (PCM)
Handset (Radio Transmitter)	Prism 7X (PCM)

Table 5-3 Flight Performance Data

Design Parameter	Performance
Cruise Speed	65 mph (95.3 $^{ft}/_{s}$)
Takeoff Distance	183.5 ft
Climb Rate	597 ft/min
Turning Radius	59.3 ft
Payload/Gross Weight Ratio	0.44
Cruise Endurance	6.3 minutes
Mission Completion Goal	All 3 missions in 5 minutes
Rated Aircraft Cost	9.29
Final Flight Score	6.2

Rated Aircraft Cost (RAC) = (A * MEW + B * REP + C * MFHR) / 1000 = 9.28

Manufacturers Empty Weight Multiplier (MEW) A= $ 100

Total Weight w/o payload = 14.9 lbs

MEW = 14.9

Rated Engine Power (REP) B= $ 1500

# of engines = 1
Battery Weight = 2.9 lbs

REP = 2.9

Manufacturing Man Hours (MFHR = SUM(WBS)) C= $ 20 /hour

WBS 1.0 Wings					
Wing Span =	7	ft *	8 hr/ft		
Max Chord =	1.13	ft *	8 hr/ft		
# of Control Surfaces =	4	c.s. *	3 hr/c.s.		
				WBS 1.0 =	77 hr
WBS 2.0 Fuselage					
Fuselage Length =	4.5	ft *	10 hr/ft		
				WBS 2.0 =	45 hr
WBS 3.0 Empenage					
# of Vertical Surfaces =	0	v.s. *	5 hr/v.s.		
# of Vertical Surfaces w/ =	1	v.s. *	10 hr/v.s.		
# of Horizon Surfaces w/ =	1	h.s. *	10 hr/h.s.		
				WBS 3.0 =	20 hr
WBS 4.0 Flight System					
# Servo or Motor Contro. =	4	ser *	5 hr/ser		
				WBS 4.0 =	20 hr
WBS 5.0 Propulsion Systems					
# of engines =	1	eng *	5 hr/eng		
# of props =	1	prop *	5 hr/prop		
				WBS 5.0 =	10 hr
				MFHR =	172
				RAC =	**9.28**

5.2.8 Weights and Balances Worksheet

Total Length (in)	54		
Length of the shell	36	11.375	Distance from the nose cone to the leading edge at the root
Boom Length (in)	18		
Item Name	Item Weight (oz.)	Arm (in)	Moment (in-oz.)
Main Wing	40	15.875	635
Rear Stabilizer	6	49.75	298.5
Tail	3.5	47.75	167.125
Boom	5.4	45	243
Fuselage Shell	29.592	18	532.656
Fuselage Spar Support	50	15.251	762.54925
Main Landing Gear (x2)	10	15.875	158.75
Nose Gear Structure	5	2	10
Engine (x1)	21.12	4	84.48
Battery Packs (x4)	48	10	480
Propellor (x1)	6	3	18
Wiring Harness	2	30	60
Speed Controller	6	4	24
Receiver	1.65	25	41.25
Main Wing Servos (x2)	2	21.5	43
Elevator Servo	1	49.75	49.75
Rudder Servo	1	47.75	47.75
S-Ball Payload	154.2857143	16.00	2468.571429
Heavy (S-Ball) Payload	392.5477143	24.5342	6124.981679
No Payload	238.262	14.8914	3655.81025

Number of Soft Balls: 24

Total Soft Balls: 24

w/ Payload		No Payload	
MEW	11.9 lb	MEW	11.9 lb
Gross T/O Weight	24.5 lb	Gross T/O Weight	14.9 lb
CG (F.S. in)	15.60 in	CG (F.S. in)	15.34 in
Dist. To Leading edge at root	11.38 in	Dist. To Cent. Press. (1/2 c) at MAC	11.38 in
CG (%MAC)	31.3 %	CG (%MAC)	29.4 %
Distance In Front of the spar	0.35 in	Distance in front of Cp	0.09 in

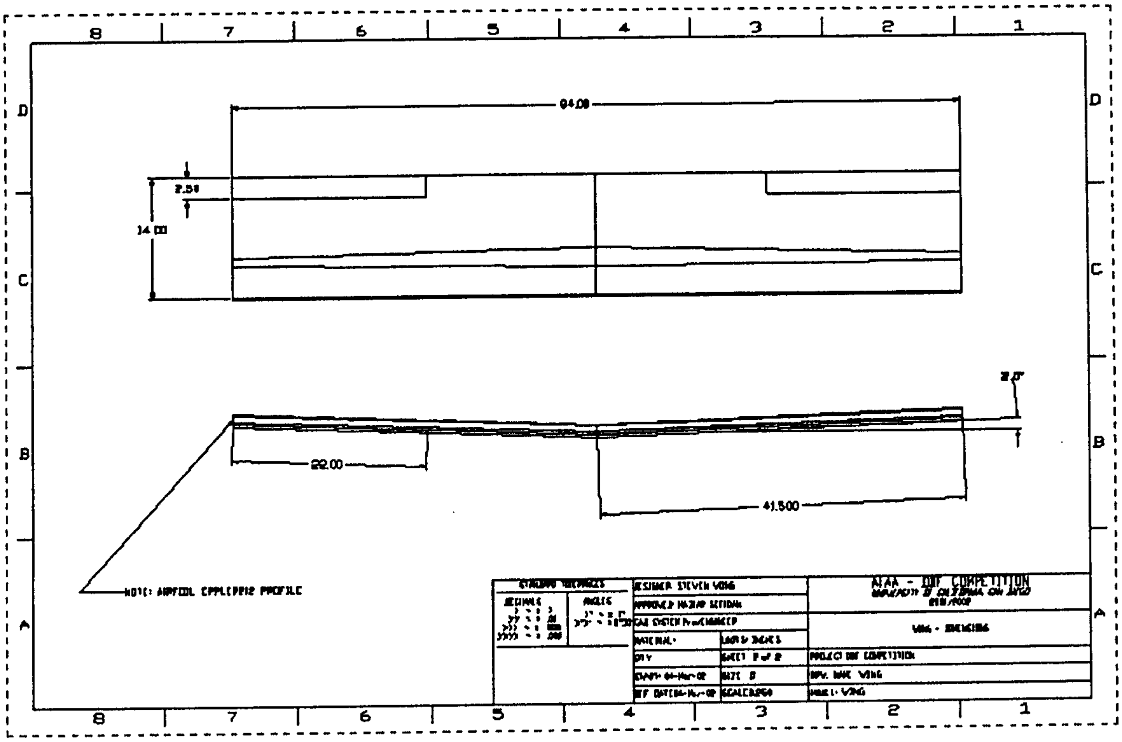
Notes:

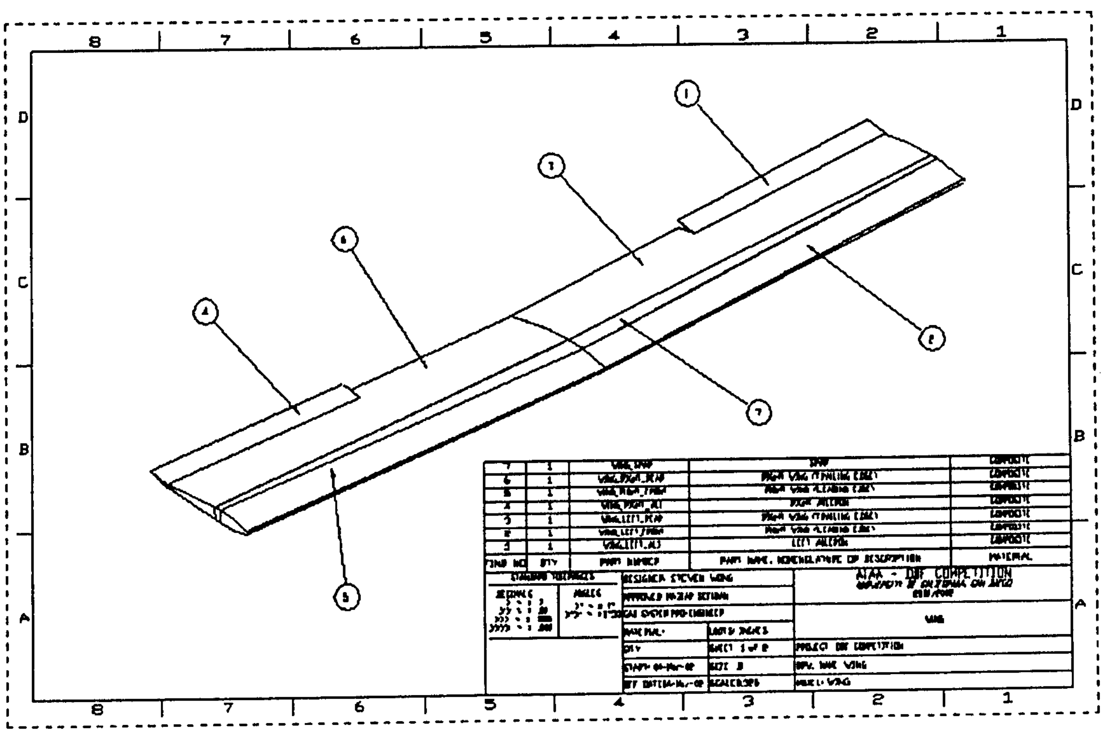
1) All distances are from the CG of the individual item to the nose of the A/C.

2) All weights include attachment hardware.

5.3.2 Flight Component Dimensions

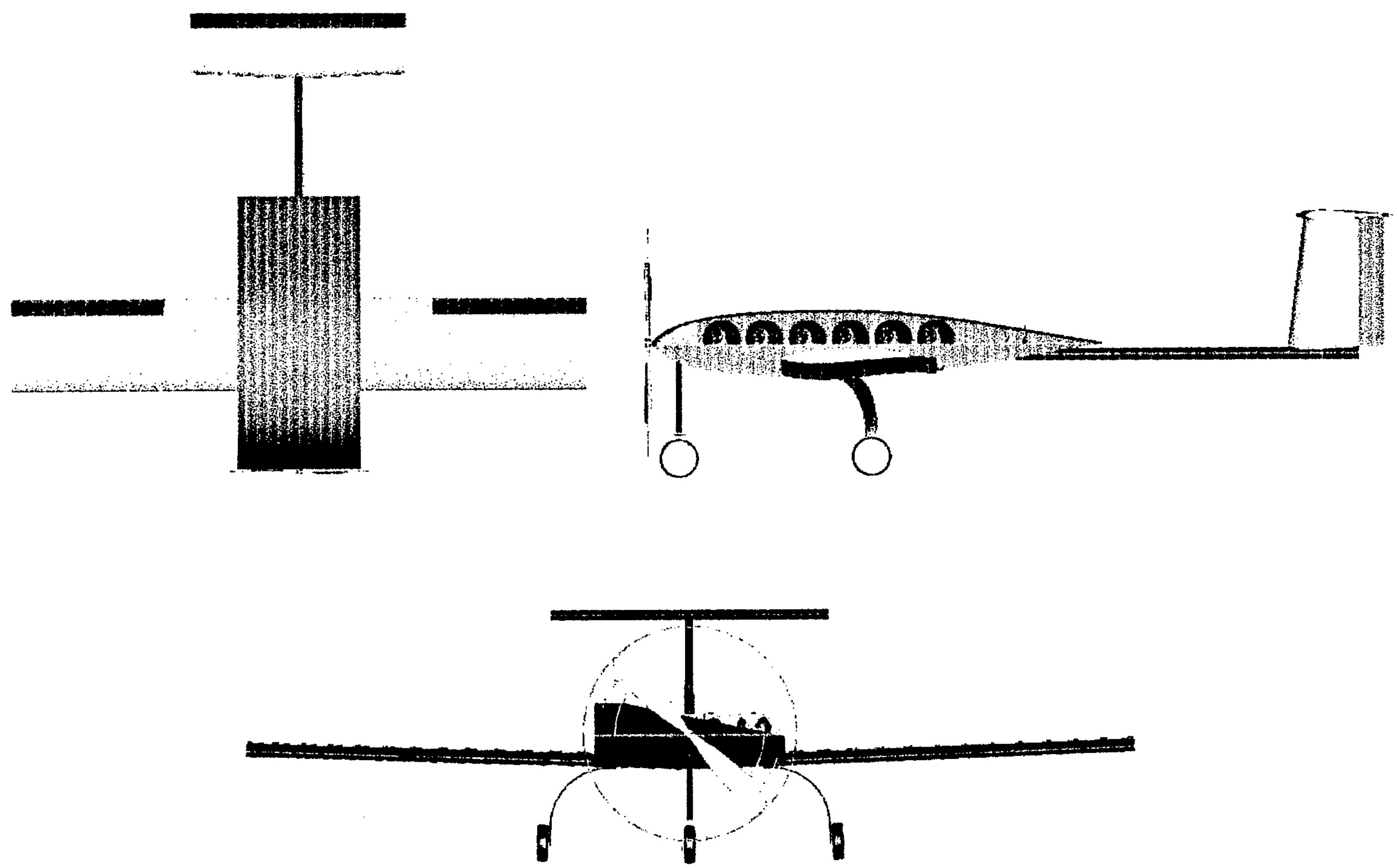
5.3.2.1 WING:





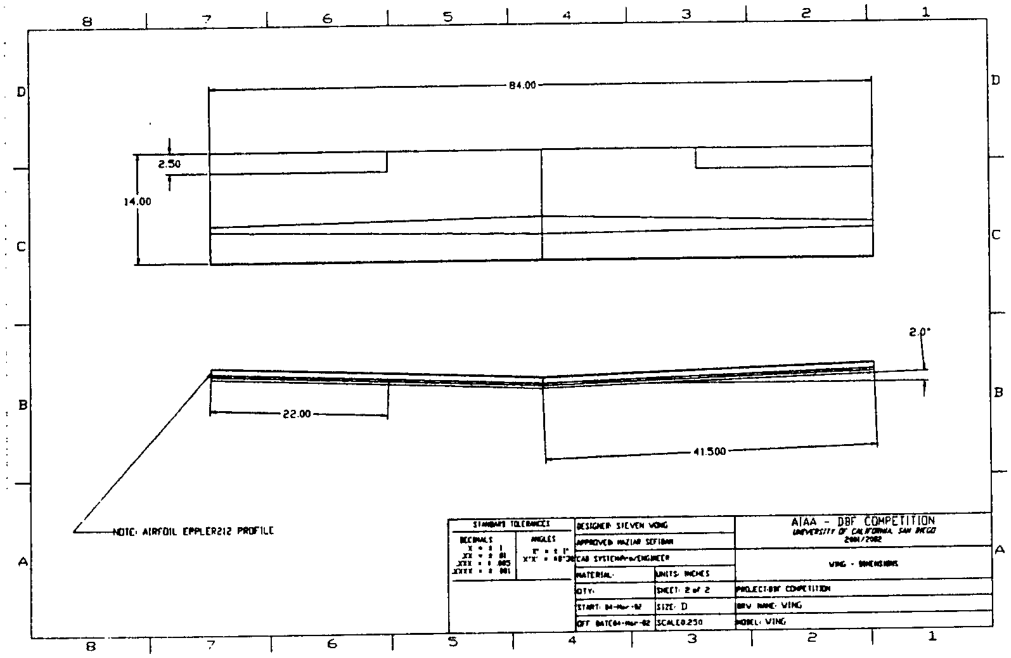
5.3 Drawing Package

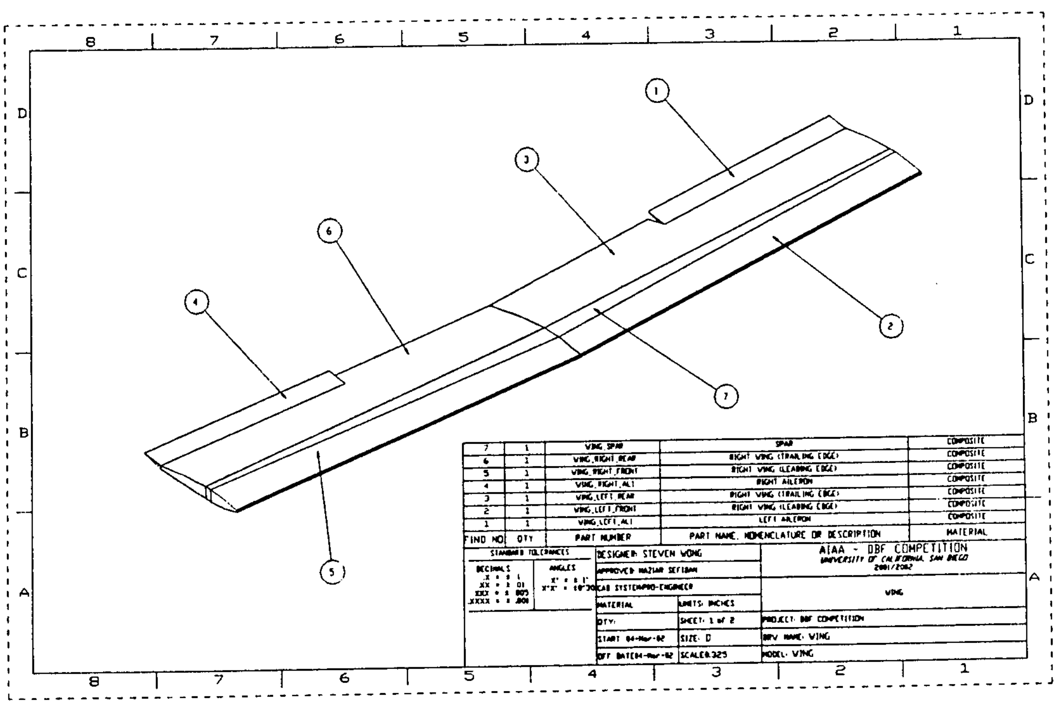
5.3.1 Three View Drawing



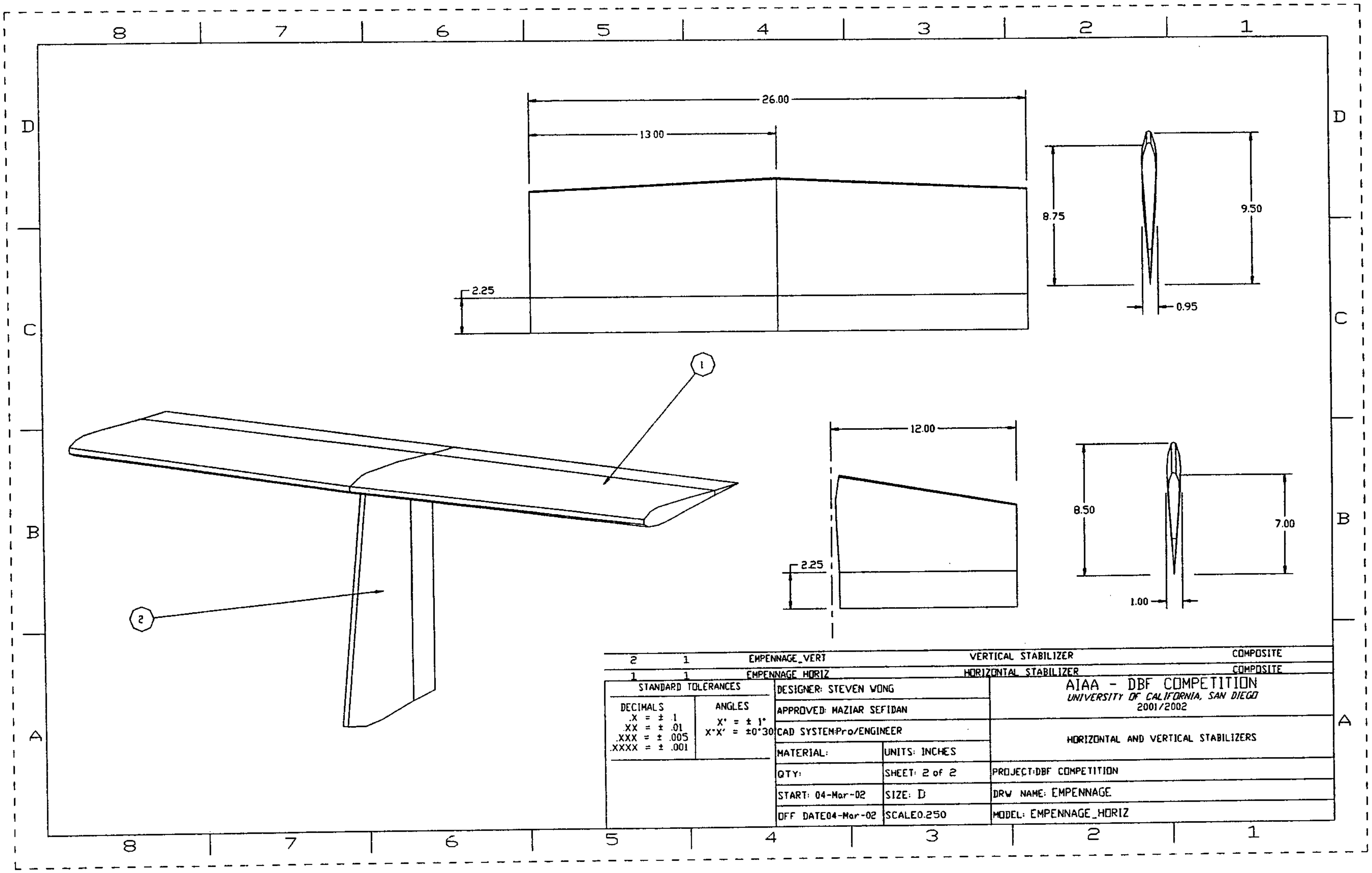
5.3.2 Flight Component Dimensions

5.3.2.1 WING:

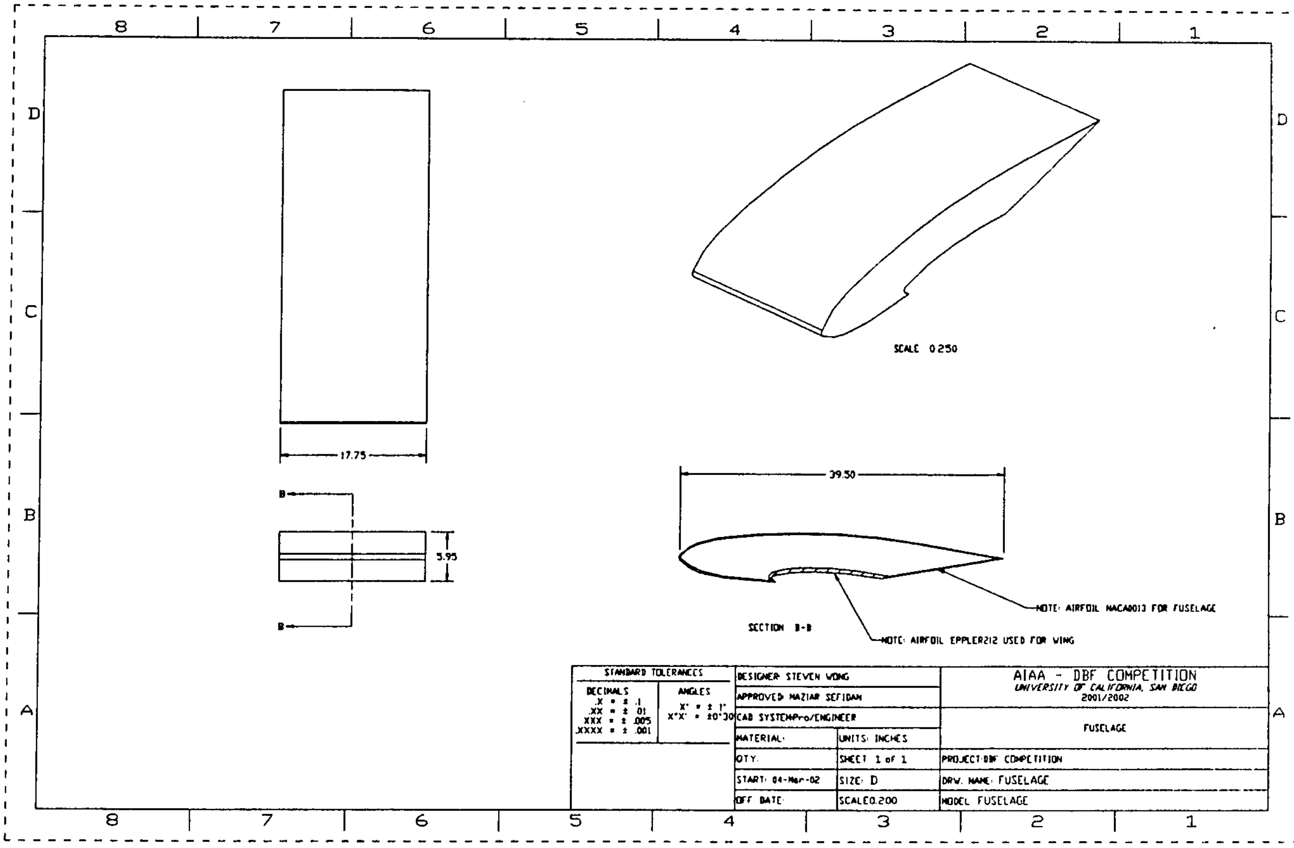


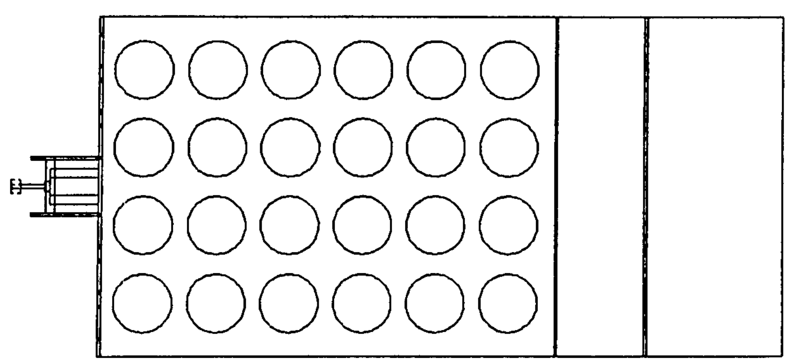


5.3.2.2 HORIZONTAL AND VERTICAL STABILIZER:



5.3.2.3 FUSELAGE:





6 Manufacturing Plan

Logistics and required skill level to manufacturing component design ideas imposed constraints on design options. The manufacturing constraints were continually considered in order to avoid conflicts between a chosen design idea and the ability to construct it. The manufacturing FOMs were used to screen construction techniques and materials based on skill level, physical properties, component and interfacing strength, and logistics. The material FOMs were sometimes used to eliminate construction methods, while construction FOMs were sometimes used to eliminate materials. The order in which the various components and assemblies were completed are shown in the manufacturing timeline presented in section 6.4.

6.1 FOM Reasoning/Discussion

Materials: Materials were screened due to their strength-to-weight ratios, availability, cost and required skill level.

Construction Techniques: "Home-built" construction with limited techniques were the only methods available to the team due to limited experience with an autoclave, time restrictions, and the unavailability of a large CNC machine. Construction FOMs are composed of required skill level, machinery, tools, tooling time, robustness and ease of repair.

6.2 Component Manufacturing Description

6.2.1 Spar Structure

The spar structure assembly consists of the shear web and spar cap. Although the wing itself is constructed of low density white foam, high density blue foam is used in the section where the spar would be placed.

Spar Cap Materials: The spar cap absorbs the majority of the bending stress and is of variable thickness (it is reinforced at the center of the spar). A carbon material was best suited for this component because of its high axial load capacity.

Spar Cap Construction Techniques: Two layers of unidirectional carbon were wetted out with epoxy The strands of 12 K carbon tow were wetted-out with epoxy and laid on top of the spar foam. This was then placed inside a polyurethane bag and vacuum-sealed to roughly 12 PSI.

Shear Web Materials: The shear web is only required to give spacing between the spar caps and to absorb a nominal amount of shear stress. Material FOMs were applied to various lightweight materials in order extract the best material. The materials investigated for the shear web were carbon plate, fiberglass, balsa, and Aircraft plywood. Aircraft plywood was eliminated due to its high relative weight and balsa was eliminated due to its low strength. Fiberglass was eventually chosen over carbon due to its weight and ease of construction.

Shear Web Construction Techniques: The fiberglass was cut out at 45° bias, as this angle provides maximum shear strength. Two layers of this cut fiberglass were then wetted out with epoxy and placed on high-density spar foam between the two spar caps. This technique was applied to both sides of the foam, after which the entire spar structure was placed inside a polyurethane bag and vacuum-sealed to roughly 12 PSI.

6.2.2 Wing Airfoil

Airfoil Templates: The first step in the process of fabricating the wings was to make Eppler 212 airfoil templates.

Materials: The materials considered for the airfoil templates were 6061 aluminum, aircraft plywood and Formica. Since the hot wire method was the desired cutting process for the wing cores, it was necessary for the chosen material to be smooth for the wire path, and not melt under hot wire temperatures. Formica was the clear choice due to smooth surface finish and low thermal conductivity.

Construction Techniques: The two techniques for construction of airfoil templates considered were the "cut-and-paste" and a CNC machine cutting method. The "cut-and-paste" process was used because it was easier. Using data points obtained from the UIUC Database, printouts of the airfoil for chord length of 13.5 inches was created. Since the wing would be rectangular, the root, midpoint, and tip of the wing would have the same chord length, making construction that much less complicated. Printouts were bonded to the Formica, which were then cut into the shape of the airfoil and pegged into both edges of a rectangular piece of low-density EPS foam.

6.2.3 Wing Cores

The purposes of the wing core are to maintain the airfoil cross-sectional shape and to contribute to the composite strength.

Materials: The materials considered were aircraft ply, and blue and EPS foam. Using aircraft ply, would result in a design using a rib structure overlaid with MonoKote, while the foam design could form a solid core for composite lay up. The foam was chosen because a solid core was more structurally sound and required less construction. The stiffness and compressive strength of the blue foam and low weight of the EPS foam made both of them viable choices for the core. A combination of blue and white EPS foam was used. The dense blue foam provides stiffness so it was used in the spar structure while the less dense white EPS foam conserves weight and it used for the rest of the wing core.

Construction Techniques: The two construction techniques investigated for cutting the foam core were utilization of a large CNC machine and a hot-wire cutting process. The CNC machine was immediately eliminated due to the unavailability. The hot-wire technique satisfied the logistical issues and the skill level requirements.

The two wings were 42" in length, which was slightly shorter than the length of the hot-wire cutter. This made construction only a one-cut process. A 45"x15"x3" (slightly larger dimensions than the wing) white EPS foam rectangular block was used to cut the airfoil profile of the wing. The Formica templates

were pegged onto the edges of the foam using reference pins. After the reference pins were inserted into the foam blocks, the hot wire was drawn through the foam, following the template, creating the desired E212 airfoil cross-section. The spar area was then hot-wire cut from the wing core. After the instersion of the spar, the two wings are then epoxied together.

6.2.4 Spar Assembly

Even thought the two wings are made separately, the spar is made as one piece. This is essential for load transfers involved with the spar. The strength of the spar also makes it the idea candidate for the insertion of landing gear hard-points. These hard-points are made of airplane grade plywood and are placed on top of the blue foam beneath the spar caps. The spar assembly is then epoxied to the rest of the wing.

6.2.5 Wing Skins

The functional requirement of the wing skin is to provide a smooth surface for airflow over the wings, which reduces parasitic drag. The skin is also required to provide torsional strength to dissipate stresses generated during flight.

Materials: Carbon and fiberglass cloths were the materials considered for the wing skins. Carbon cloth was eliminated due to high cost and difficulty of working with the dimensions that were required. Fiberglass cloth was chosen because it was cheap and easy to work with.

Construction Techniques: The use of foam as the core excluded the possibility of using an autoclave, therefore a wet lay up process was needed. Fiberglass was laid at a 45° bias over Mylar. Each wing core was then sandwiched between the sheets of fiberglass/Mylar. The whole assembly was then placed in a vacuum bag to compress the fiberglass/Mylar. After the epoxy had finished curing, the wings were removed from the vacuum bag, the Mylar was peeled off and the excess fiberglass was trimmed or sanded off.

6.2.6 Final Wing Assembly

The fiberglass covering the hard points was trimmed off, allowing access to the plywood below. The spoilerons were cut out and trimmed to allow enough clearance for full range of motion. The recesses to hold the servos were cut into the wings and the wing tips were cut at a 45-degree angle. Fiberglass was then adhered to the exposed core surfaces to provide strength.

6.2.7 Stabilizers

The airfoil decided on for the horizontal and vertical stabilizers was the NACA 0009. The same materials and construction techniques were chosen for the two stabilizers.

Materials: The horizontal and vertical stabilizers were made from foam and fiberglass for the cores and skins, respectively. The only difference from the wings was that only blue foam was used. To provide stiffness and strength unidirectional carbon cloth was laid along the span at half the chord length. The final material used in the tail was aircraft ply that acted as hard points at the interface with the boom.

Construction Techniques: The final lay-up of the stabilizers was exactly the same as the wings.

6.2.8 Main Landing Gear Strut

The gear struts need to withstand fatigue and to absorb the majority of the loads applied at landing.

Materials: The materials considered for the main landing gear struts were carbon, 6061 aluminum and 1095 cold rolled steel piano wire. Aluminum was eliminated due to its lack of strength, and the steel eliminated due to its weight. This left carbon as the only option. The best type of carbon for this job would be multiple layers of pre-impregnated carbon which would have to baked within an autoclave.

Construction Techniques: Carbon landing gears are not easy to manufacture. First a mold was made from a sheet of aluminum by bending it to semi-circle shape. Next, pre-impregnated carbon was cut at 60°, -60°, 90°, and 0° biases and 22 layers were placed on top of the mold. This created a layout that has high torsional and tensile strength. This whole assembly was then vacuum bagged and placed inside the autoclave. Once the epoxy cured under the intense heat and pressure, it was cut into the desired shape using the band saw.

6.2.9 Gear Blocks

Materials: The two possible materials investigated were aluminum and aircraft ply. Tooling time required caused the aluminum to be eliminated. Aircraft ply was chosen since tooling time, cost and skill level were nominal.

Construction Techniques: Prior experience with the materials and component design eliminated the need to investigate construction techniques. The aircraft ply was cut into the required dimensions and then stacked together and bonded with epoxy. A second layer of plywood consisted of two pieces, thinner than the first layer, for the landing gear strut. Epoxy was applied to T-nuts that were pressed into the wood to hold the bolts that would keep the piano wire in place.

6.2.10 Wheels

Materials: Two possible materials for the wheels were aluminum and high-density rubber. Although the aluminum wheels would be lighter, their manufacturing would be difficult and their lack of traction makes them a poor choice. Because of this, 3" diameter commercially available rubber wheels were used for the aircraft.

6.2.11 Nose Landing Gear

Materials: Due to ease of manufacturing, a 3/16" strut commercially available landing gear was used for the construction of this aircraft.

6.2.12 Fuselage

Materials: The three materials investigated for the construction of the skeleton of the fuselage was bi-directional carbon, spider-foam, and balsa wood. Since the skin would be also constructed from carbon, it was crucial that a light variety of carbon cloth be chosen. After some investigation, a 4.8 oz/yd^2 bi-directional carbon fiber cloth was chosen for this purpose. Although lighter cloth could have been purchased, the cost of this very specialized cloth would have been unaffordable. The spider-foam and balsa wood were considered for the inner layer of the sandwich material. They are both light, yet strong.

Construction Techniques: The construction of the fuselage was a very time consuming and laborious task. First, the skeleton of the fuselage was constructed from sandwich material as described in the PDP section. The spider foam was cut into a 1/8" thick piece and into an airfoil shape using a razor blade. Similarly, the bottom piece spiderfoam, was cut into a 1/8" thick piece and circular holes where softballs sit were cut out of the foam. Lastly, the same technique and material was used to make bulkheads that would fit between the two airfoil-shaped pieces. This foam was then sandwiched by two layers of bi-directional carbon essentially creating an I-Beam structure. Each piece was then vacuum-sealed overnight and trimmed to the exact necessary shape. These pieces were then bonded together using an epoxy/cavasil mixture. The fuselage skin was constructed of bi-directional carbon that was shaped using a male mold of the fuselage. The

6.2.13 Boom

The functional requirements for the boom were stiffness and bending strength to support loads from the tail and landing gear.

Materials: Two materials that were considered for the boom were carbon and aluminum. The carbon was chosen over the aluminum due to issues with weight and stiffness.

Construction Techniques: Due to prior experience with carbon tubes, no addition construction techniques were investigated. A 60" long (1" ID x .032" WT) carbon tube was cut to the necessary 22" length. Aircraft ply was bonded to the inner diameter to provide strength for the connection to the vertical stabilizer and tail gear.

6.3 Manufacturing Timeline

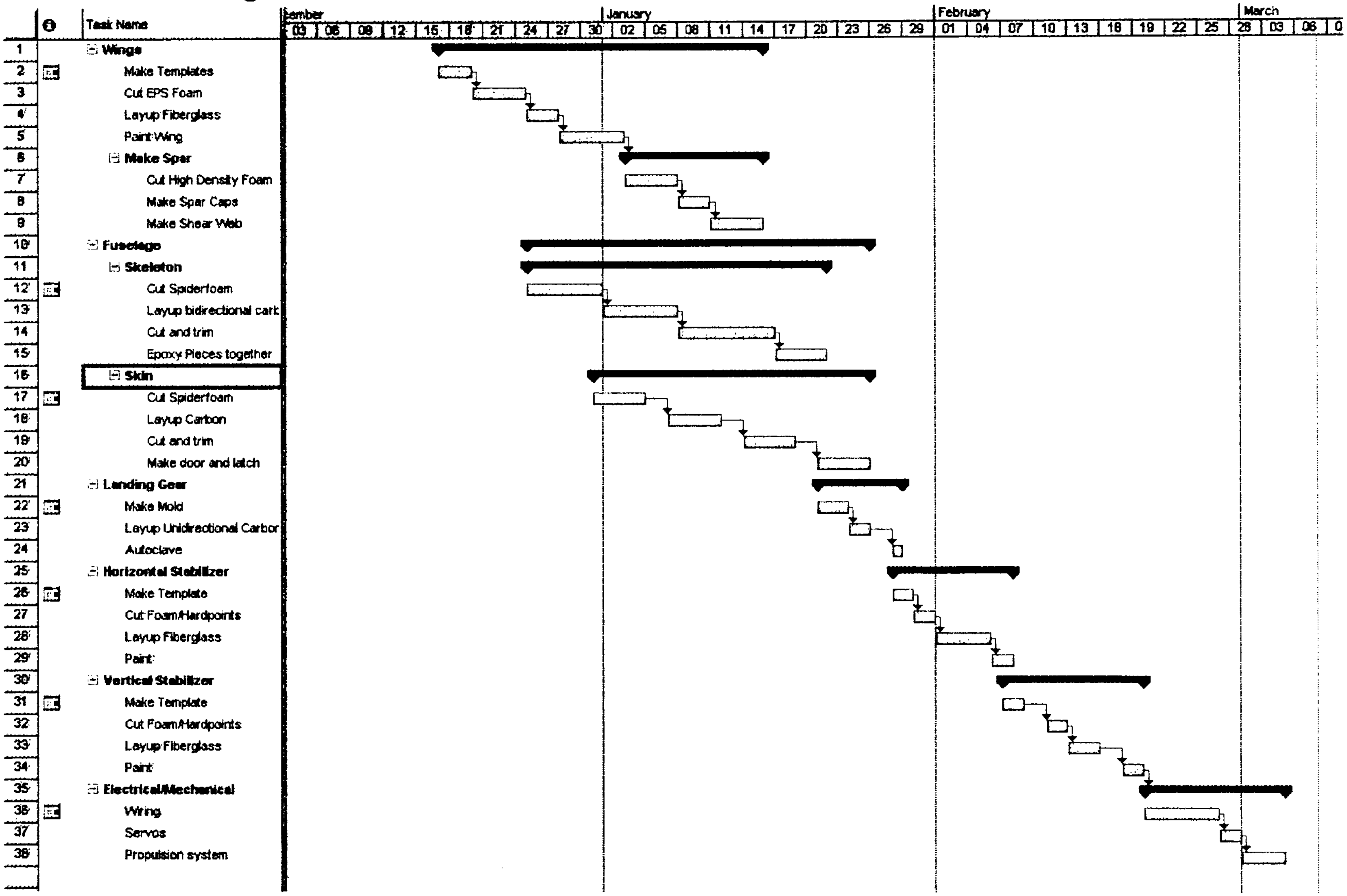


Figure 6-1 Manufacturing Timeline

6.3.1 Cost Matrix

Description	Cost/Unit	Unit	Unit Description	Total Cost
Bi-Directional Carbon Cloth (4.8oz/yard^2)	$ 15.00	5	Linear Yards	$ 75.00
Uni-Directional Carbon Strands	$ 10.00	2	Square Yards	$ 20.00
Bi-Directional Fiberglass Cloth (1.0 oz/yard^2)	$ 10.00	7	Linear Yards	$ 70.00
Epoxy Resin and Hardener	$ 30.00	1	Quart	$ 30.00
EPS Foam (1 lb/ft^3)	$ 25.00	2	3"x36"x60"	$ 50.00
High Density Blue Foam (2 lb/ft^3)	$ 50.00	1	3"x36"x60"	$ 50.00
Spider Foam (4 lb/ft^3)	$ 20.00	1	2"x12"x24"	$ 20.00
Tape/Paint/Glue/Other consumables	$ 100.00	1		$ 100.00
Graupner 3300/7 Electric Motors	$ 300.00	1	Motor	$ 300.00
HS-225MG	$ 25.00	1	Servos	$ 25.00
HS-545BB	$ 35.00	3	Servos	$ 105.00
Recievers	$ 75.00	1	Reciever	$ 75.00
Transimitter	$ 350.00	1	Transmitter	$ 350.00
Speed Controllers	$ 50.00	1	Controller	$ 50.00
Propellors (20" Diameter 20" Pitch)	$ 80.00	1	Prop	$ 80.00
Graupner Spinner	$ 32.00	1	Spinner	$ 32.00
Sanyo 2400 Batteries	$ 5.00	44	Battery	$ 220.00
Hysteris Brakes	$ 25.00	2	Brake	$ 50.00
2:1 ratio Gearbox	$ 150.00	1	Gearbox	$ 150.00
Axel/Wheels/Bearings/Nose Strut	$ 45.00	1		$ 45.00
Expendable Tools	$ 125.00	1		$ 125.00
Softballs	$ 3.00	24	ball	$ 72.00
Total				**$ 2,094.00**

7 Proposal Phase References

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2. Simons, M., Model Aircraft Aerodynamics, Argus Books, 3rd edition, 1994.

3. Lennon, A., The Basics of R/C Model Aircraft Design, Air Age Inc., 1996.

4. McCormick, B.W., Aerodynamics Aeronautics and Flight Mechanics, John Wiley and Sons, 2nd edition, 1995.

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6. Stinton, D., The Design of the Airplane, Van Nostrand Reinhold, 1983.

8 Lessons Learned

8.1 Introduction

Throughout the design process, many important lessons were learned and valuable experience was gained. The success of this project relied on the diverse backgrounds and skills contributed by each team member. This included experience with composite materials, structural and aerodynamic analysis, machining techniques, electronics, power systems, computer programs, team/time management, and fundraising.

8.2 Time Management

Establishing a management structure and starting the design process as early as possible allowed room to deal with unforeseen obstacles. An early start enabled the conceptual phase of the design to be completed so that the basic aircraft layout was defined. With this layout, estimations for manufacturing timelines and expenses set a baseline for planning the rest of the project. As the manufacturing process continued, preliminary and detail design could be completed with more information of the actual aircraft characteristics, resulting in the optimum design.

It was important to set a timeline with realistic goals at the start of the project and then adhere to it throughout the design process. One example was planning the Proposal Phase paper well in advance. An outline was created and sections were divided among team members to distribute the workload.

8.3 Teamwork

Creating an organized team was essential in order to allocate responsibilities among the team members. One of the major lessons learned working as a team, was that productivity can decrease with too many people assigned to one element of the project. Delegating separate individual responsibilities allowed specialization. Incorporating each team member's concepts and ideas about their specialized areas into the final design was a process that a great deal was learned from. Dealing with trade off issues between the various component configurations was important to realize the optimum design.

8.4 Design Process

While many of the senior team members had taken advanced design courses, a lesson that was learned early was that the classroom is an ideal. With a short term project, it was found that less complex designs which enabled more simple analysis and took less time to manufacture were desirable. Experience gained from the design classes, combined with practical experience gained by last year's team members provided knowledge that was shared by the newer team members to learn from and apply to their efforts for this year. By providing this knowledge openly, the full talent of every team member could be used most effectively.

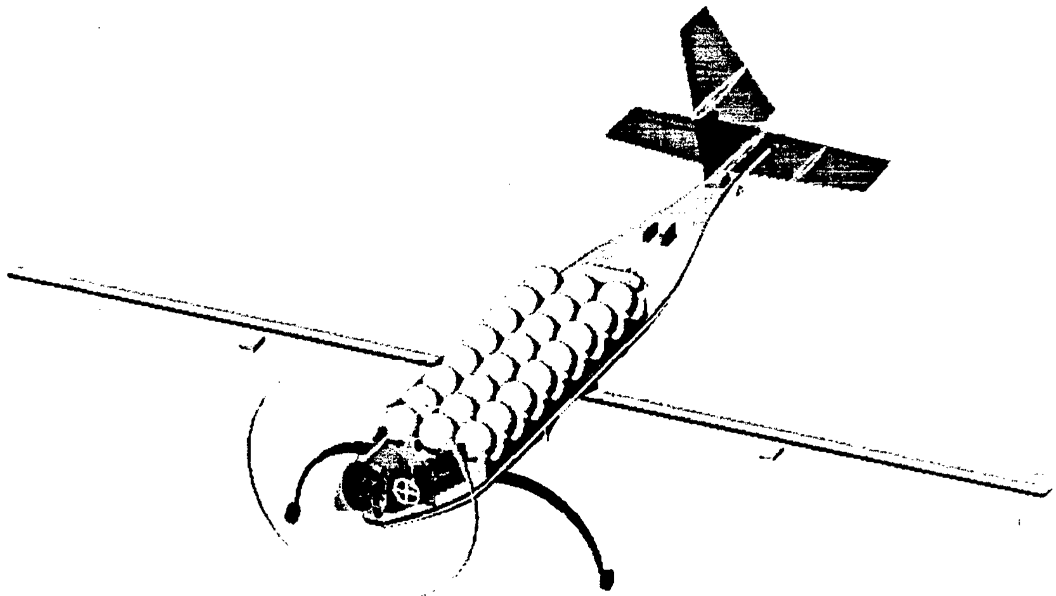
8.5 Theoretical vs. Actual

A lesson that was learned last year and re-affirmed with this year's design was that the actual characteristics of TLAR III differed from the designed theoretical characteristics. With even the most careful manufacturing techniques, the components will inevitably change. For example, as the propeller diameter changes, so will the required landing gear length, as the weight increases or decreases so will the required engine thrust. These types of trade-offs were needed throughout the construction even with

the extensive factors of safety used during the proposal phase of the design. This is a very important point to realize for any design process.

8.6 Finance

Working within a set budget emulated a real-world design setting. Due to the limited budget, the team was required to rely heavily on analysis to ensure that fabrication went smoothly and to avoid errors that could potentially increase the amount of materials needed. One of the main fundraising techniques learned last year was how to create a well-written proposal package. In addition, establishing and maintaining good relationships with sponsors and keeping them up-to-date on the project was important.



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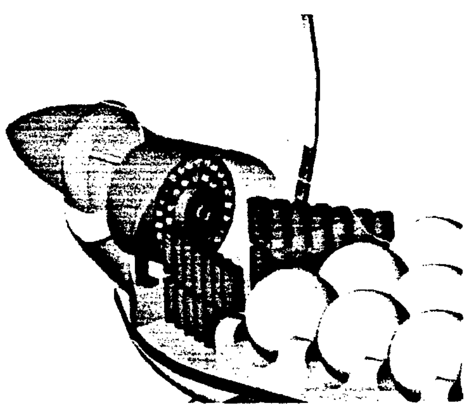
SCrewball

Cessna AIAA DBF Entry 2001-2002



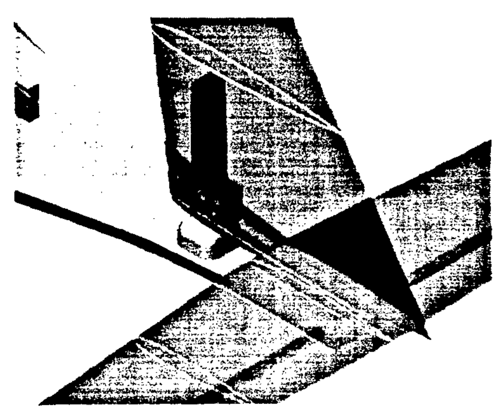


Table of Contents

1. EXECUTIVE SUMMARY

1.1. Introduction

The University of Southern California's Student Aero Design Team will participate in the 2001-2002 AIAA/Cessna/ONR Student Design/Build/Fly Competition. The competition calls for designing and building a propeller driven, remotely piloted aircraft using an electric motor. Each year, the competition presents new rules and regulations, which prevents teams from using the same exact design from a previous year for the contest mission. The goal of the Aero Design Team was to design and build a plane that will achieve the highest possible score in the contest. The final score is determined by three other factors: the mission score, the rated aircraft cost and the written report score. The schedule was broken down into five phases: Conceptual, Preliminary, and Detailed Design followed by construction and flight testing.

1.2. Overview of Design Tools Used

The mission simulator, called "Mission" was an Excel spreadsheet originally written to support the 1997/1998 AIAA DBF efforts. It consists of an Excel workbook with several spreadsheets that model various components of the aircraft during a scoring flight. It models all legs of the flight including: takeoff, cruise, turning flight, descent, landing, and ground handling. Some of the different modeling spreadsheets include: electrical, propulsion, weights, cost and aerodynamics. All these pages take geometry inputs from the user and rapidly evaluate different aircraft configurations and score the design based on payload carried and the cost of the craft. Some of these inputs are: airfoil type, wing area, propeller diameter and advance ratio, motor and gearbox type and count, battery type and count, and amount of payload carried. "Mission" derives its primary parameters from these inputs plus throttle setting, airspeed, initial altitude, aircraft weight, and load factor of each leg. This data from each sheet is used to calculate the energy consumption, time required, and altitude change for each leg. The output consists of the sum of the total energy and time consumed and predicts how many sorties will be completed given the initial parameters and constraints. The overall score is determined by multiplying the total number of sorties completed by the amount of payload carried then dividing by the calculated cost of the airplane.

Over the years, "Mission" has been optimized and modified to accurately predict performance based on previous years data. In addition, new abilities are added each year to "Mission". This year, one of the more important additions was the "Shotgun" feature. This feature would freeze several design parameters of the plane, and allow the user to then compare components such as number of motors or propeller selection with an identical plane in all cases. This feature helped the team to conduct trade studies with relative ease.

The following are all the models present in "Mission".

1. *Electrical model*: The electrical model is designed to analyze electrical motors used for small aircraft. The equations programmed into "Mission" follow the analysis presented by R.J. Boucher (1995). Inputs into this page include: throttle setting for each leg, propeller diameter and pitch, number of battery cells, volts per cell etc. taken from the input page of "Mission". Using the selected motor, battery, and gearbox type, it calculated the values of torque, voltage, current loaded and unloaded, and resistance. The electrical model sets an initial value of current, and the propeller page provides efficiencies and coefficients of thrust and power. It then feeds this data into a set of formulae to calculate thrust, motor RPM, and power. A feature in Excel called "goal seek" then adjusts current draw until it matches the current needed to spin the propeller at the desired RPM. This current is then used to find the energy consumption, thrust produced, and total system efficiency to export to the front page of "Mission". A correction factor of 0.65 is applied to the theoretical battery energy to improve correlation with flight test.
2. *Propeller model*: This model takes the propeller diameter and pitch and generates a map of thrust and power coefficients, C_T and C_p respectively. A plot of these coefficients and efficiencies (Figure 4.4) was generated from this data and brought to the test flights for comparisons with actual measured thrust. During test flights, differences between the predicted and measured thrust resulted in 1.1 power coefficient multiplier added to the propeller model.
3. *Weights model*: The weight of the plane is calculated by summing the weights of the individual components. For the constructed materials, a weight estimate was calculated from the volume of material used in the design multiplied by the density of the specific material. The weights of the pre-manufactured parts, i.e. motor, propeller, batteries etc., were stored in their respective models. The weight page looks up the component weights from each of the component pages and adds them to find the total weight of the aircraft. The center-of-gravity is also calculated.
4. *Stability and Control*: Handling characteristics were displayed in the stability and control worksheet. The worksheet takes inputs from the main page and displays flight conditions at cruise and at minimum airspeed, ground handling, and stability characteristics of the plane. In addition, the worksheet displays a two dimensional drawing of the current plane and shows data such as the airplane's C.G., heavy and light lap C.G. configurations, and the static margin. The aircraft had to pass all requirements set in the S&C page, such as yaw frequency, pitch stability, and the max allowable crosswind for takeoff and landing. Cruise wing incidence, required wing washout, and gear rigging are also addressed in the stability & control worksheet.

1.3. Conceptual Design

During this design phase, all possible configurations of the aircraft were discussed. The primary design tool used during the conceptual design phase was "Mission".

Design Alternatives Investigated

Initially, the team considered five different planes configurations at the start of this design phase: conventional plane with tricycle gear, flying wing, conventional with tail dragger, and biplane. For figures of merit, the team used sortie time, structural robustness, and rated cost to evaluate each design. Design parameters within these FOM's were also used to further break down and evaluate all five configurations.

Results

After careful review of each concept, the team decided on one design to be further studied in the preliminary design phase. The conventional plane with tail dragger was the design chosen to be further studied and configured during the preliminary design phase.

1.4 Preliminary Design

Many different components for the aircraft where discussed and analyzed during this phase. The primary design tool used was "MISSION".

Design Alternatives Investigated

Further trade studies were conducted on the monoplane configuration that resulted from the Conceptual Design Phase. Specific items such as payload to be carried, construction materials needed, number of motors, landing gear configuration, airfoils, gearbox, prop diameter & pitch, and number of battery cells were addressed during the Preliminary Design Phase. The figures of merit generated in the Conceptual Design Phase were used during the Preliminary Design Phase to evaluate all designs.

Results

At the end of the design phase, a conventional low-wing aft tail monoplane was selected. The plane will carry 24 softballs (for maximum flight score),, use only one Cobalt 60 motor with a 2.7:1 gearbox (major cost benefit), driving a 24-24 prop (big diameter for high static thrust, low pitch for high straight speeds & efficiency), have a design takeoff field length of 150ft, and a wing area of 6.61ft^2, with a wingspan of 9ft and carry 30 Ni-Cad cells for power. A tail wheel configuration was also decided upon for weight and drag savings.

1.5 Detailed Design

The finalized sizing for every component was done using the "Mission" design tool.

Design Alternatives Investigated

The team studied several alternatives in this design stage such as a full flying vertical vs conventional rudder, number of batteries to be used, and alternatives to the cooling system which used combinations of heat sinks, different cooling areas, and exhaust areas. The final plane has a full flying vertical tail for better crosswind at takeoff capabilities and 30 Ni-Cad batteries and a cooling system which includes an aluminum honeycomb heat sink around the motor, an air intake at the front of the aircraft, exhaust vents on the sides of the forward fuselage, and battery placement along the sidewalls of the forward fuselage.

2. Management Summary

2.1 The Aero Design Team

USC's Aero Design Team consisted of five faculty and industry advisors, a core group of experienced upperclassmen, and a number of new members. Dr. Ron Blackwelder and Mark Page served as the faculty advisors. Wyatt Sadler and Stuart Sechrist, both from Aerovironment Inc., and Nathaniel Palmer from Raytheon were the team's industry advisors. The student leaders included David Lazzara, George Cano, Jacob Evert, Cheng-Yuan(Jerry) Chen, Phillippe Kassouf, Charles Heintz, George Sechrist, Tim Bentley, Michael Mace, Jonathan Hartley, Tai Merzel, and Stephane Gallet. The team was roughly divided into Design/Report and Construction sections and had a semiformal command structure. Positions were assigned according to experience and interest; the responsibilities were flexible, with team members helping out when and where needed.

David Lazzara served as the overall student leader for the team. His responsibilities included setting the agenda for the weekly meetings, generating a schedule with input from each captain, and to oversee all of the design phases. Dr. Ron Blackwelder also served as our liaison with the university and the contest. Mark Page, an adjunct professor at the university who is also chief aerodynamicist at Swift Racing, advised the team on aerodynamics and helped develop the analysis spreadsheet. Stuart Sechrist, Wyatt Sadler, and Nathaniel Palmer, all participants of past DBF competitions, helped provide assistance in constructing the plane. Wyatt Sadler coached the build teams and served as the primary pilot, with Jerry Chen as the back-up pilot. Nathan Palmer coached the students involved in software development.

The team was organized into groups each having two to four students. Group captains were upperclassmen and each group contained some underclassmen. One of the more experienced members was chosen as group captain and had the responsibility to organize the group and insure that the group's deadlines and goals were achieved. Weekly team meetings were held for everyone to discuss overall problems, discuss common issues, and inform everyone of the team's progress. Initially these meetings concentrated on developing the aircraft design and setting a schedule. As the design matured, topics changed to construction issues and later to test flying. The group leaders and the team captain met separately one additional time during the week to discuss management and personnel issues. The main groups and their responsibilities follow.

2.2 Group and Personnel Assignments

Conceptual, Preliminary, and Detailed Design

The Conceptual, Preliminary, and Detailed Design sections were lead by David Lazzara. The areas of responsibility under this task were coordination of Information Technology, Propulsion Testing, Design Spreadsheet Development, Weights and Balance, Structural Sizing, and Configuration and Structural Design. In addition, further responsibilities involved managing the entire design and construction schedules as well as designating dates of important events throughout each phase.

Critical Design Review

A Critical Design Review was held on November 19th, 2001. Reviewers from Boeing Phantom Works were invited who had considerable AIAA DBF experience. All team Captains presented a problem statement, design criteria, alternatives considered analysis process, and design selection for their area. This proved to be quite helpful. Critique from the reviewers led the team to revise the wing position from above the fuselage plank to below, and encouraged the development of a no-spar wing design.

Report Captain

The report captain was responsible for taking minutes at every meeting, as well as preparing the report at an early stage. This included submitting rough drafts of each design phase a week after the phase was completed. In addition, editing the final report was the responsibility of the report captain. George Cano was chosen to carry out this task, with additional editing assistance from Dr. Ron Blackwelder, Mark Page, and David Lazzara.

Information Technology and Design Spreadsheet Development

The Informational Technology group, led by Phillippe Kassouf, created the computer graphics in the report, and generated isometric views of the aircraft during each design phase. The Design Spreadsheet Development group updated, refined and integrated the various analysis spreadsheets used to design the plane. Jake Evert, Nathaniel Palmer, and Andres Figueroa were responsible for these tasks, with considerable aid from Mark Page.

Configuration and Structural Design/Sizing

The Configuration and Structural Design group was responsible for the overall design of the plane using SolidWorks CAD software. Phillippe Kassouf was the configurator of the plane throughout each stage of development, with assistance provided by Stephane Gallet for sub components of the aircraft.

Construction

The construction captain was George Sechrist with assistance provided by Wyatt Sadler. The construction captain ran the laboratory and insured that component group had their part finished by their designated deadline. The construction group was assisted by a number of underclassmen that lent their time when needed. These people include Tyler Golightly, Tim Schoen, Jonathan Hartley, Stephane Galley, Tai Merzel, Doris Pease, Andres Figueroa, Cristina Nichitean, Stephanie Hunt, Billy Kaplan, Tim Phillips, Brian Wetzel, Lester Kang, and Boris Gee. In addition, several smaller projects were assigned to this group to involve the newer members of the team. These projects included designing and constructing parts such as an aircraft tow bar and a center of gravity test apparatus.

Aero Surfaces and Fuselage

This group was responsible for constructing the aero surfaces and the fuselage including the wing, fuselage plank, fuselage shell, and all aero-farings required for the aircraft. George Sechrist was captain of this group.

Landing Gear and Lab Manager

The Landing Gear and Brakes group was responsible for constructing, testing and installing the tail and main landing gear, brakes, wheels and steering system. Charles Heintz was the group captain for these tasks.

Controls and Propulsion

The Controls group took responsibility for the radio and servo systems, as well as the servo linkages. The Propulsion group was tasked with constructing and testing the entire power plant and power supply, as well as designing and constructing the motor mount. Jerry Chen lead both of these groups with aid from Tim Bentley for propulsion and Stephane Gallet for the motor mount.

Flight Test

The Flight Test group was in charge of organizing the test flight, as well as collecting and processing data. The group devised a flight test procedure used during the flight test that included the necessary setup required for the aircraft, preflight check on subsystems, and post-flight feedback from the pilot. Michael Mace was the captain for this group, with assistance from David Lazzara and Wyatt Sadler.

2.3 Configuration and Schedule Control

To visualize the deadlines for every phase of design and construction, a schedule was set for the team to follow (Figure 2.1). Sufficient time was allowed for schedule delays so that paths critical paths for completion could be maintained. For example, much of the construction was planned to be completed in December but did not need to be finished until late January in time for flight testing.The team leader updated and maintained the schedule every week, and informed the group on approaching deadlines.

The team held meetings every Monday to discuss the progression of the aircraft design and construction phases. In addition, weekly Friday meetings for the group leaders were held during the design phases to discuss changes made to the current aircraft configuration. These changes were incorporated into new configuration drawings to be shown at the team meeting the following week. The team created a web page that provided information such as a team member contact information list, a copy of the latest schedule for download, and weekly meeting agendas for members who were unable to attend.

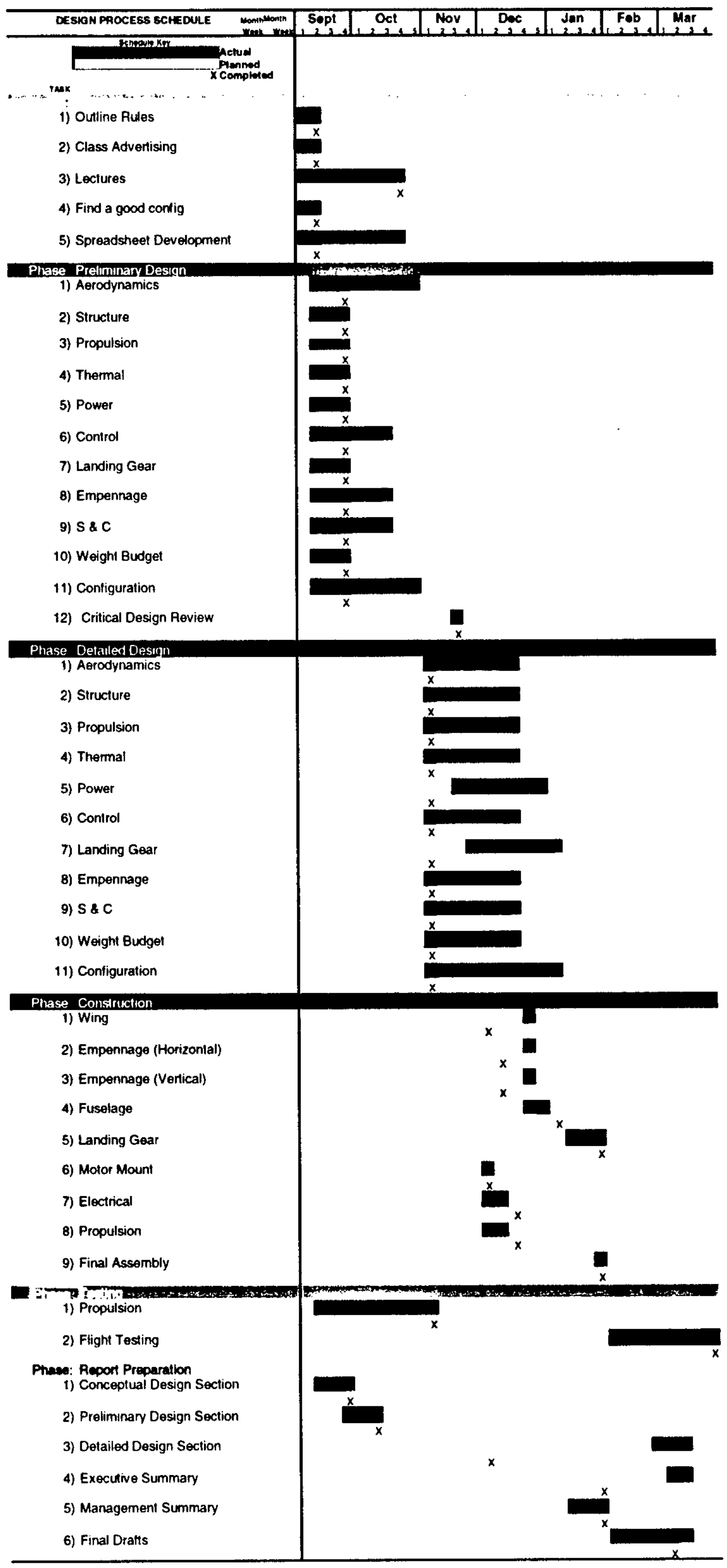


Figure 2.1. Management Schedule and Milestone Chart

3. Conceptual Design Phase

Five aircraft were defined to address the 2001/2002 Design/Build/Fly rules; conventional with tricycle gear, flying wing, conventional tail-dragger, biplane tricycle, and a biplane tail dragger. Three *Figures of Merit* (FOM) were selected; Sortie Time, Structural Robustness, and Rated Aircraft Cost. In addition, eight key assumptions (listed below) were made that applied to all five designs. These conceptual designs were all drawn with the following assumptions so that wetted area and complexity assessments could be made. Each subjective FOM was awarded a 1 to 5 rating. The rated aircraft cost was added to the subjective FOM's for the final ranking.

3.1 Assumptions Made and Justifications for their Validity

The following ground rules were applied to all five candidates for a fair comparison.

1. *10-24 Softball Payload Capacity*: The contest rules mandated this payload requirement. Each conceptual configuration required sufficient volume for this payload range.
2. *Aircraft Center of Gravity (CG) coincident with Payload CG at the Wing c/4*: The wing center of lift was taken to be at the root quarter-chord location. The payload centroid was located to be coincident with the quarter chord. This minimizes C.G. variation with and without payload.
3. *Composite Materials for Construction*: The high strength-to-weight ratios of composite materials, such as carbon-fiber and honeycomb core material, minimized weight while protecting the aircraft against any design load scenario.
4. *1 to 3 Motors Required:* To meet the 200' TOFL requirement, each conceptual configuration needed the ability to house 1 to 3 motors.
5. *10 ft^2 Wing Area if Empennage Used; 20 ft^2 Wing Area if No Empennage Used*: The wing area was similar to that used by the 2000-2001 USC *DBF* team due to comparable payload mass requirements. Configurations that lacked an empennage were required twice the wing area. This accounts for halving the trimmed $C_{l,max}$ due to the negative flap angles for trim. Otherwise, a conventional elevator with a large lever arm to the CG was assumed.
6. *Spacing between Payload, Fuselage Bulkheads, Batteries, and Motors are Constant*: These components were equally spaced fore and aft of the CG to negate any differences in longitudinal moments among the five conceptual aircraft because they dominated the weight and balance considerations. Fuselage designs that enclosed these components were variable.
7. *150' TOFL*: The 2000-2001 USC *DBF* Team experimentally determined an uncertainty of ±50' in TOFL for an aircraft designed with payload mass requirements similar to the 2001-2002 *DBF* payload rules. This maximum TOFL minimized the possibility of overshooting the maximum allowed 200' TOFL set by the 2001-2002 contest rules.

8. *5 lb. Battery Weight Capacity*: In addition to meeting the maximum allowable battery weight condition in the 2001-2002 contest rules, this limit ensured that each conceptual configuration could carry the maximum specific energy when needed.

3.2 Alternate Configurations Investigated

Five configurations were considered for the Conceptual Design Phase.

1. *Monoplane with Empennage & Tricycle Landing Gear*: This aircraft configuration resembled the USC 2000-2001 *DBF* contest aircraft due to similarities in the payload mass requirements and the mission profile with the 2001-2002 contest rules. This general configuration supported various fuselage designs, payload restraint and egress techniques, and wing locations above or below the payload compartment.
2. *Monoplane with Tricycle Landing Gear & without Empennage*: The flying wing configuration offered a potential cost advantage that could be exploited. A fuselage was replaced by locating the payload inside the wing center section, thereby reducing wing-body aerodynamic affects. Fewer control surfaces and servos reduced the aircraft weight as well. The use of a canard surface for pitch control was also considered.
3. *Monoplane with Empennage & Tail-Dragger Landing Gear*: This aircraft only varied from the first conventional Monoplane configuration above by replacing a nose gear with a tail wheel capable of steering. The biggest benefit of the tail wheel is removing the heavy nose gear and its dedicated servo. In the past, the USC team reasoned a nose gear would offer protection to the prop, gear, box and motor shaft in the case of a runway departure. In practice, the nose gear could fail and break the prop as well.
4. *Biplane with Empennage and Tricycle Landing Gear*: A conventional Biplane configuration varied from the conventional Monoplane by the addition of an extra wing. Vertical struts between the high and low wings would serve as structural members and reduce vortex drag by acting as winglets. Furthermore, with the total wing area and span held constant, the Biplane could achieve lower vortex drag than the conventional Monoplane configuration if its wings were separated substantially.
5. *Biplane with Empennage and Tail-Dragger Landing Gear*: This Biplane configuration varied from the previous biplane concept by replacing the nose gear with a tail wheel. The advantages/disadvantages are the same as given in the third conceptual design above. In addition, the wings achieved higher C_L at lower takeoff velocity due to the large angle of attack resulting from the lowered aft section of the fuselage.

3.3 Figures of Merit

To discriminate configurations with good performance and high reliability, the following FOM's were selected.

1. *Sortie Time*: The 2001-2002 contest rules specified a formula for determining a flight score that contained sortie time as a normalizing term. The inverse proportionality between flight score and sortie time provides maximum flight score with minimal sortie time for a given payload carried and sorties flown. A conceptual configuration could accomplish this by minimizing total aircraft C_D for higher flight velocity and decreasing the ground handling time.
2. *Structural Robustness*: Structural restrictions were set in order for the conceptual aircraft to survive a host of scenarios where aircraft damage was most likely to occur. The extent of damage was a function of the aircraft structural integrity. Structural integrity was deemed excellent for configurations with minimal number of joints or other geometric discontinuity. Such features are vulnerable to crash loads even if they are adequate for flight loads. The number of structural discontinuities was used to indicate poor robustness.
3. *Rated Aircraft Cost*: The 2001-2002 contest rules define a total score formula with a normalizing term that penalized complex designs., This parameter is a function of the number and size of components on the aircraft used to accomplish the mission profile. In summary, lower costs are achieved by using the fewest motors for propulsion with the lowest battery specific energy, the smallest wing area for lift, the minimal number of actuators for aircraft control, and the leanest MTOGW. The inverse proportionality of the *Rated Aircraft Cost* to total score indicated that the lowest possible cost parameter would maximize the total score, for a given flight score and report score.

Further quantification of these FOM's was necessary to fairly compare configurations. The following discussion shows how the 3 FOM's were broken down into finer discriminators.

3.4 Design Parameters Quantifying the *Sortie Time* Figure of Merit

The *Sortie Time* FOM was specifically defined with design parameters that had the greatest influence on the overall time required to complete a mission. These parameters were explained as follows:

1. *Minimum Total Aircraft C_D*: This parameter is a linear combination of the aircraft parasite drag and vortex drag. Reducing the total aircraft C_D increases max cruise speed for a fixed available thrust.
2. *Minimum Taxi and Payload Swap Time*: Taxi time was determined by the ability of a conceptual design to maneuver under high crosswind conditions. Unacceptable ground handling in such conditions could cause the aircraft to roll off the runway or increase the necessary time to correct for overshoot on a landing approach. The affect of using brakes

was deferred for study in a later design phase. Payload swap time was a function of how easily the payload insertion and egress was possible with a given configuration. Mitigating the interference of wings, propellers, and empennage with respect to the payload hatch access was an important consideration. Minimal mechanisms and restraint devices were necessary for installing and removing payload efficiently.

3.5 Design Parameters Quantifying the *Structural Robustness* Figure of Merit

1. *Minimal Structural Discontinuities*: The conceptual aircraft needed to withstand all design loads with the fewest structural members to satisfy this requirement. Fewer junctions between load bearing components minimized the need for reinforcing such attachment locations. The assumption of composite material construction enabled unique structural designs that did not rely on trusses or various load bearing members for the distribution of load paths. Instead, a limited number of load paths among fewer load-bearing members reduced the need for complex construction in the conceptual designs.
2. *Efficient Distribution of Load Paths*: Long load paths between a load source and the structural components of the conceptual designs were deemed inefficient due to the exaggerated stresses caused by large moment arms. These large stresses required larger area moments of inertia among the load bearing members to provide ample reaction loads up to the ultimate design load limits. Load distribution throughout skins and shells provided torsional rigidity that structural beam components would lack, in general. The use of efficient load paths through skins and specific structural components in areas of concentrated loads, such as attachment point regions, increased the strength to weight ratio of the structural design by minimizing structural weight.

3.6 Design Parameters Quantifying the *Rated Aircraft Cost* Figure of Merit

Optimizing the *Rated Aircraft Cost* FOM required minimizing the use of aircraft components that received the greatest penalty due to their geometry or quantity. This was also reflected in an improvement in the overall performance of the aircraft in accomplishing the mission profile. The geometry and quantity of the following four aircraft components were considered the most important design parameters for the *Rated Aircraft Cost* FOM:

1. *Propulsion System*: The number of motors and the total battery weight influenced a large percentage of the *Rated Aircraft Cost* because its weighted value was one order of magnitude larger than any other parameter in the cost formula. The balance between cost penalties and performance was critical; sufficient thrust was needed to meet TOFL limits mandated by the contest rules and ample specific energy was necessary to complete the three sorties in the mission profile.

2. *Primary Lifting Mechanism*: The number of wings used, increased wing area and the number of control surfaces used to generate lift were penalized. These issues required intelligent compromise with performance criteria such as TOFL requirements, C_L, $C_{L,\alpha}$, and both longitudinal and lateral stability. In addition, the number of control surfaces was related to the number of required actuators, which added a small percentage to the *Rated Aircraft Cost* in comparison.
3. *Empennage*: The means for providing pitch and yaw authority were the function of the horizontal and vertical stabilizer, respectively. The number of elevators and rudders used on these control surfaces also contributed to the total score reduction. Although not directly accounted for in the cost formula, the contest rules also defined an upper span limit for the horizontal stabilizer that was $0.25b_w$.
4. *Mean Takeoff Gross Weight*: The penalties associated by the *Primary Lift Mechanisms* and *Empennage* were of the same order of magnitude as the penalty for average takeoff gross weight of the aircraft without payload. This penalty was only minimized through the use of high strength to weight ratio materials, efficient structural design, and high specific energy from the propulsion system. Improving the affect of this parameter also assisted in reducing the aircraft wing loading and TOFL, which increased turning rates and reduced the power required for liftoff.

The five conceptual configurations were analyzed to judge their technical merit before converging on a single conceptual design and initiating the Preliminary Design Phase.

3.7 Configuration Downselect

Derivative Trade Study Results: Trade studies were conducted on derivative designs of the five conceptual configurations. These studies varied the number of motors used, the use of winglets (or interplanar struts, the payload quantity, and the affect of non-empennage and canard utilizing configurations. These studies used the USC developed "MISSION" simulation package. "MISSION" sizes wing area and cruise speed to satisfy takeoff field length and cruise altitude constraints. "Mission" results include: cost, sortie time, gross weight, percentage of excess energy, and both L/D_{crz} and L/D_{turn}. This full simulation allowed the fixed $S_w = 10$ ft^2 assumption to be waived. Thus, in terms of S_w, a trend was illustrated throughout the trade studies of 4 ft$^2 < S_w <$ 8 ft^2. Such results were initial clues for converging subsystem sizing in the Preliminary and Detailed Design Phases. Other trends and conclusions derived from the trade studies were summarized below.

Variable Number of Motors Used: Each conceptual design and their derivatives provided the same conclusive information about the optimal number of motors. Figure 3.1 depicted a continual decrease of 25% for each additional motor. Thus the optimal total score was possible with the use of only one motor. The additional thrust requirements for TOFL are addressed in the preliminary design section.

Variable Winglet Height: Winglet height was varied for various $S_{winglet}$ values, such as 1 ft^2 and greater. Figure 3.2 illustrates a minimal change in total score for various winglet heights at $S_{winglet} = 1$ ft^2. This was attributed to the L/D_{turn} increase with winglet height. However, the reduction in vortex drag was countered by the increase in parasite drag and weight due to the presence of the winglet. The trade studies that utilized winglets consistently provided a total score that was approximately 20% lower than the total scores obtained without the use of winglets. Thus an increase in b_w was the only primary option that was variable for reducing vortex drag at the conclusion of this trade study.

Variable Payload Quantity: This trade study was conducted to determine the payload within the 10 to 24 softball capacity range appropriate for the highest total score. Each configuration simulated in the MISSION software concluded that only 24 softballs provided the highest total score. Figure 3.3 showed the monotonically increasing total score, where only 50% of the maximum score was obtained at the lowest payload capacity. The possibility of obtaining a higher total score with a payload capacity of 10 softballs and much faster sortie time was considered as well. Trends indicated that a large reduction in aircraft size was necessary to sufficiently decrease the sortie time and reduce the overall cost. In order to match the score of a 24 softball payload the following two alternatives were studied.

1. *Case 1:* Rated Aircraft Cost must be reduced by at least 1% per unused softball *and* flight velocity increased for a faster sortie time.
2. *Case 2:* Sortie time must be reduced by almost 50% at a rated cost similar to the 24 ball case. The second case required flight velocities near 100 mph. Optimizing the propulsion system to accomplish this required an exotic gearbox and propeller. Thus, a 24 softball payload capacity was selected.

Flying Wing and Canard Alternatives: The affect of having no empennage on the conceptual design aircraft posed critical performance considerations. To maintain $\partial C_{m,cg}/\partial\alpha < 0$, the condition for longitudinal stability, negative flap angles were necessary to trim at higher lift coefficients. This reduces trim $C_{L,max}$ at any given α by approximately 50% of the trim $C_{L,max}$ on a configuration with an empennage. This trade study also incorporated a canard control surface ahead of the wing. The canard contained two major flaws that inhibited maximum performance of the conceptual aircraft, namely downwash from the canard would reduce the effective angle of attack on the center section of the wing, where the majority of lift was generated, and both the wing flaps and canard contained approximately equal moment arms about the CG. The weight of the canard and its supporting structure moved the aircraft CG forward, thereby requiring the wing to substantially reduce lift with negative flap angles or airfoil reflex for a pitch-down moment. This design derivative was inherently unstable longitudinally, and required a large aft wing shift to recover acceptable stability. At the conclusion of this trade study the efficient pitch and yaw control available from

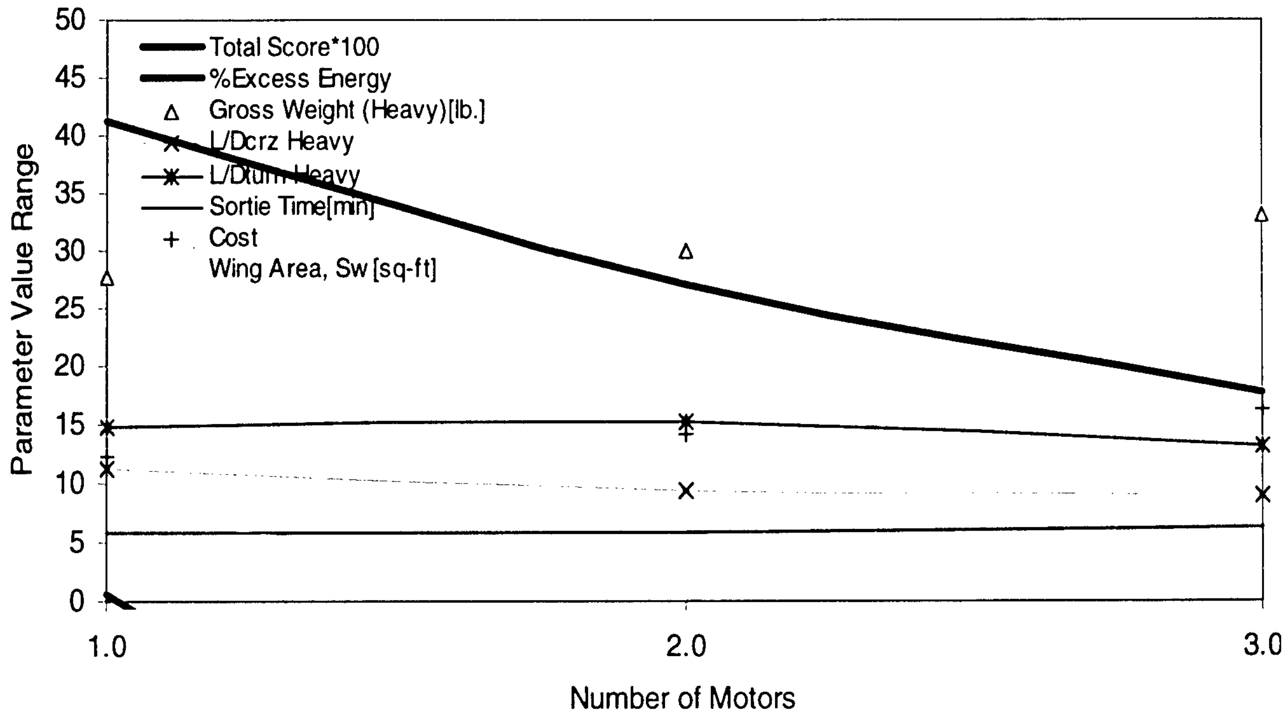


Figure 3.1. Trade study result for variable motor number.

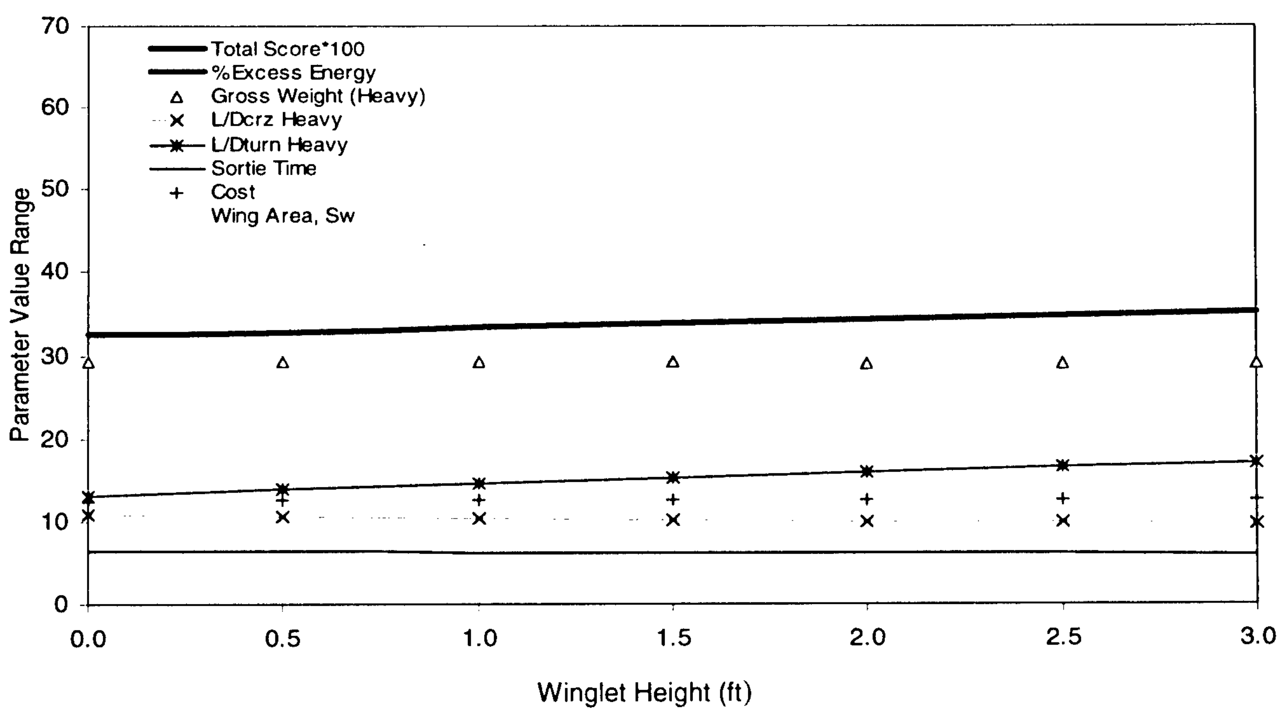


Figure 3.2. Trade study results for variable winglet height at S_w=1 ft^2.

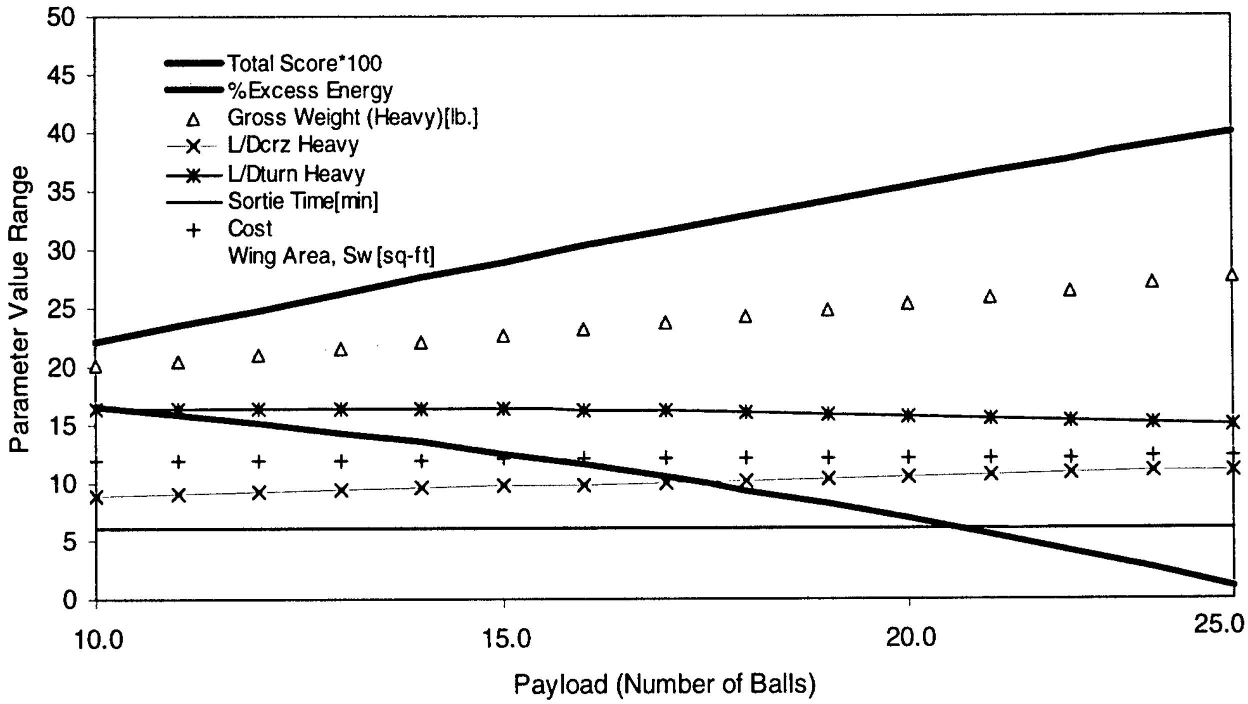


Figure 3.3. Payload capacity trade study results.

empennage was deemed most beneficial due to the longer moment arms about the CG and inherent longitudinal stability.

Monoplane and Biplane Comparison: A trade study initially indicated that the total score possible for a biplane was equivalent to that of a monoplane. Compared to the monoplane, the sortie time was improved and flight velocity was greater. In reality, though, ample vertical spacing between the wings was necessary to reduce the downwash interference and vertical struts at the wingtips were also needed to reduce vortex drag. These changes required a tall fuselage configuration, an extra junction for mounting the second wing, and structure at the wingtips for the wing strut attachments. These additions added structural discontinuities and complicated the load path correlation throughout the wings and fuselage. Furthermore, payload restraint and egress became inefficient with the interference posed by a second wing. Tests were conducted to validate the payload restraint and egress time with different hatch configurations on a fuselage. The test results demonstrated that the most influential parameter for efficient payload handling was a single-hinged hatch located on the fuselage topside; however, such a payload hatch was inaccessible with the presence of a high wing on a biplane.

The possibility of the biplane achieving the same maximum score as a monoplane was unlikely also because of the added structural discontinuities. Inefficient load paths would require reinforcements that were not modeled in the "MISSION" simulation, thereby increasing the true MTOGW of the aircraft and the cost parameter. Furthermore, two identical wings were required to accomplish the same mission

that a single wing was capable of completing. Considering the time frame, the optimization of a single wing was more practical than the construction and validation of two wings. The total aircraft C_D was expected to improve with the biplane; yet any vortex drag savings earned with sufficient vertical wing spacing was negated by the added frictional drag on the fuselage due to the substantial fuselage length required to recover the pressure losses from a blunt fuselage forward section. Designs with more than one fuselage to divide the payload were also unacceptable due to the increase in the aircraft parasite drag with greater surface area. Furthermore, using wing struts also added to the aircraft parasite drag and weight. These issues eliminated the biplane configurations from any further design considerations.

Landing Gear Configuration Comparison: For all conceptual designs, the tail wheel landing gear reduced the cost parameter in comparison to designs that incorporated a nose gear. The tail wheel configuration was less massive than a nose gear because under taxing conditions it typically bore only 4%-7% of the aircraft weight, compared to the 10%-15% for a nose gear system. A nose gear also required extraneous structure to provide sufficient propeller ground clearance; in contrast, the tail wheel offered improved propeller ground clearance by lowering the aft end of the fuselage. Propeller strike was also limited in a tail wheel configuration by locating the main landing gear forward of the CG until the moment arm between the main landing gear and the aircraft center of gravity mitigated a downward pitching moment during deceleration. The tail wheel configuration also contained a larger moment arm from the main landing gear than a nose gear system. This superior steering authority allowed the aircraft to direct itself windward during crosswind perturbations. These findings led to the selection of a tail wheel.

3.8 *Rated Aircraft Cost* Comparison with Conceptual Design Rankings and Final Design Features

The foregoing trade studies led to the selection of a single motor and a full 24 ball payload, Table 3.1 summarized the five refined conceptual configurations with their respective FOM rankings. The *Sortie Time* and *Structural Robustness* FOM were each subdivided into two sub-FOM and assigned values of 1-5 where 5 is the superior design. The total score for both FOM's was the sum of its two sub components. Each FOM was given a 1-10 rating, where 10 was the best design. The *Rated Aircraft Cost* FOM value in Table 3.1 was obtained from the "Mission" spreadsheet for each of the configurations; e.g. the classical empennage with nose wheel had a RAC of 12.28. The breakdown showed the percentage of the total cost contributed by each cost design parameter. These percentages did not sum to 100% because other lower percentage cost design parameters were not listed. The Total Score in Table 3.1 was the sum of the Sortie Time and Structural FOMs divided by the RAC. This provided an analytical measure of the design rankings.

Rated Aircraft Cost Comparison: The flying wing configuration had insufficient energy to complete the mission due to the increased vortex drag of the reflexed airfoil and parasite drag of the massive wing needed for TOFL. However, the final cost for this configuration was the lowest of all because it did not have an empennage. The monoplane with empennage and a tail wheel had the

second lowest total cost, followed by the monoplane with empennage and nose wheel. The tail wheel landing gear configuration decreased MTOGW and reduced overall cost. The biplane configurations

Table 3.1 Final Conceptual Design Rankings and Rated Aircraft Cost Summary

	Monoplane			Biplane	
	Classical Empennage & Nose Wheel	*Flying Wing with Nose Wheel*	*Classical Empennage & Tail Wheel*	*Classical Empennage & Nose Wheel*	*Classical Empennage & Tail Wheel*
***Sortie Time* FOM**	**7**	**6**	**10**	**5**	**6**
• Minimum Aircraft C_D	*3*	*2*	*5*	*3*	*4*
• Minimum Taxi & Payload Exchange Time	*4*	*4*	*5*	*2*	*2*
***Structural Robustness* FOM**	**7**	**9**	**9**	**5**	**5**
• Minimal Structural Discontinuities	*3*	*5*	*4*	*2*	*2*
• Efficient Load Path Distribution	*4*	*4*	*5*	*3*	*3*
***Rated Aircraft Cost* FOM**	**12.28**	**11.68**	**12.26**	**12.41**	**12.39**
• Propulsion System	*50.8%*	*53.4%*	*50.8%*	*50.2%*	*50.3%*
• Primary Lifting Mechanism	*13.7%*	*14.4%*	*13.8%*	*13.6%*	*13.6%*
• Empennage	*3.3%*	*0.0%*	*3.3%*	*4.8%*	*4.8%*
• MTOGW	*14.6%*	*15.3%*	*14.4%*	*14.2%*	*14.0%*
TOTAL SCORE	**1.14**	**1.28**	**1.55**	**0.81**	**0.89**

contained the highest total cost due to the use of vertical wing struts. Without the vertical wing struts, a 0.2 reduction in cost was achieved by lowering the Empennage design parameter percentage, causing the biplane to surpass the monoplane with empennage cost rating.

Conceptual Design Rankings: Despite the *Rated Aircraft Cost* results, the *Sortie Time* and *Structural Robustness* FOM were the decisive FOM in the final configuration selection. The *Total Score* depicted in Table 3.1 reflected the ratio between the summation of *Sortie Time* and *Structural Robustness* FOM values to the *Rated Aircraft Cost*. The minimal aircraft C_D was heavily influenced by the vortex drag in a configuration, the parasite drag of a nose gear system and skin friction drag from excessive surface area. The taxi and payload exchange design parameter was generally a function of payload access and the superior steering capabilities of a tail wheel system rather than a nose gear system. Structural discontinuities and efficient load paths were characterized by the number of attachment locations in a

configuration, the possibility of concentrated loads with large moment arms, and the inefficient use of skins or load bearing members for load path transmission.

The biplane configurations contained the most structural discontinuities and inefficient load paths due to redundant wing structure, whereas the flying wing configuration was deemed the most efficient design for structural efficiency due to less attachment locations.

Final Configuration Features: The monoplane with empennage and a tail wheel configuration received the highest final score in the Conceptual Design Phase. This design achieved the best *Sortie Time* FOM rating due to the lower parasite drag of a tail wheel system, minimal wing parasite drag by virtue of the high trimmed C_{Lmax}, and minimal vortex drag since no outboard airfoil reflex was needed for trim. The ability to install and remove payload from the fuselage topside with minimal interference also provided a reduction in payload exchange time. The *Structural Robustness* FOM was satisfied by the final configuration as well. Even though an empennage was used for the final configuration, there was only one wing with no struts or winglets, and no nose gear.

The *Rated Aircraft Cost* FOM was optimized using the final configuration and conclusions derived from the derivative subsystem trade studies. A minimal cost parameter was obtained by using a single motor for propulsion, no winglets or nose gear, and efficient structural design for lower MTOGW.

The maximum total score was achieved by containing the maximum payload capacity within a single fuselage. The monoplane with a classical empennage and tail wheel was selected for further development in the Preliminary Design Phase.

4. Preliminary Design Phase

This second phase in the design process was organized to conduct further trade studies for each subsystem on the monoplane configuration derived in the Conceptual Design Phase. These trade studies incorporated a different set of assumptions and design parameters; however, the Conceptual Design Phase FOM's were not changed. Wing and Power loading requirements were established during the trade studies to optimize each FOM.

4.1 Revised Assumptions

Assumptions used in the Conceptual Design Phase were valid for the Preliminary Design Phase in terms of defining a general aircraft configuration. The overall justification for these assumptions was explained in the Conceptual Design Phase as well. Modifications to these assumptions allowed a specific design focus on certain subsystems studied in the preliminary design. These reasons are summarized below:

1. *24 Softball Payload Capacity*: This payload capacity was a direct result of the Conceptual Design Phase trade studies which yielded the highest possible flight score for any given configuration and flight time.
2. *Aircraft Center of Gravity (CG) coincident with Payload CG at the Wing c/4*: Variable CG locations were studied in the stability analysis in the Detailed Design Phase. The coincidence of the CG with and without payload eases the stability and control issues as stated in the Conceptual Design section.
3. *Composite Materials for Construction*: This assumption was not changed because a materials trade study was necessary to select which composites were adequate to tolerate given design load conditions.
4. *Only 1 Motor Used:* Motor count was fixed because the Conceptual Design Phase associated a severe cost penalty for using more than one motor.
5. *Monoplane with $4\ ft^2 < S_w < 8\ ft^2$*: Previously, $S_w = 10$ ft^2 was assumed for the configuration downselect. Trade studies conducted in MISSION simulations during the Conceptual Design Phase converged on solutions with wing area to this specific range instead.
6. *Variable Spacing between Payload, Fuselage Bulkheads, Batteries, and Motors*: These internal components were no longer fixed in order to explore various subsystem locations within the fuselage.
7. *150' TOFL*: Until the aircraft was completed and true TOFL measurements were taken, the reasons stated in the Conceptual Design Phase sufficed for assuming this maximum TOFL.
8. *4-5 lb. Battery Pack Weight*: The battery pack weight assumption was relaxed to allow for various battery pack configurations that matched the specific energy needed to the power loading requirements.

9. *Tail-Wheel Landing Gear Configuration*: The Conceptual Design Phase trade studies provided sufficient evidence that this landing gear configuration was superior to a nose gear system.

A consideration of the FOM's in the Preliminary Design Phase was possible using this new set of assumptions.

4.2 Figures of Merit Summary

The same FOM's used in the Conceptual Design Phase were implemented in the Preliminary Design Phase. These FOM's were repeated below with an emphasis on the different trade studies that were developed for this design phase:

1. *Sortie Time*: New trade studies in aerodynamic efficiency quantified the aircraft C_D. Propulsion efficiency studies established the available and required aircraft power.
2. *Structural Robustness*: The wing, fuselage, and landing gear structure were considered the main structural components of the aircraft. Their structures were analyzed to optimize the distribution of load paths.
3. *Rated Aircraft Cost*: Cost was accounted for during iterations on fuselage design and necessary battery pack weight. A cost analysis was also done for the general sizing of the wing and MTOGW in addition to the performance analysis. This analysis changed the rated aircraft cost of the monoplane configuration rather insignificantly. A small percentage of the cost was eliminated due to the lower wing area assumption and lower battery pack weight. Small deviations from the cost parameter during subsequent analysis of structure, aerodynamic efficiency, and propulsive efficiency were not discussed.

These FOM's were refined for specific technical studies that involved the individual subsystems on the aircraft rather than the overall aircraft configuration.

4.3 Design Parameters & Sizing Trade Studies for Wing & Power Loading Requirements

The "Mission" simulation software was used to conduct the following trade studies for minimizing sortie time:

Aerodynamic Efficiency: Aircraft efficiency was defined by its L/D ratio both in cruise and in turning flight. An aerodynamics subroutine in the "Mission" simulation calculated the parasite drag for the landing gear, fuselage, wing-body junction, controls gaps, cooling, empennage-body junction, and wing at $C_L = 0.5$. Each non-wing component parasite drag was a weighted percentage of the airfoil parasite drag at the given C_L.

Airfoil selection was achieved by studying the relative C_D induced at a given C_L for different airfoils. Considerations included high lift capability to meet TOFL requirements, minimal C_D at high velocity, and gradual C_D increase when approaching $C_{L,max}$ at lower velocity. An airfoil database was compiled using a Selig web-based solver employing a panel-method calculation of the pressure

distribution, C_L vs. α, and C_{MO} vs. α for a given airfoil coordinate matrix. Six airfoils were studied, ranging from high speed airfoils to high lift airfoils: S1210, SD 6060, S1223, MH 45, FX74CL5140, SD 7032, and LA 203a. By integrating the pressure distribution around each airfoil at Re = 1,000,000, a collection of drag polars was compiled in Figure 4.1, including full flap deflection at approximately 22°. Both the SD 7032 and LA 203a airfoils had the lowest C_D for the largest range of C_L. However, the LA 203a was a high lift airfoil that lowered the necessary wing area to achieve TOFL. Predicted cruise and turn C_L were generally at lower C_D for the LA 203a than the SD 7032 as well. By selecting the LA 203a airfoil the wing parasite drag was minimized.

Wingspan was varied to study its effects on the aerodynamic parameters. Figure 4.2 indicates that a 9 ft. wingspan was optimal for an overall score. As expected from aerodynamic theory, higher AR wings reduced the vortex drag in cruise. This did not imply a small taper ratio, though, because the shedding vortex system could strengthen inboard from the wingtips and create a large upwash felt by the horizontal stabilizer. A negative pitching moment could ensue that could destabilized the aircraft. In addition, at $C_{L,max}$ the empennage could be in the stalled wing wake and lose control authority. A substantial taper ratio, on the order of 0.7, ensured a downwash in the vicinity of the horizontal stabilizer that created a positive pitching moment and kept the vortex system near the wingtips at $C_{L,max}$.

Furthermore, aerodynamic theory also predicted that an elliptical lift distribution was ideal for minimal induced drag. However, an elliptical wing planform was disregarded because the root chord increased substantially in order to maintain the above wingspan. Each trade study solution converged on a wing area that satisfied TOFL and reduced wing loading for lower turning radius and higher turning rate. These trends indicated that $S_w = 6.61$ ft^2 was sufficient. The aerodynamic performance curves were deferred for studies in the Detailed Design Phase.

Propulsive Efficiency: Available energy in the batteries needed efficient conversion to kinetic energy of the aircraft via thrust. Reductions in the available stored energy were accounted for by taking into account the temperature-dependent internal resistance of each battery cell, speed controller, motor, and wiring, as well as I^2R losses and back emf. This resulted in only 60% of the listed storage capacity of each battery cell being used when sizing the battery pack.

Proper motor, gear box, and propeller matching was a critical trade study because the appropriate propeller advance ratio and activity factor was required to match its best efficiency with the predicted flight velocities. Propeller RPM was directly related to the gear ratios in the gear box and the load it induced on the motor was limited by the 40 A maximum current draw. Various Astroflight motors were considered, each containing unique torque and speed constants. Higher torque and speed constants were important for conservation of battery energy because more torque was achieved per amp and higher RPMs were available for a given voltage. Low motor internal resistance was essential as well. Large pitch propellers required more current draw, whereas larger diameter propellers permitted faster tip speed for a given RPM, where much of the propeller load was concentrated due to the propeller twist. Current draw was also reduced with an appropriate gear box ratio, which increased the propeller RPMs

for a given motor RPM. Surface plots for various gear box, propeller, and motor configurations were generated using the "Mission" simulation software to find which combination provided the highest possible score for a given aircraft. Figure 4.3 represented the best scoring configuration consisting of an Astroflight Cobalt 60 motor, a 24-24 propeller, and a 2.7:1 gear box ratio. This configuration also scored the highest propulsive efficiency. The "Mission" simulation was unable to converge for solutions that used a smaller diameter propeller or advance ratio because TOFL requirements were unsatisfied despite a compromise with increasing wing area.

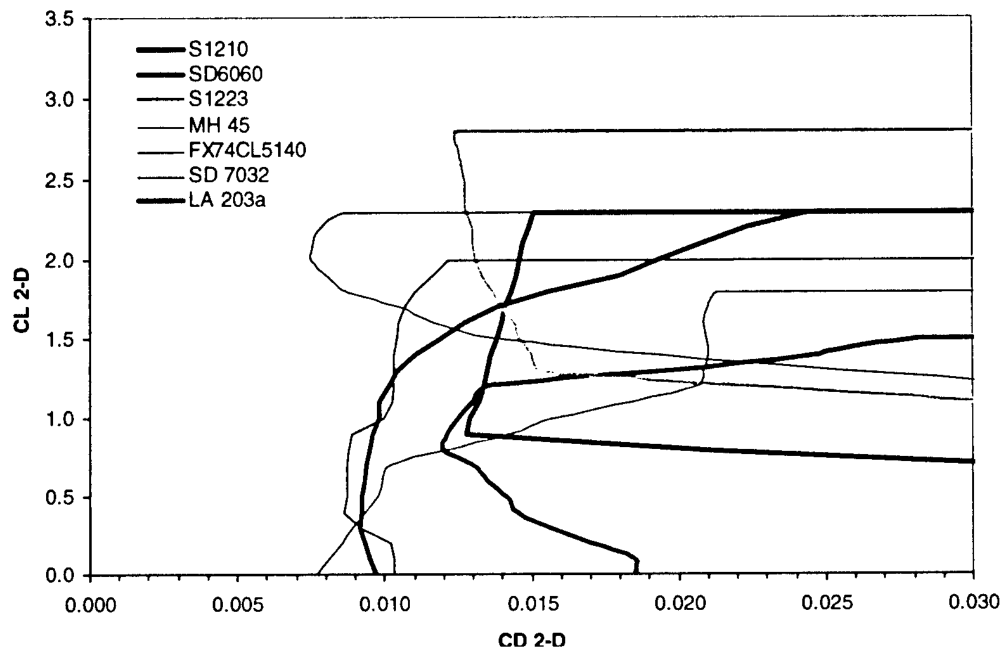


Figure 4.1 Airfoil drag polars, including geared flap effects.

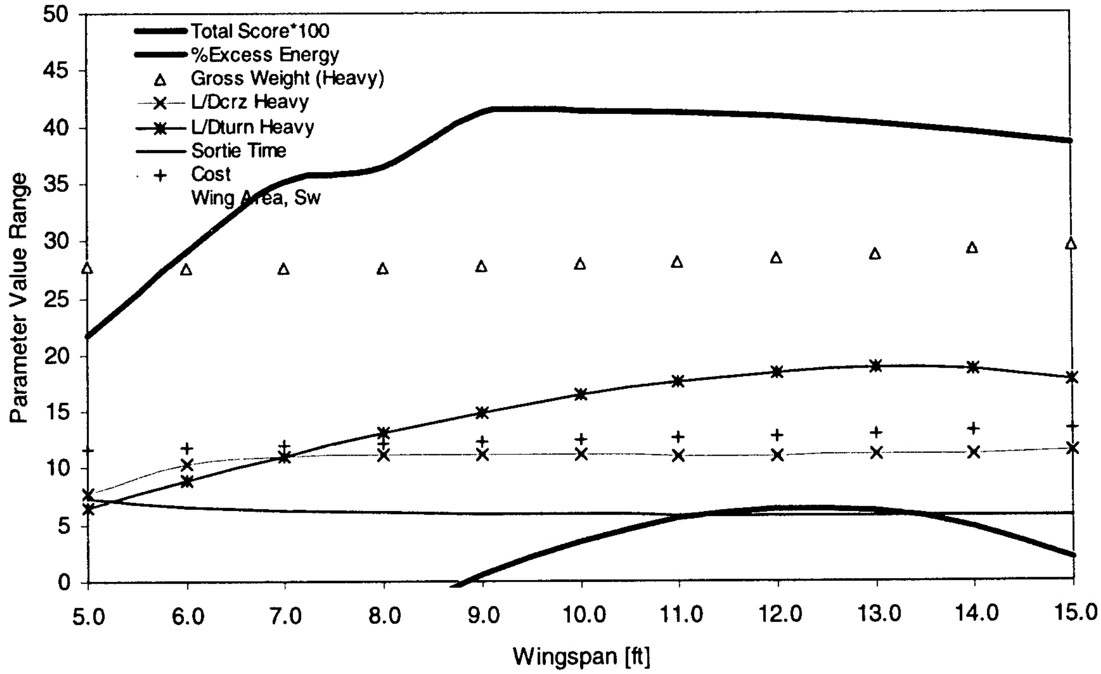


Figure 4.2 Wingspan trade study showing optimum $b_w = 9$ ft.

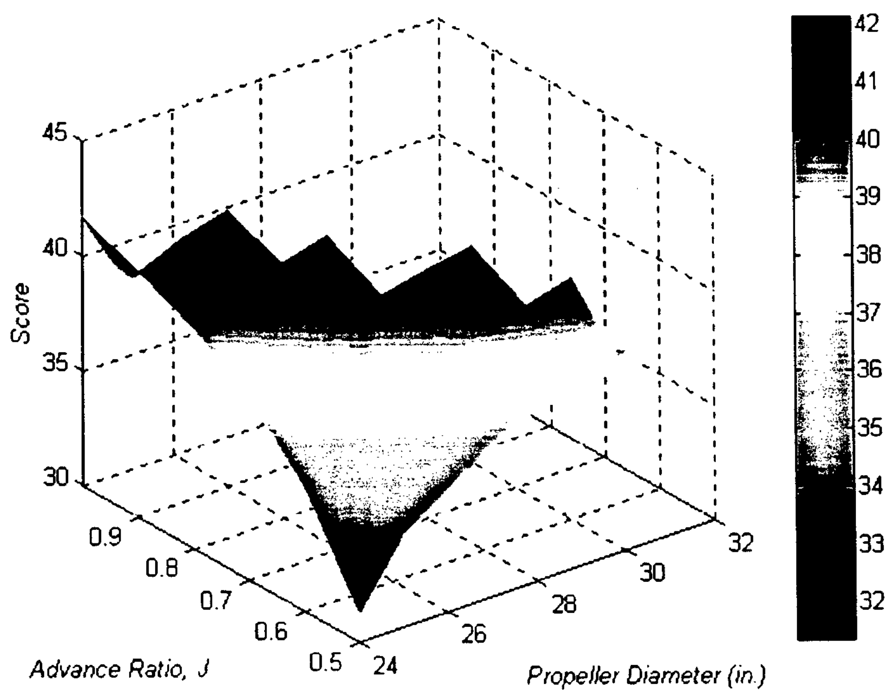


Figure 4.3 Surface plot depicting a maximum score possible with a 24" diameter propeller at $J = 1$, 2.7:1 gear box ratio, and Cobalt 60 motor.

Thrust curves were generated after the results from the motor, propeller, and gear box matching trade studies were presented. These trade studies provided insight concerning the affect of velocity and variable throttle settings on thrust (Figure 4.4), total propulsive efficiency (Figure 4.5), and energy consumption (Figure 4.6). Thrust declined at higher velocities because the ratio between propeller RPM and incident freestream velocity increased, meaning that the propeller imparted less momentum into the incoming flow at a given RPM. The total propulsive efficiency curves (Figure 4.5) demonstrated that at 100% throttle the highest efficiency was permitted at speeds near 60 mph. Matching cruise velocity to this value would ensure optimum usage of battery energy and lower the sortie time. Plots of energy consumption versus velocity further indicated lower energy consumption for these higher velocities.

In addition to the trade studies discussed above, a heat transfer model in "Mission" was implemented to see how the propulsion system dissipated energy at cruise velocities near 60 mph. The internal resistance of the motor, batteries, and speed controller were a function of temperature. Figure 4.7 shows that by the end of a mission, the motor remained under its recommended temperature limit. Similar results were obtained for the batteries and controller. To accomplish this, the heat transfer model required a heat sink and forced convection by a flow ducting mechanism within the fuselage. This study was deferred for the Detailed Design Phase.

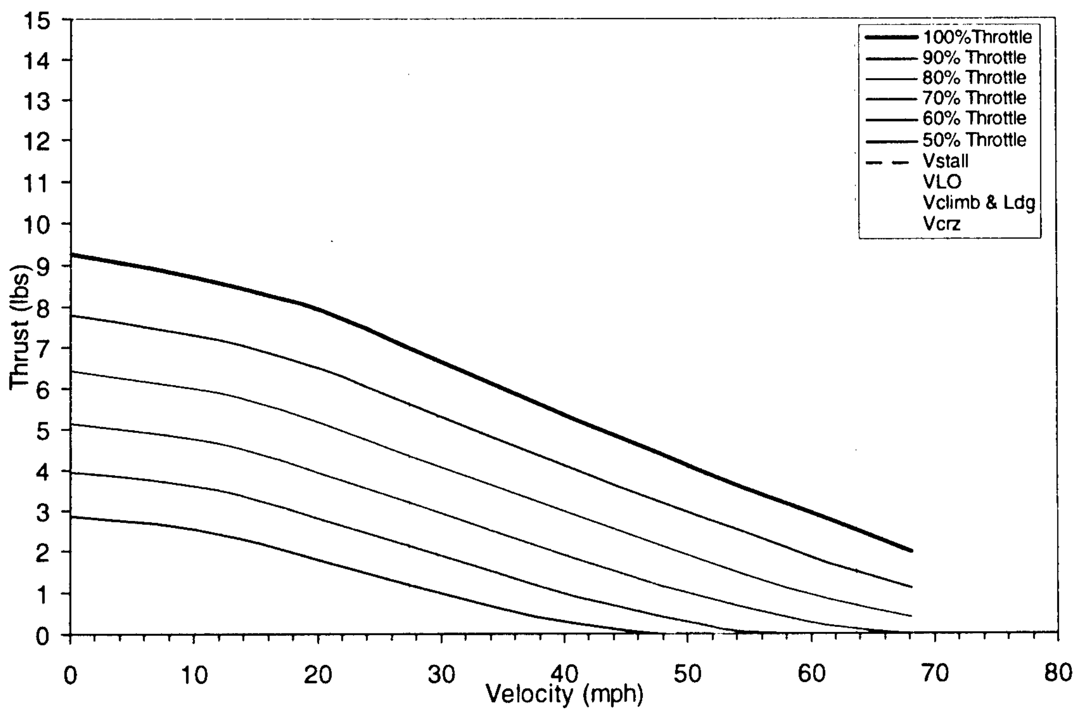


Figure 4.4 Affect of variable throttle and cruise velocity on thrust.

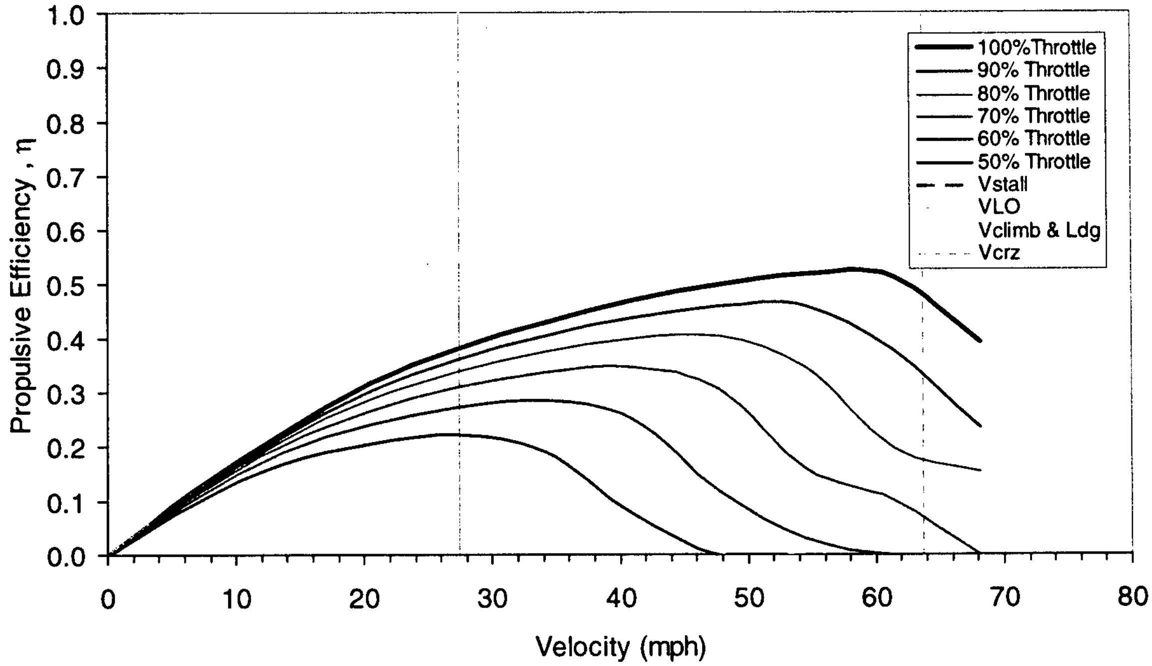


Figure 4.5 Propulsive efficiency plotted as a function of cruise velocity and variable throttle settings.

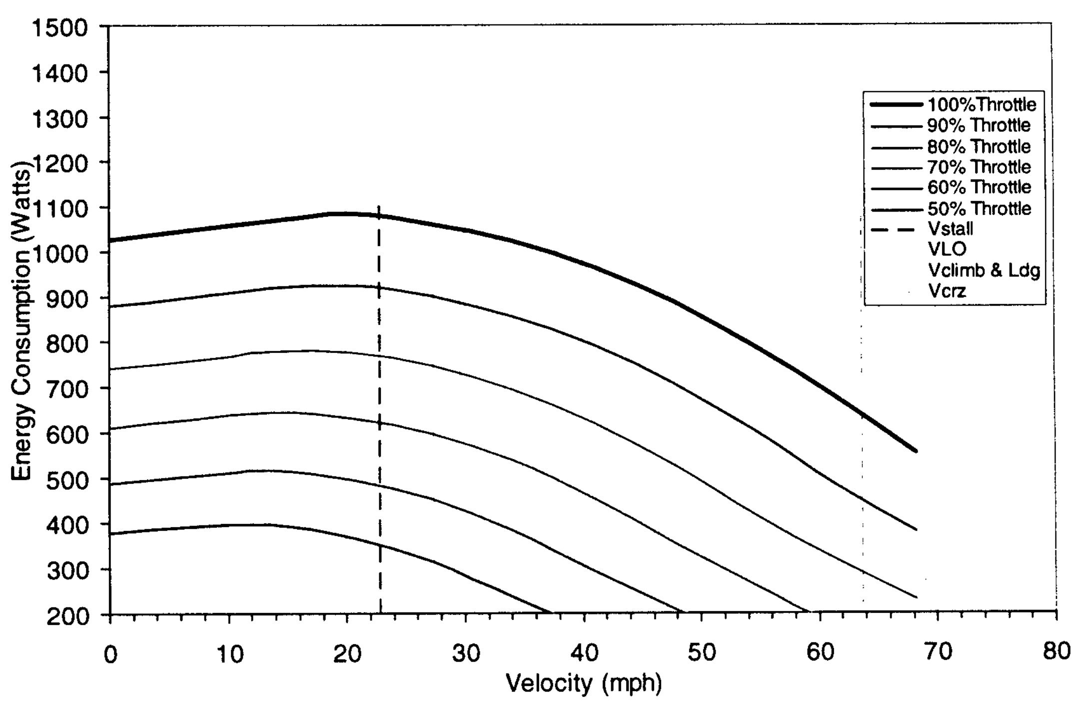


Figure 4.6 Energy consumption at various throttle settings and cruise velocities.

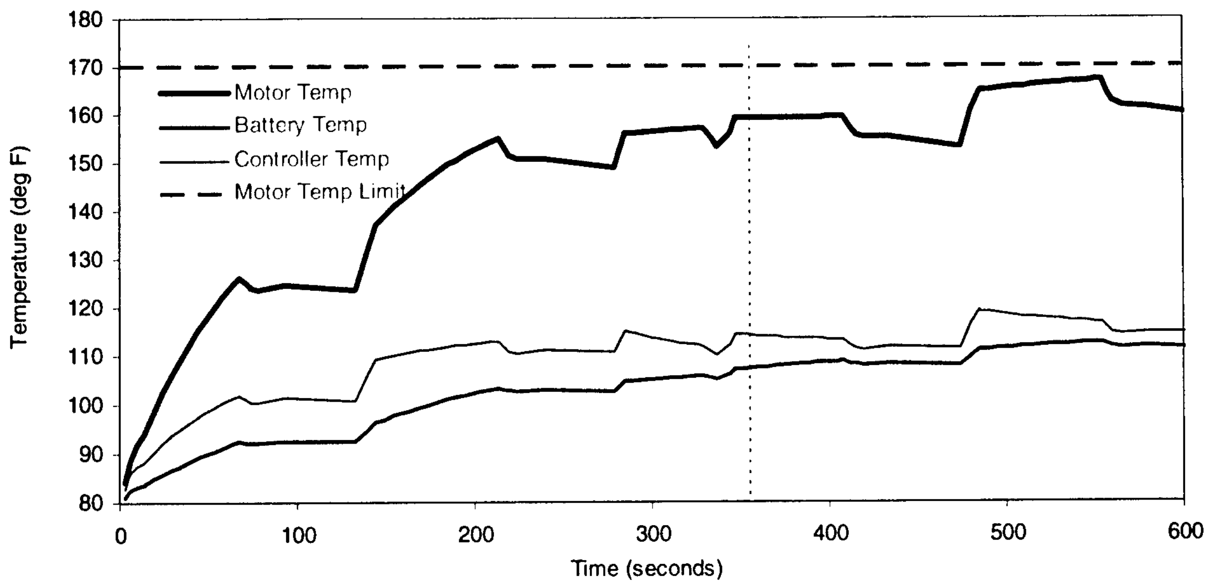


Figure 4.7. Propulsion system component predicted temperatures during mission operation.

Design Parameters & Sizing Trade Studies Focused on *Structural Robustness*

Wing Structure: A spar was the main element that distributed the load from the fuselage throughout the wing. Different spar designs were considered, such as a tube spar, I-beam spar, and spar caps with a shear resisting D-section on the forward section of the airfoil rather than a solid I-beam web. A subroutine in the "Mission" simulation incorporated a cantilevered beam model for a wing. Different materials for the spar stucture and wing core were studied for each spar configuration. An elliptical lift distribution was modeled and the center of pressure felt by discrete lengths along the wing was used to generate shear and moment diagrams versus wing span.

Maximum assumed loading for the spars were 5.25g, with an added safety factor of 1.5. Under this condition, no skin rupture, shear failure, core crushing, or skin buckling was allowed. Calculations for ultimate stresses, shear stresses, core crushing, wing bending radius, and face wrinkling were conducted in the structure model to size appropriate spar cap width and thickness, as well as the spar webbing width.

The tube spar posed several disadvantages, particularly a lack of shear web that caused a predicted failure in shear and possible core crushing. Torsional stiffness was excellent in this design; however integration with a foam core was difficult due to alignment. Airfoil shape preservation and the required area moment of inertia was difficult to achieve without a thick tube. An I-beam spar was possible for satisfying every design criteria mentioned above. Load concentrations occurred, though, at the cap and web junction. To make use of the vast wing area and wing skins, a derivative of the I-beam was

analyzed that did not have the concentrated loads at the cap and web junction. This design used spar caps mated with the wing skin section that was forward of 25% MAC, thus creating a D-section. The wing skin in this region needed greater thickness than the aft wing section to act as the shear web. In addition, the D-section was filled with foam core for rigidity and provided torsional stiffness.

Carbon fiber spar caps and fiberglass wing skins were the optimal solution for the wing structure. Each material provided high strength-to-weight ratios in comparison to materials such as balsa wood, spruce, and aluminum. The carbon-fiber caps were tapered in thickness and width from the root chord to the wingtips, where loading conditions were negligible. Figures 4.8 and 4.9 displayed the distribution of stresses and shear loads in the spar caps and fiber-glass D-section, respectively. The material's ultimate stress limits are well above the design loading conditions; face wrinkling was not a problem as well because the material would fail before experiencing any face wrinkling. Discontinuities in spar cap stress occur due to the changing spar cap width at different distances along the wing semispan. The shear discontinuity at the side of body was due to the attachment load concentration in that area, which was offset from the wing centerline.

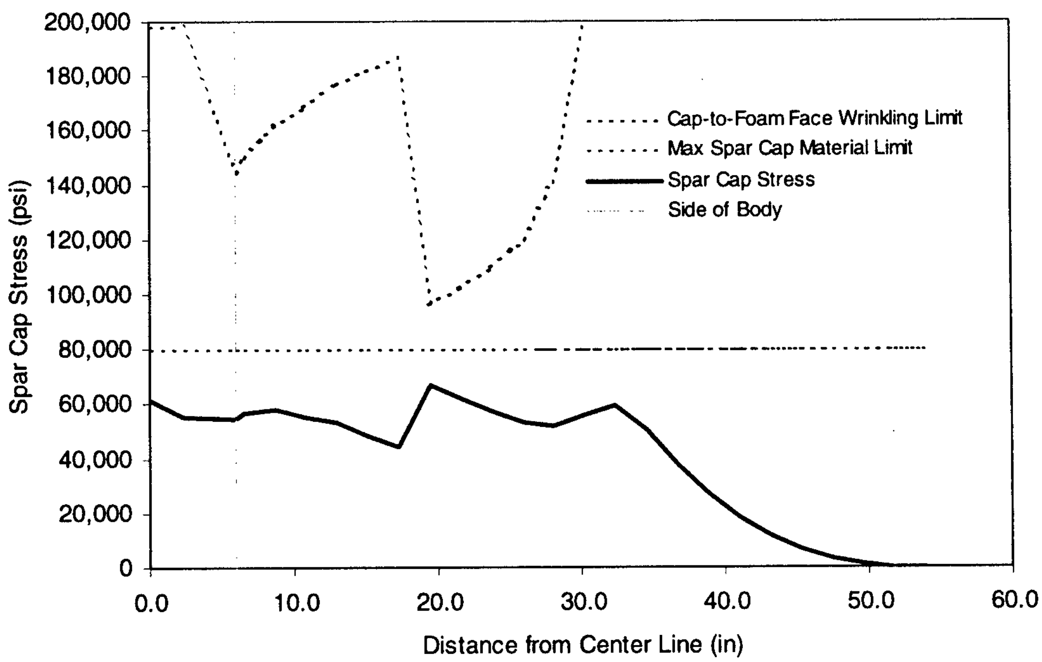


Figure 4.8. Spar cap stress and face wrinkling trade study results.

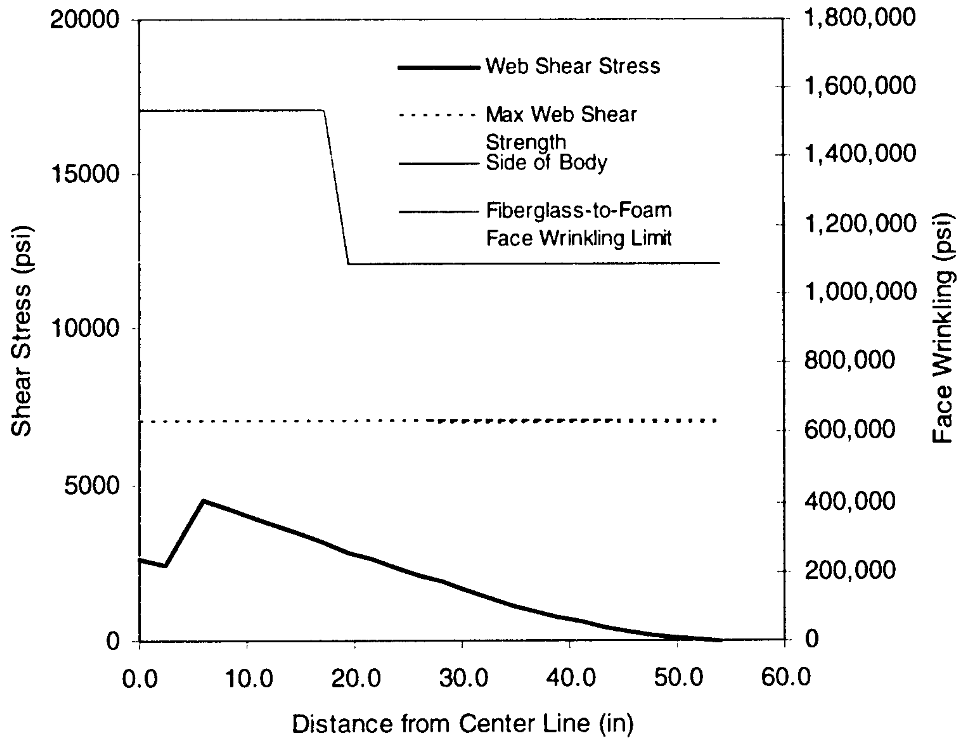


Figure 4.9. D-section shear and face wrinkling behavior.

Fuselage Structure: A monocoque fuselage was compared with a fuselage containing various bulkheads joined together by a structural skin. The monocoque structure eliminated the need for bulkheads and provided torsional stiffness; the bulkhead structure fuselage was advantageous because only the necessary structure was located near regions of concentrated loads. A derivative of both of these ideas was a fuselage that contained a single longitudinal structural member and a non-structural aerodynamic shell. This concept reduced the weight of the skin and minimized the necessary skin-bulkhead junctions. A light weight board to be used as the fuselage floor was considered for this case and trade studies were conducted to verify the location of hard point structure and appropriate thickness.

In addition to sizing the fuselage board, the attachment points for large structural members, including the wing, landing gear, empennage, and motor mounts, were sized to fail under certain loading conditions prior to the failure of each component. The landing gear attachment bolts were sized for a 10 g-load in the drag direction; wing bolts were sized for a 10 g-load in drag, as well as a 3 g-load from a wingtip strike. The empennage and tail-wheel assembly was designed for a 4 g-load in the vertical direction, and a 3 g-load in side force.

The stress analysis trade study indicated that the fuselage board could consist of a honeycomb core covered by carbon-fiber skins. The skins would provide ample shear and bending stress rigidity,

whereas the honeycomb core provided strength against core crushing. The attachment hardpoints for the wing, landing gear, empennage, and motor mount contained balsa wood inserts in the honeycomb core.

Landing Gear: The main gear was sized for a 4 g-load vertical load case. This scenario reflected a hard landing where sufficient energy dissipation was required by structural flexing. A carbon-fiber landing gear provided stiffness against bending stress and face wrinkling. The required dissipation energy was calculated by incorporating a 4 ft/s rate of sink at a 5 deg. descent angle. Propeller clearance was also an issue; therefore, the landing gear height was designed to provide 1.5" of clearance at full deflection. Furthermore, the landing gear width (wheel to wheel) was established with a 12° angle of roll clearance for protecting against wingtip strike.

5. Detailed Design Phase

The final design phase incorporated the engineering parameters obtained in trade studies during the Preliminary Design Phase. New assumptions were implemented that focused on particular components in various subsystems, rather than the overall configuration. The component selection process and system integration were evaluated to allow uniform functionality among the various aircraft subsystems without interference. This permitted final mission performance estimates using the "MISSION" simulation software. True performance data was also obtained through various flight tests, structural tests, and propulsion tests that provided feedback in "MISSION" for optimizing the final aircraft once it was built.

5.1 New Assumptions for Specific Subsystems

The following new assumptions were necessary to simulate the contest mission with realistic flight conditions:

1. *25 fps Winds*: The contest location in Wichita, Kansas may subject the aircraft to winds with this velocity magnitude due to climate conditions in that region during the spring.
2. *1320 ft Altitude at 60 °F*: Air density and temperature at this altitude were obtained from Standard Atmosphere Tables and average weather conditions in Wichita, Kansas.
3. *0.65 Battery Energy Factor*: Ample discharge studies with battery packs determined that only 65% of their rated capacity was available due to rapid discharge rates.
4. *40 sec. Average Payload Swap Time*: Payload exchange time was estimated based on average payload swap times recorded by USC teams in previous DBF competitions.
5. *100% Throttle Setting*: Full throttle setting was enabled to obtain the highest cruise velocity and operate at the highest propulsive efficiency.
6. *No Energy Penalty in Turns*: The cosine losses from incident airflow on the propeller cause the motor to continue with constant torque while moving fewer slugs of air, thereby increase current draw and decreasing the available energy.

5.2 Component Selection & Systems Architecture

The propulsion, control, cooling, brake, and payload restraint systems were arranged in the aircraft to maintain the CG at the desired location and still provide easy access. Each subsystem was mounted to the fuselage board. The components were summarized as follows:

Propulsion System: The Astroflight Cobalt 60 and PC-2400 NiCd batteries were chosen for propulsion and power. This battery type was rated at the lowest internal resistance and highest specific energy out of the extensive battery database in the "MISSION" simulation. The battery pack was designed for 30 cells to provide sufficient energy to complete a single mission sortie. The total battery pack weight was well under the 5 lb. limit required by the contest rules, thereby allowing for extra cells to be used if the mission sortie cannot be completed.

Control: Flaperons were incorporated on the wings instead of separate flaps and ailerons. This reduced the servo count and enabled a full span increase in C_L when deflected. Slower landing speed was possible in this configuration. The elevator and flaperons were geared inversely to have a net elliptical lift distribution during cruise for low trim drag. The vertical stabilizer was sized as a full-flying rudder to allow ample control authority for the 25 fps winds in Wichita, Kansas. Otherwise only 12 fps crosswinds were manageable. In order to avoid flutter, the vertical tail's center of pressure was located forward of the rotation point so that the control servo could provide yaw stability. In this configuration, divergence was possible; yet sufficiently large servos were selected based on hinge moment calculations for the predicted flight velocity. The divergence limit was not considered an issue because the average dynamic pressure in the mission flight envelope was substantially lower than that needed for divergence. A single servo was utilized to control both the vertical tail and the tail wheel to reduce the servo count. In crosswind scenarios during takeoff and landing, the coupled steering ability with rudder deflection would provide the pilot with sufficient yaw control.

All control surfaces were designed using a skin-hinge; i.e. the fiberglass skin on the wing and empennage served as the hinge and provided the point of rotation. This ensured that the airfoil skin was continuous at the control surface rotation point. The discontinuity on the opposite side of the hinged surface was covered with thin mylar to provide a smooth surface.

The "MISSION" simulation software was used for stability and control exercises. By iterating the wing position and tail sizes, the design stability parameters specified in Table 5.1 were obtained.

Table 5.1 Stability Parameters calculated for cruise conditions.

Stability Characteristics		
Pitch Stability, Static Margin	%mac	34.8%
Pitch Frequency (@ Cruise)	Hz	3.28
Pitch Damping Ratio (@ Cruise)	n.d.	88%
Directional Stability, CnBeta	1/deg	0.00216
Yaw Frequency (@ Cruise)	Hz	1.63
Recommended Aileron-Rudder-Interconnect Gain	n.d.	0.32
Instantaneous Roll/Yaw in a side gust	n.d.	0.39
Unrecoverable Spin Factor (%fin blanking by Htail)	%	11%
Adverse Yaw due to 15deg aileron (@climb CL)	deg	6.4
$C_{M,Cl,cg}$	n.d.	-0.3482
$C_{M,\alpha}$	n.d.	-0.0324
Lateral Stability	1/deg	0.0004

The static margin was substantial for any variation in CG and pitch stability was guaranteed with a negative moment coefficient for all variations of C_L and α. This was necessary to accommodate different CG locations for sorties with and without payload. In addition pilot trim inputs were required for both flight conditions. Pitch frequency was sufficient to avoid sluggish handling and the 88% pitch damping ratio ensured a lower overshoot and faster settling time after the pilot provide elevator inputs.

Lateral stability was indicated by a positive directional stability term. Yaw frequency was sufficient to avoid sluggish handling as well. Due to the possibility of strong cross-winds in Wichita, Kansas, a 39% gain in aileron deflection per degree of rudder deflection was necessary to negate any adverse yaw, especially since approximately 6° of adverse yaw was possible with any flaperon deflections. This lateral stability behavior was important when initiating turns towards landing approach, since strong cross winds could exacerbate the adverse yaw condition and cause unwanted sideslip or roll due to corrective pilot inputs.

Cooling System: The "MISSION" heat transfer simulation indicated that 40 in.^{2} aluminum heat sink would conduct sufficient heat away from the motor casing to maintain its temperature within specifications. The steady-state model also looked at the effects of forced convection and converged on a solution for inlet and exhaust duct areas. A 9 in.^{2} inlet duct was placed in the fuselage nose and exposed the motor heat sink to incoming freestream air. An internal foam flow diverter did not allow the flow to diffuse, but rather accelerated it around the battery packs and towards exhausts placed laterally on both sides of the fuselage. A third exhaust on the topside of the fuselage allowed the air heated by the motor heat sink to exit without heating the battery packs. The total exhaust area required was 16 in.^{2}.

Payload Restraint & Brake System: The 24 softball payload needed a simple restraining solution. A foam cradle was designed to carry the softballs in a 3x8 configuration. This configuration resulted from a compromise between reducing the hidden wing area and minimizing the overall fuselage length.

The pneumatic braking system contained an air bottle and a proportional release valve that was servo controlled. The amount of pressure released to the brakes was determined by conducting rolling ground tests. Since the main landing gear was placed forward of the CG, the CG moment arm was calculated to avoid tipover and propeller strike when brakes were applied. However, only 80% of the brake valve was opened in order to avoid a sudden negative pitching moment during taxi.

Estimated Mission Performance

Using "Mission", the team was able to estimate the mission performance for the aircraft. The takeoff field length was estimated to be 55 feet for the first empty lap, 150 feet for the heavy lap, and 58 feet for the second empty lap. "Mission" predicted that the plane will be able to complete all six laps, and have an estimated mission time of 5.93 minutes, which includes 40 seconds for payload swap and 15 seconds for taxiing..

Table 5.2 shows the different velocities, propulsion, and aero data at certain periods during the mission, for both light and heavy laps. The program predicted an average cruise velocity of 63mph throughout the entire mission. The table also the estimated values for the aircraft's propulsion system. Aero data, at the bottom of the table, shows different C_L during parts of the flight, as well as lift to drag ratios.

Table 5.2. Estimated Mission Performance Table

Velocity Breakdown	**Light**	**Heavy**	
V_{Climb}	30	37	mph
V_{LO}	27	34	mph
V_{Crz}	64	61	mph
V_{Ldg}	30	37	mph
V_{stall}	23	28	mph
Propulsion			
Available Energy	143,382	ft-lbs	
Max Current	35.15	amps	
Motor Temp at end of mission	160	deg F	
Battery	108	deg F	
Speed Controller	114	deg F	
Aero	**Lap 1 (Light)**	**Lap 2 (Heavy)**	**Lap 3 (Light)**
Unpowered C_{Lmax}	1.93	1.93	1.93
Powered C_{Lmax}	2.12	2.12	2.12
$C_{LStraight}$	0.27	0.47	0.26
$C_{L\ Turn}$	1.35	1.35	1.35
L/D Straight	7.29	11.22	7.11
L/D Turn	14.9	14.9	14.9
Turn Bank Angle	78.4	69.9	78.8

The rate of climb and sink for both light laps and heavy laps are shown in Figures 5.3 and 5.4 respectively. For light flights, the rate of climb was estimated to be just above 950 fpm, with a rate of sink slightly above -150 fpm. For heavy flights, the predicted rate of climb was about 580 fpm, while the rate of sink was estimated to be around -200 fpm. "Mission" estimated a flight score of 4.90. The spreadsheet also predicted that the plane would only be able to complete one sortie, due to limitations in energy. The excess energy "Mission" predicted was 1%.

5.4 Performance Data

Takeoff Performance: More energy is needed for our plane. Observed from our test flights, we need additional excess energy to supply enough power to complete its mission. From take off, our plane without payload travels approximately 70 feet before lifting off the ground. With payload, the distance required is about 190 feet. Additional excess energy to our power system is needed to be able to complete additional laps in the mission.

Bolt Testing: Structural tests were conducted on the bolts to determine their performance and load capacity. The bolts selected for use were nylon bolts ranging from 1/4" to 3/8" in diameter. The test on the nylon bolts determined its ultimate shear and ultimate tensile strength. The ultimate shear load of the bolts was found to be approximately 600 pounds, while the ultimate tensile load was about 1000 pounds. Our design intent was to break the bolts at 10 g loadings in the drag direction to prevent major damage to the plane body. From observations in test flights, the bolts must have ¼" diameter to rupture at the desired loading. The ideal bolt diameter selected for the landing gear was 5/16" and 1/4" for the wing to achieve the design condition.

Flight Test Performance Summary: For a successful flight performance, one of the key components for the plane is cooling the motor. Based upon the temperatures recorded after a series of flight tests, the cooling performance was excellent. The motor temperature measured 110° F; the battery pack ranged from 90° -100° F; and the controller reached 80° F. The observed temperatures were all lower than the listed manufacturer limits.

Battery conditioning proved to be a key component of the plane's endurance. The batteries performed best when charged slowly. Based upon the test flights, all the available energy within the batteries is necessary to perform the entire mission. After testing, it was observed that the optimum performance could only be obtained with a slow charging rate. Efforts to increase the battery energy further are presently being looked at.

The observed cruise speed average was measured to be within 10% of the predicted value. This was done by measuring the distance our plane traveled along the straightaway, divided by the time needed to travel.

Ground handling of the plane performed sufficiently well. Braking and steering were non-issues as we were able to control the plane with ease. During flight tests in heavy crosswinds, the plane showed sufficient control authority as it was able to correct and compensate for the crosswind.

Pitch and lateral stability of the plane performed adequately during the test flights. However, the pilot required new trim inputs when flying laps with payload due to the aft CG shift. Power On/Off stalls were also conducted and the aircraft did not diverge from pilot control; the pilot had sufficient recovery time to correct for stall. A speed brake was tested by fully deflecting the flaperons on landing approach. The aircraft approach speed decreased substantially as a result and decreased landing overshoot.

5.5 Weight, Balance and Rated Aircraft Cost Worksheets

Weight Budget: The weight budget is summarized in Table 5.3, showing each subsystem and the individual components it contains. Different weighting was provided with the K_{fudge} factor among the subsystems. This parameter was an estimate used to provide weight estimates for subsystems that were historically overweight or underweight in USC's DBF competition aircraft. The relative weight of each component with respect to the total heavy gross weight was also shown. The dominating components included the motor, batteries, and the fuselage.

Balance Distribution: Table 5.4 lists the relative longitudinal separation from a reference location behind the motor and battery compartment. This origin was used to determine the aircraft CG with respect to the wing MAC, as well as the differential loading between the main gear and tail wheel. The wing leading edge position was found with respect to the origin in the XLE_{MAC} parameter and both the empty CG and heavy CG locations were also shown. The payload moved the CG aft about 22%, which was acceptable since the pitch stability static margin was on the order of 35%.

Rated Aircraft Cost: The final cost breakdown was summarized in Table 5.5. Cost was generally attributed to the propulsion system, which contained a multiplier that was 15 times larger than any other cost multiple. Thus the single motor and 30 cell battery pack were the most influential aircraft components. As seen in the Conceptual and Preliminary Design Phases, the wing and fuselage geometry were the second most influential parameters in the cost determination. The final rated aircraft cost was calculated to be 12.26.

Table 5.3. Weight Budget Worksheet

System	KFudge	Sub-Component	Weight Breakdown	% of Heavy Weight
PROPULSION	1.0	Sub-Total	**7.186**	**25.9%**
		1 Motor	1.563	5.63%
		1 Motor Mount / Heatsink	0.115	
		Battery wt incl. solder & jack	4.156	14.97%
		All Wiring	0.100	0.36%
		1 Speed Controller	0.150	0.54%
		1 Propeller	0.403	1.45%
		1 Spinner & Prop Nut	0.230	0.83%
		1 Motor Mount	0.46875	1.69%
WING	1.0	Sub-Total	**1.984**	**7.1%**
		Wing Spar	0.453	1.63%
		Wing Core	0.564	2.03%
		Wing Skin	0.967	3.48%
TAIL & Winglets	1.5	Sub-Total	**0.626**	**2.3%**
		HTail Skin	0.277	1.00%
		HTail Core	0.075	0.27%
		VTail Skin	0.201	0.72%
		VTail Core	0.072	0.26%
		Winglets	0.000	0.00%
RADIO	1.0	Sub-Total	**1.563**	**5.6%**
		Receiver	0.125	0.45%
		Servos	0.938	3.38%
		Battery Pack	0.500	1.80%
LANDING GEAR	1.0	Sub-Total	**2.229**	**8.0%**
		Main Gear Struts & Bolts	1.408	5.07%
		Main Wheels	0.541	1.95%
		MG Axle Hardware	0.100	0.36%
		Nose Wheel or Tail Wheel	0.030	0.11%
		Nose Gear Strut & Mount	0.000	0.00%
		Brakes, Tubing, Air Tank	0.149	0.54%
FUSELAGE	0.85	Sub-Total	**4.244**	**15.3%**
		Fuselage Skin	1.747	6.29%
		Bulkheads	2.496	8.99%

Airframe Weight =	**8.93**
Payload Weight =	**9.94**
Heavy Gross Weight =	**27.77**
Light Gross Weight =	**17.83**

Table 5.4. Balance Distribution Worksheet

Parts	C.G (midpoint) distance from Motor Blkhd (in)	Weight (lb)	Moment (in-lb)
Propeller	**-7.92**	0.40	-3.19
Servo for Nose Gear	60.72	0.00	0.00
Nose Wheel	60.72	0.03	1.85
Nosegear	60.72	0.00	0.00
Main Wheels	5.50	0.54	2.98
Horseshoe Strut (maingear)	5.50	1.41	7.74
Axle Hardware	5.50	0.10	0.55
Battery Pack	**1.08**	4.16	4.49
Radio Receiver	**38.12**	0.13	4.77
Receiver Battery	38.12	0.50	19.06
Motor + Mount	**-4.32**	2.03	-8.78
Brakes, Tubing, Bottle+Servo	**1.08**	0.34	0.36
Fuselage (40% length)	28.07	4.24	119.13
Speed Controller (Batt Pack +2"))	1.25	0.15	0.19
Wiring (@ controller)	38.12	0.10	3.81
Spinner & Prop Nut	-7.92	0.23	-1.82
Horizontal Tail+Servo	65.05	0.54	35.12
Vertical Tail +Servo	56.85	0.46	26.21
Wing (cg@mac/4)	16.05	1.98	31.85
2 Wing Servos (cg@mac/2)	18.35	0.38	6.88
Heavy Payload (softballs)	**19.92**	9.94	197.93

Total Weight (lbs) =	27.65
Total Moment (in-lbs) =	449.12
Actual X C.G. (total) =	**16.24**

Empty Weight (lbs) =	17.72
Payload MAX Weight (lbs)=	9.94
Gross Weight (no tail pad (lbs)) =	27.65

MAC (in) from Stable Sheet =	9.18
Target C.G.(%mac) from Stable Sheet =	27.00%
% Weight on Nose or Tail wheel =	19.5%
% Weight on Mainwheels =	80.5%
Required XLEmac (in) for desired balance =	**13.76**

Payload Length (in) =	32.4
Heavy Pitch Inertia (slug ft2) =	1.489

Empty C.G. (no payload) =	**14.18**

Table 5.5. Rated Aircraft Cost Worksheet

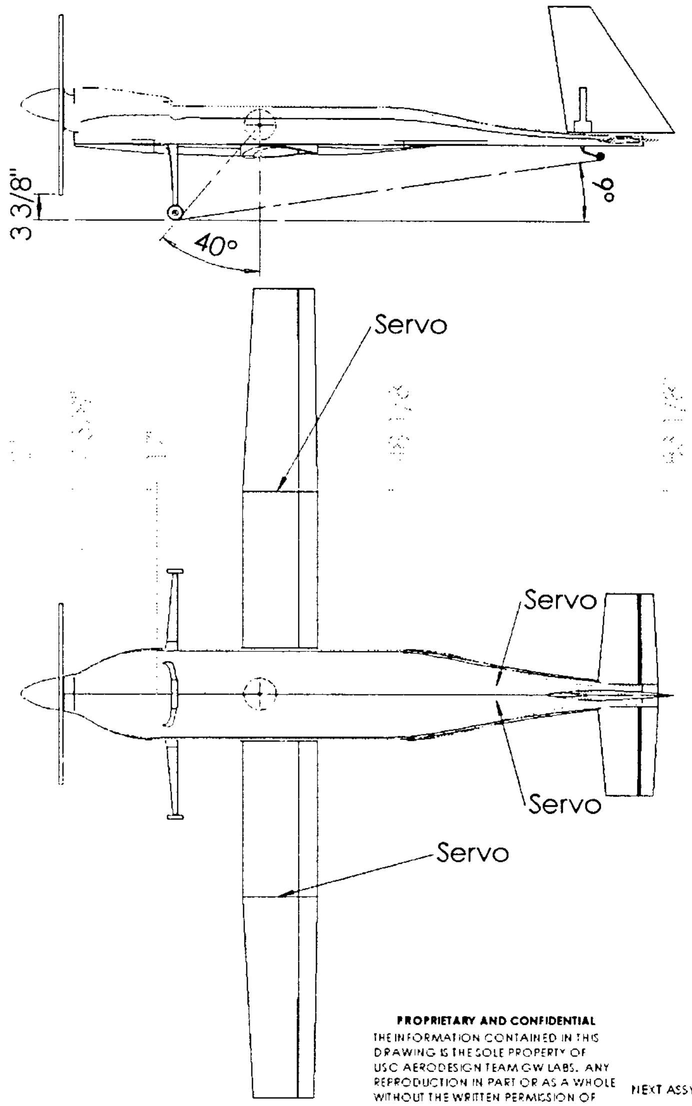
Cost Table						Total Cost	Rated Cost
Rated Aircraft Cost		**Dims**	**Manuf. Hrs**	**Hrs Sub-Total**	**Cost Sub-Total**	$12,261	12.26
MEW (Manufacturers Empty Weight) (lb.)				17.68	1767.65		14.4%
REP (Rated Engine Power)	= (1+.25*(#Motors-1))*BattWt			4.16	6234.3		50.8%
	# of Engines	1					
	Total Battery Weight (lbs)	4.16					
MFHR (Manufacturing Man Hours)				212.94	4258.8		34.7%
WBS1.0 (8hr/ftSpan + 8hr/ftMaxChord + 3 hr/control surface)	# of wings	1	84.3			13.8%	
	Wing Span	9.00					
	Max Wing Chord	0.79					
	# of control surfaces	2					
WBS2.0 (10hr/ft of length)	Fuselage Length (ft)	6.86	68.6			11.2%	
WBS3.0 (10hr/VertWithRudder + 10hr/horizWithElev)	Number of Vertical Surfaces	0	0.0			3.3%	
	Number of Vertical Tails	1	10.0				
	# of Horrizontal Tails	1	10.0				
WBS4.0 (5hr/srevo or controller)	# of Servos and Motor Contollers	6	30.0			4.9%	
WBS5.0 (5hr/Motor+5hr/Prop)	# of Motors	1	10.0			1.6%	

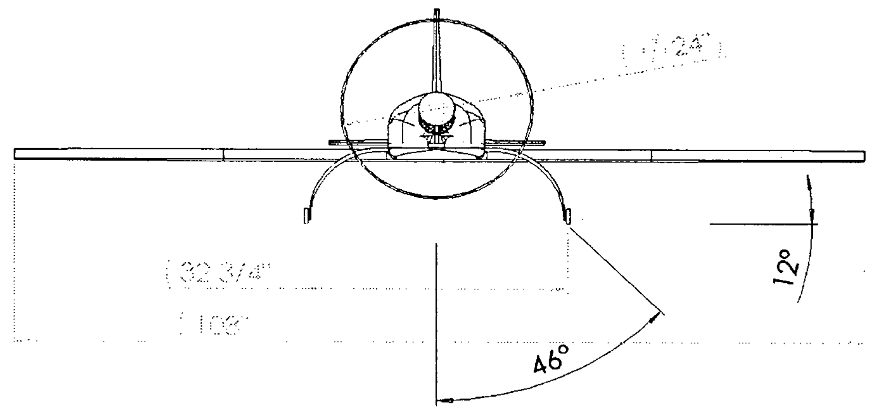
Fixed Parameters	
A (Manuafcturers Empty Weight Mulitiplier) ($/lb.)	100
B (Rated Engine Power Mulitiplier) ($/Watt)	1500
C (Manufacturing Cost Mulitiplier) ($/hour)	20

5.6 Detailed Drawing Package

The following are the summaries of the drawing packages.

- 3 view of airplane
- Isometric view of airplane, with labeled subsystems
- Top view of fuselage, showing all major structural components and attachment locations
- Side view of vertical stabilizer, with detailed specifications on the tail wheel assembly
- Isometric close up view of cooling system in the forward fuselage





Calculated Geometry		Wing	H.Tail	V.Tail	Ventral
Total Area Sw, SH, Sv	ft2	6.85	1.18	1.13	0.00
MAC	ft	0.76	0.51	0.84	0.84
Y MAC	ft	2.17	0.55	0.61	0.00
Span	ft	9.00	2.30	1.42	0.00
Root Chord	ft	0.80	0.57	1.13	1.13
Tip Chord	ft	0.64	0.45	0.45	0.45
Root Incidence	deg	-1.40	-1.00	-	-
Tip Incidence	deg	-2.60	-1.00	-	-
Fuselage Height, Width, Length(incl spinner)	ft	0.65	1.08	6.92	

NEXT ASSY USED ON
APPLICATION

DIMENSIONS ARE IN INCHES
TOLERANCES:
FRACTIONAL ±
ANGULAR: MACH ± BEND ±
TWO PLACE DECIMAL ±
THREE PLACE DECIMAL ±

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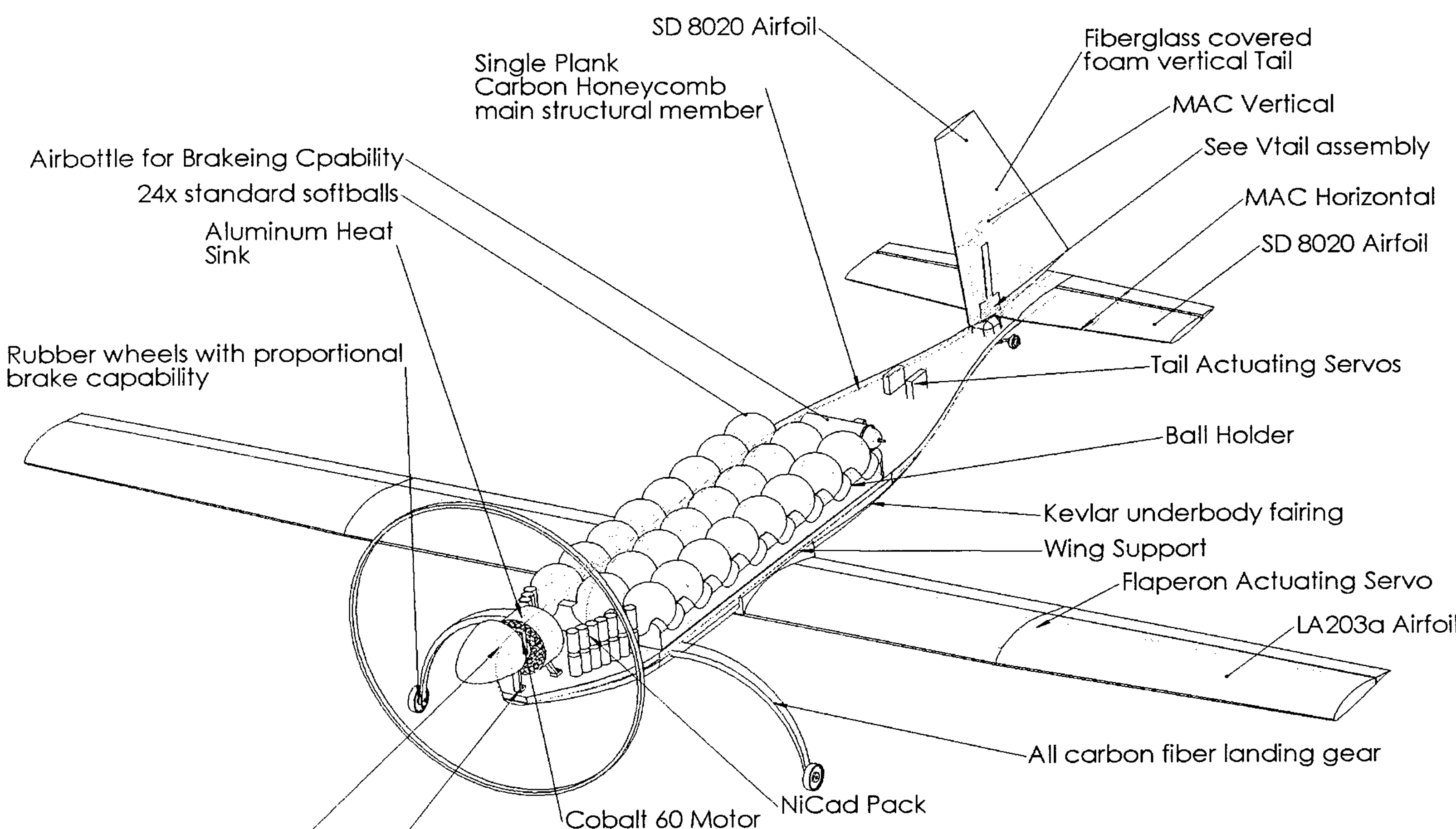

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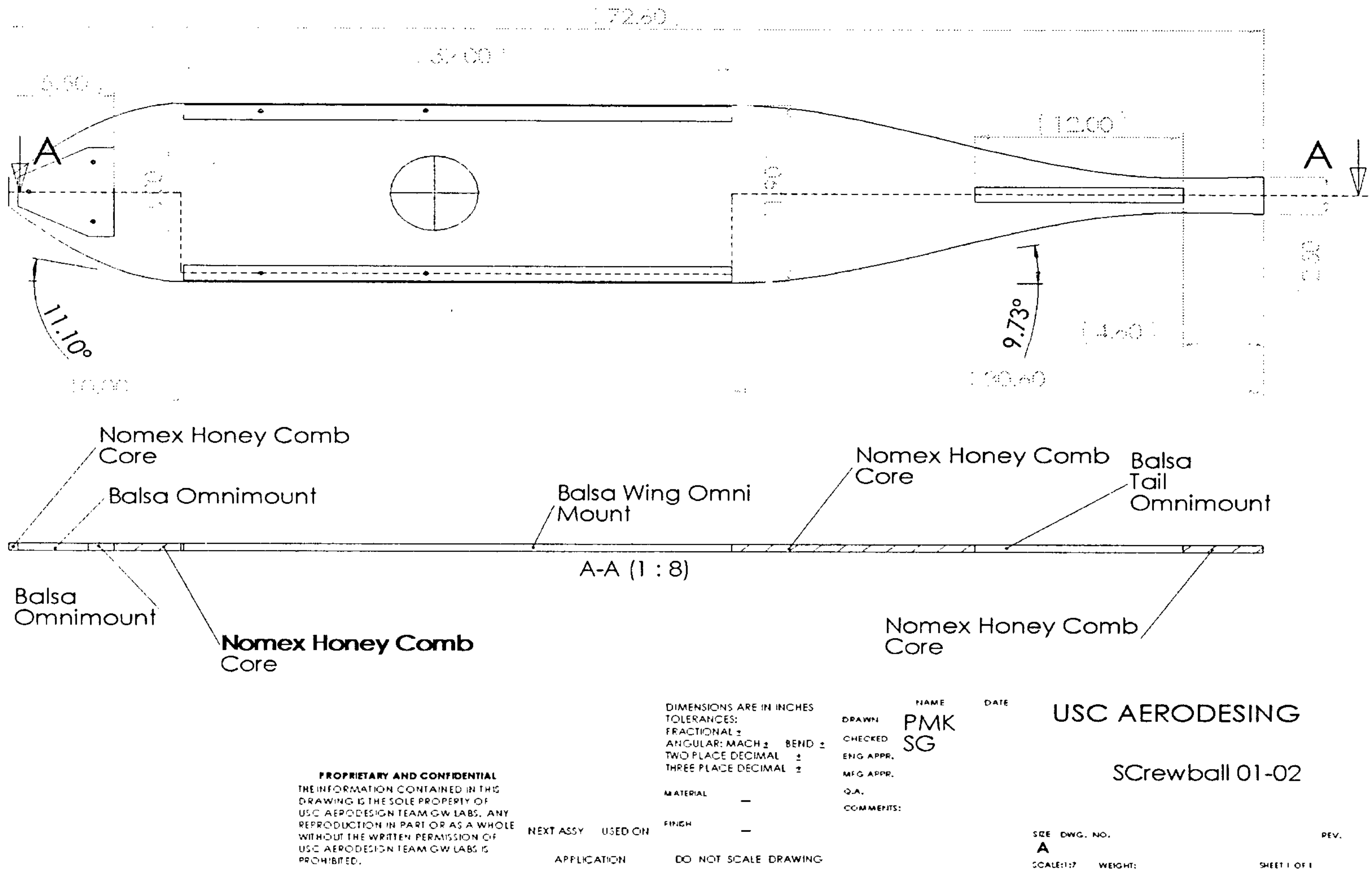

72.60
32.00
12.00
9.73°
11.10°
A
A
Nomex Honey Comb Core
Balsa Omnimount
Balsa Wing Omni Mount
Nomex Honey Comb Core
Balsa Tail Omnimount
A-A (1 : 8)
Balsa Omnimount
Nomex Honey Comb Core
Nomex Honey Comb Core
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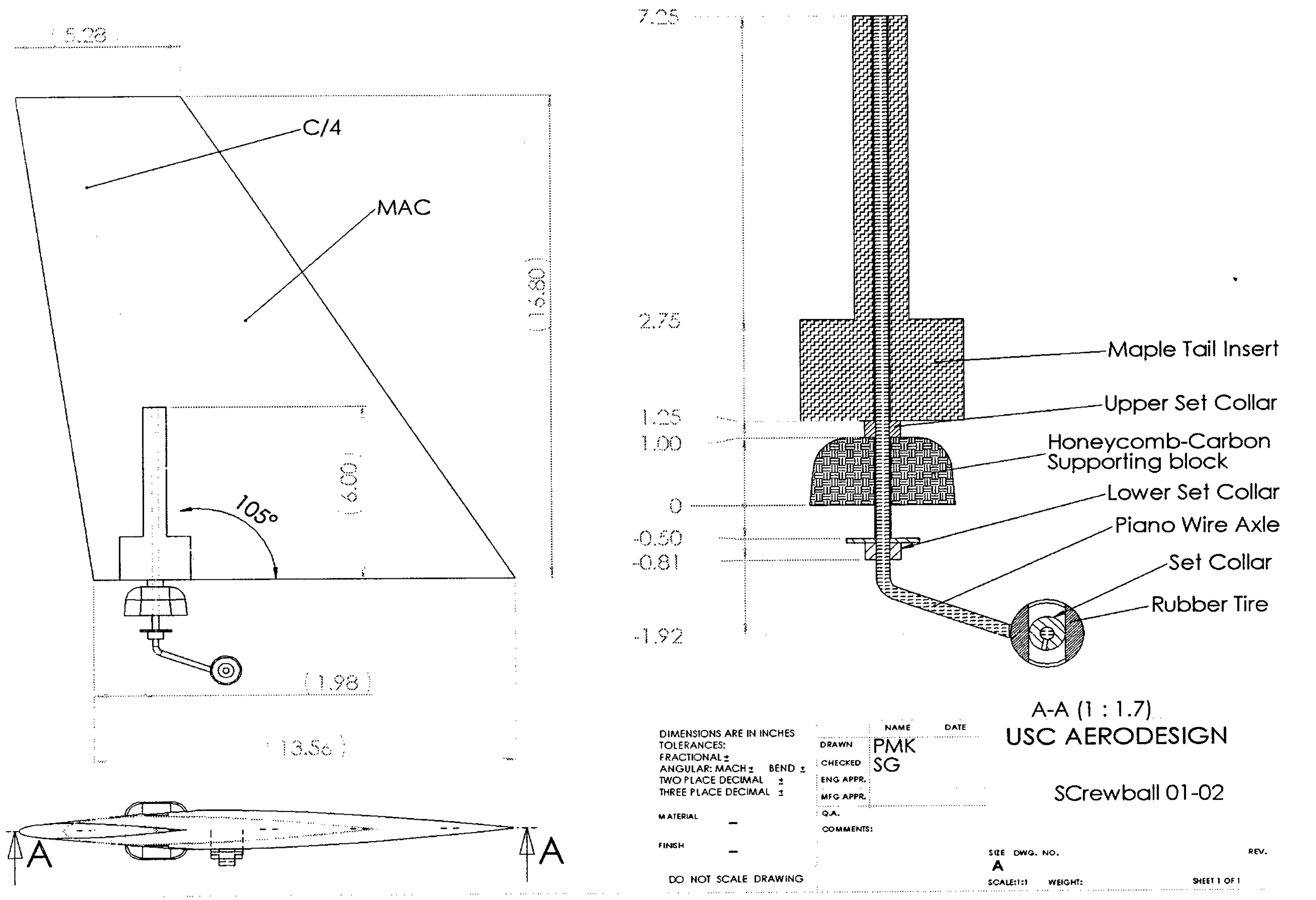

5.28
C/4
MAC
(16.80)
(6.00)
105°
(1.98)
(13.56)
A
A
7.25
2.75
1.25
1.00
0
-0.50
-0.81
-1.92
Maple Tail Insert
Upper Set Collar
Honeycomb-Carbon Supporting block
Lower Set Collar
Piano Wire Axle
Set Collar
Rubber Tire
A-A (1 : 1.7)
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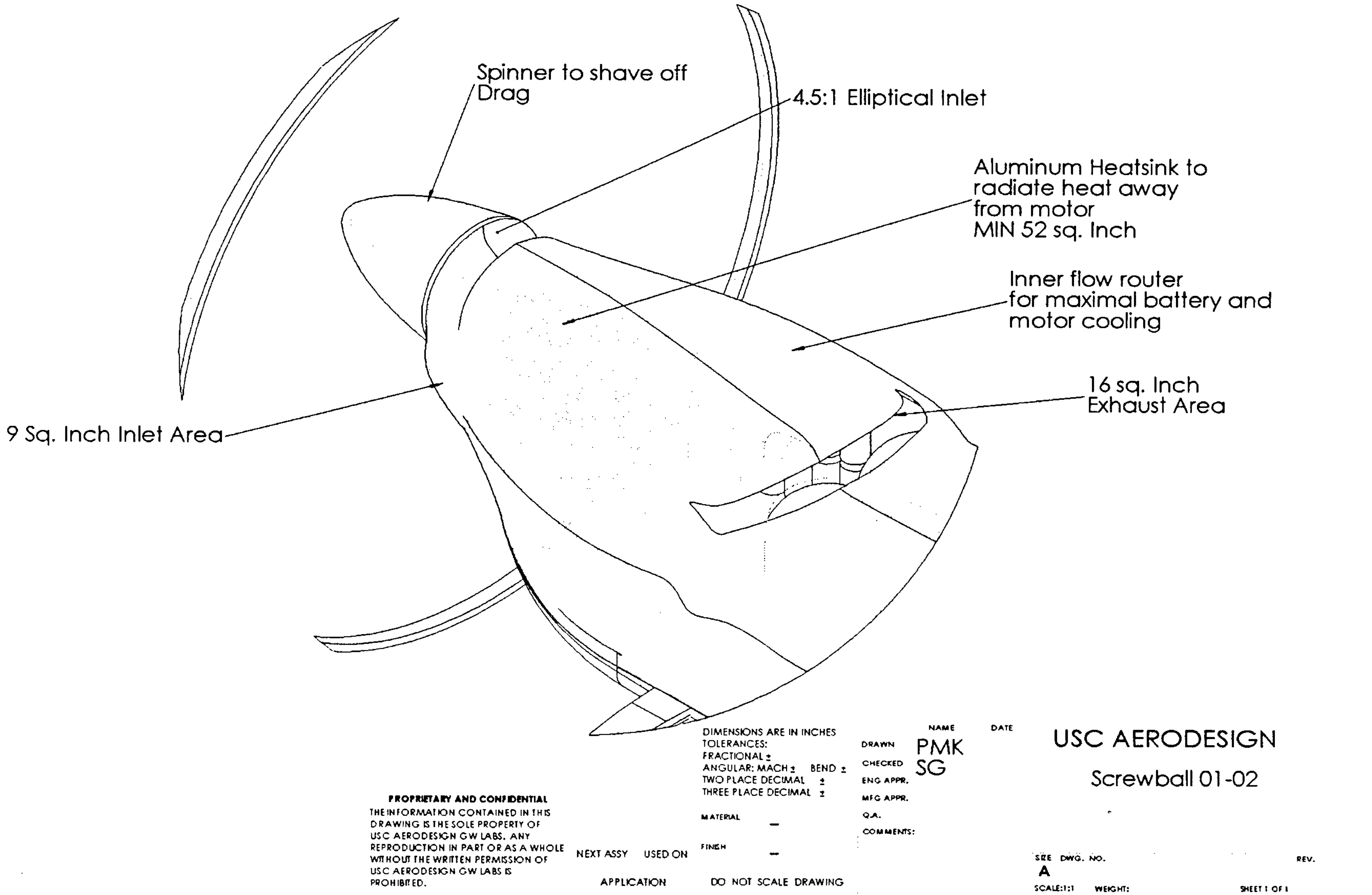
Spinner to shave off Drag
4.5:1 Elliptical Inlet
Aluminum Heatsink to radiate heat away from motor MIN 52 sq. Inch
Inner flow router for maximal battery and motor cooling
16 sq. Inch Exhaust Area
9 Sq. Inch Inlet Area
DIMENSIONS ARE IN INCHES
TOLERANCES:
FRACTIONAL ±
ANGULAR: MACH ± BEND ±
TWO PLACE DECIMAL ±
THREE PLACE DECIMAL ±
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6. Manufacturing Plan

6.1 Figures of Merit

FOM's were developed to help the team select manufacturing processes.

1. *Material Cost and Availability:* Materials that are difficult to obtain could increases construction time. Consideration was given more to materials that were readily available for purchase and relatively inexpensive.
2. *Required Level of Workmanship*: Many methods can be used to create the same component for the aircraft. Some methods, although better, require a high level of experience. Table 6.1 lists the team's skill with certain materials and methods. Although the team opted to use more common methods of construction, the option of using advanced manufacturing methods was frequently discussed. If use of an advanced method for a part was considered necessary, the more experienced members carried out the task.
3. *Repeatability and Crash-worthiness*: The idea of being able to create a part easily and quickly became an important factor for the manufacturing plan. Should a part fail during flight testing due to a crash or other reasons, the ability to make a new one or repair the damaged component was an essential element in choosing a manufacturing process. For this FOM, the key factor was to reduce the amount of downtime between flight tests.

6.2 Analytic Methods Used to Select Final Set of Manufacturing Processes

The amount of time needed to build the part was deemed crucial by the team. New processes cause longer build times, and there was often no guarantee that the part would come out right the first time. Thus, the team decided to use familiar methods of construction for complicated components such as the wing and fuselage. Cutting the amount of time required for building parts also allowed more time for flight testing. Instead of directly estimating build-time, those processes with the least manufacturing steps were favored. This is closely relates to part count as well. If a match existed in the skills matrix for the simplest process, it was selected. If not, the second preferred process was chosen.

6.3 Processes Selected for Major Component Manufacture

For the following components, a summary of the manufacturing processes used for construction, as well as any alternatives investigated, is included.

1. *Motor Mount*: Unidirectional carbon tape wrapped around the balsa/hardwood armature, formed the motor mount. The team also discussed using aluminum exclusively; this method would require a high level of workmanship, and was therefore not used for this sub component.
2. *Fuselage Plank*: The plank, located on the bottom of the fuselage, was constructed from carbon skins over a half inch Nomex honeycomb core. This part is featureless flat-stock except for simple balsa "rails" used for crush strength at hardpoints. The alternative would have been a

molded structural shell with structural bulkheads. The plank obviated the need for structural bulkheads and therefore simplified the build process.

3. *Fuselage Shell & Hatch*: The shell was shaped by hand from blue foam with templates for guidance. The foam was covered in MonoKote so a female mold could be laid up in fiberglass. A 2-ply carbon skin was laid-up in the mold. All details would be cut into the shell after it was fabbed. The hatch features 3 small hinges and a latch on the side for easy access to the payload. One alternative was considered. The blue foam master could have been used as a male mold, eliminating a step. However, this process was used last year, and the surface quality proved very poor.
4. *Wing and Empennage*: The wing was constructed by hot-wiring foam core sections using templates, then covering the foam with a layer of fiberglass. Carbon caps were added to carry bending loads. Separate structural tests on two-foot wing samples indicated that the caps alone would carry the bending loads and hence no "hard" spar webs were used. This eliminated a cutting step, web fab, and web installation. Instead, extra material was added to the leading edge section to provide shear continuity cap-to-cap. The foam provided core crush resistance. The pieces of foam and the spar caps where then put together, and vacuum bagged to finalize the wing. The empennage followed a similar construction to the wing. A dedicated spar web was a serious alternative since it was the lightest approach. However, reduced complexity and better surface finish argued for a spar-less wing. A Balsa built-up wing was eliminated by virtue of complexity.

6.4 Manufacturing Schedule

Figure 6.1 shows the manufacturing schedule, which details the build times for each component of the aircraft.

Table 6.1. Skill Matrix for ADT Workmanship rated on a scale from 1-5, with 5 defining a very high level of workmanship.

Member Name	Team Experience (Yrs)	Composite Workmanship	Machining Workmanship	Wood Workmanship
George Sechrist	5	5	5	5
Jerry Chen	4	5	5	5
Phillippe Kassouf	4	4	5	5
David Lazzara	4	3	5	5
Jacob Evert	5	5	5	5
Tim Bentley	3	3	5	5
Charles Heintz	4	4	5	5
George Cano	2	2	4	4
Stephane Gallet	2	3	4	4
Tyler Golightly	2	3	4	4
Jonathan Hartley	2	4	3	4
Tai Merzel	2	2	3	3
Brian Wetzel	2	1	3	3
Andres Figureroa	1	1	3	3
Billy Kaplan	1	1	3	3
Michael Mace	1	1	3	3
Cristina Nichitean	1	1	2	2
Doris Pease	1	1	2	2
Tim Schoen	1	1	3	3
Lester Kang	1	1	1	1

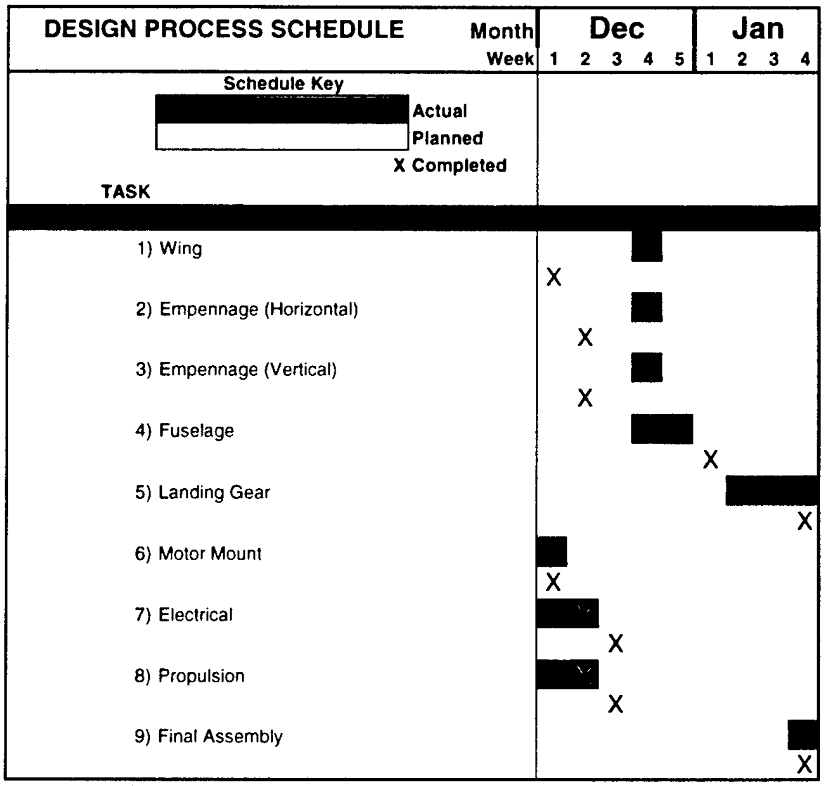


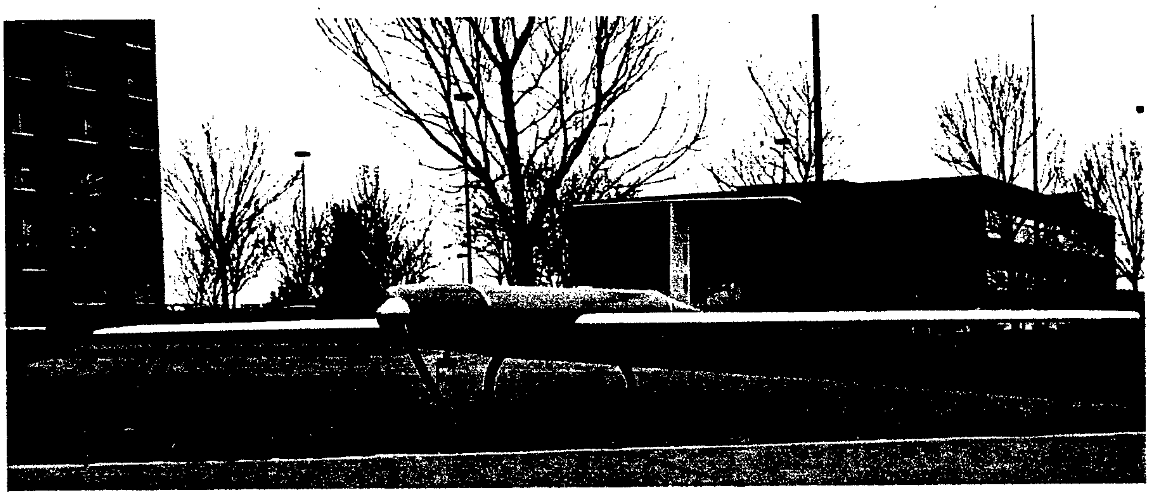
Figure 6.1. The Manufacturing Milestones Chart, showing planned and actual timing of construction events.

phastball

WEST VIRGINIA UNIVERSITY

April 26th – 28th, 2002

AIAA Cessna ONR – Student Design/Build/Fly Competition



DEPARTMENT OF MECHANICAL AND AEROSPACE ENGINEERING

WEST VIRGINIA UNIVERSITY

MORGANTOWN, WV 26506

Table of Contents

1.0 Executive Summary

From the beginning of the design process, phastball was to be one of the most competitive planes that West Virginia University has ever built. Utilizing a combination of composite and traditional materials along with a focus on weight and flight performance, the plane is a razors edge balance of weight, strength, and performance. Newfound levels of construction detail have been incorporated into this year's entry, which is apparent when one notices the immaculate internal detail as well as the clean aerodynamic design and shape. From tip to tail, phastball promises to outperform the competition in all respects.

Upon consideration of this year's requirements, a streamlined design with low drag was going to be one of our greatest concerns in order to remain competitive. The balance of composite and traditional materials was considered carefully with respect to strength in target areas, weight in non-load bearing sections, and speed and ease of construction.

Based on the team members' previous knowledge of composite materials, the fuselage was constructed with composite materials, with structural reinforcement using traditional materials such as balsa wood and plywood. The wings are of a less exotic build, balsawood-sheeted foam, due to the inherent complexity that arises from attempting to produce exacting, complex wing geometries. With proper reinforcement, there is very little structural performance difference between the composite wings and the foam and balsawood wings. The fact that the wings do not require the time intensive process of making a mold means that they can be constructed in about one week.

The next consideration was the speed and flight performance of the aircraft. In order to reduce drag, the smallest possible fuselage cross-section was designed. There were as few sharp transitions in geometry as possible with the best utilization of internal space. All external surfaces were blended, with the purpose of reducing drag as much as possible.

The wing geometry is another facet of the design that greatly influences the performance. A triple break planform with slight dihedral and twist allows for an amazing combination of near elliptical lift distribution, high lift to drag ratio, and increased maneuverability.

Several tail geometries were considered, T-tail, V configuration and conventional tail. The decision to use the T-tail seemed natural due to the fact that it nearly eliminates the effects of downwash on the tail section.

Propulsion was debated down to two considerations. The first was a single motor, tractor configuration. The second was two motors mounted in the wing. Endurance, efficiency and low drag were the real deciding factors along with the added weight and point penalties for the extra motor. It was calculated that the single motor with proper prop choice could provide the needed thrust. This could be done without the added weight and loss of flight time due to the additional weight and battery drain created by the second motor. In addition, wing mounted motor nacelles add unwanted complexity and flow disturbance to the wing.

Another addition to the design was the use of flow-through air-cooling of internal components. Utilizing strategically placed air intakes, the motor and battery packs are cooled using forced air convection. Efficiency of both the batteries and motor are enhanced by lowering the temperature, which minimizes losses. Granted, there are some losses in the area of drag produced, but it is felt that the cooling effect outweighs the small drag increase.

To achieve this design several software packages were used in the design analysis. Programs were written in MATLAB and Maple to analyze to simulate the aircraft flight simulation. Additional programs such as Microsoft Excel, AutoCAD, ElectricCalc and MotoCalc were also used in the design phase of the project.

2.0 Management Summary

2.1 Architecture of the Design Team

Team phastball was comprised of three graduate students Kevin Ford, Michael Julius, and Edward Wen, two upperclassmen Jesse Ashby, Jason Gill and three underclassmen Jonathan Breckenridge, Peter Cooke, and Kuntal Vora. Peter Cooke, Edward Wen, and Kevin Ford have competed in the Design/Build/Fly competition in previous years. The three of them were in charge of organizing the team and organizing the different aspects of the design and construction process. The team met once a week to discuss the different aspects of the project and assigned individual work accordingly. For example, if the team were working on the wing, the work would be divided among those who were present. Some members would rough cut blocks of foam, others would cut templates, while others made skins to sheet the wings. This approach to the team architecture allowed everyone to work at their leisure depending on their individual schedules. Our goal with this approach was to have as many team members able to share any ideas they might have, fabricate any part, as it was needed, and to bring everyone's experience to the same level.

2.2 Assignment Areas and Management Summary

As described above, each team member was able to participate in every phase of the design and construction of the aircraft. There were however, three areas were experienced team members needed to take charge. Peter Cooke was the chief construction engineer, having over fourteen years of experience in constructing and flying RC aircraft. Kevin Ford was in charge of making purchases and getting materials and money donated to the team. Edward Wen was in charge of the design of the aerodynamics and the stability and control of the aircraft. Everyone on the team either had classes with each other or saw each other on a daily basis this kept the team members in close contact so ideas could be shared. Weekends proved to be the best time for everyone to meet to work for long periods of time. However, a lot of the work was done whenever the team members had free time. When team members had free time, they would get in contact with one another to work on the aircraft. Using this approach maximized the amount that was done by everyone. No one would be required to be at any certain place or time, which

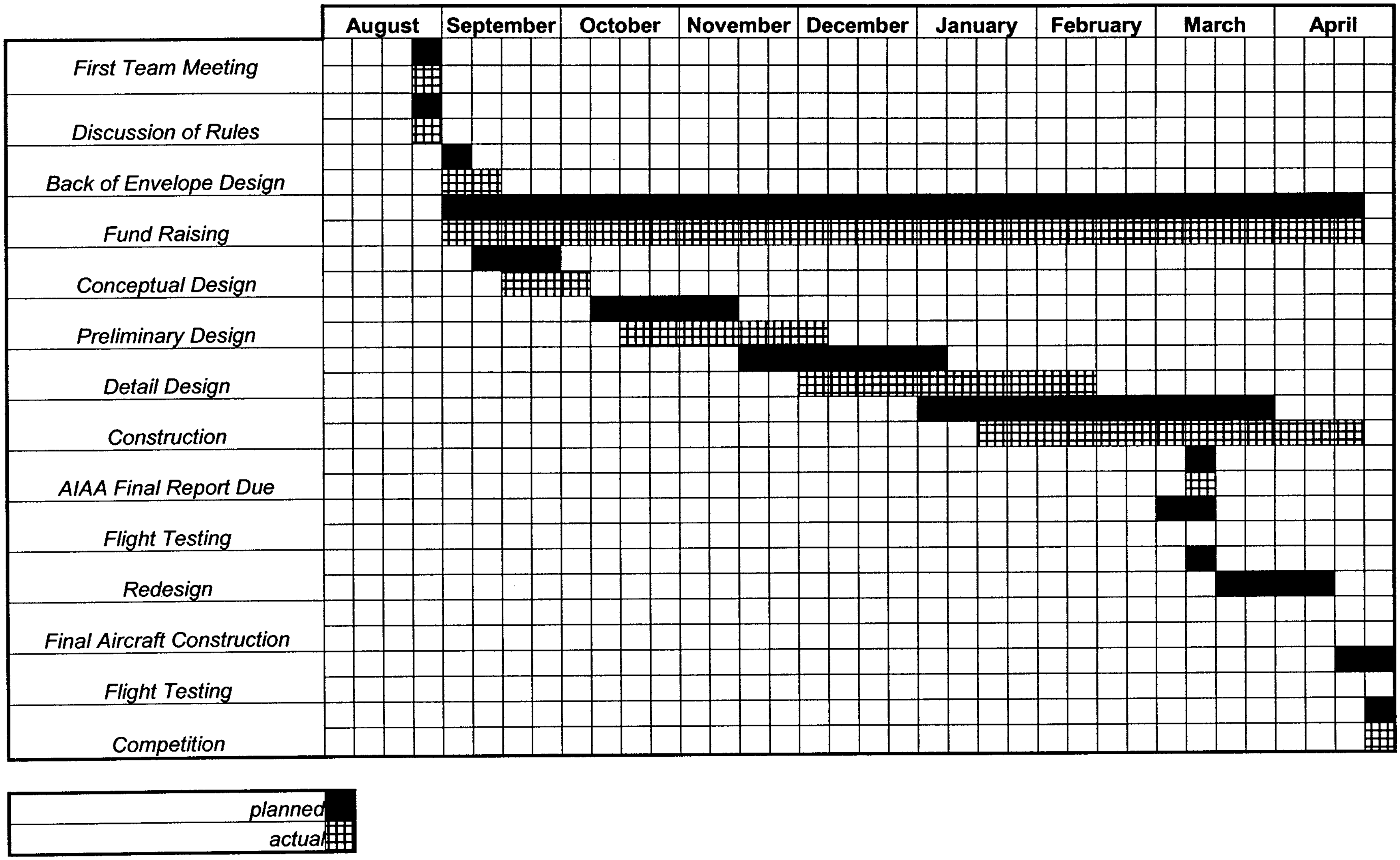
kept the project fresh for everyone. Table 2.1 is a summary of the project milestones' planned and actual start and finish dates. These dates were set so that each phase of the project (report and fly-off) would be finished by the dates set forth in the competition rules.

2.3 Personnel Goals and Accomplishments

Even though the majority of the work done on the aircraft was preformed by who was there at the time the team would like to acknowledge the members who went above and beyond on certain phases of the aircraft.

- Construction of the plug, mold, and fuselage – Pete and Kevin
- Landing gear and wheel pants - Jason and Jesse
- Booking reservations – Kuntal
- Design of the wing and stability and control analysis – Ed
- Fund raising – Jonathan and Mike

Table 2.1 Project Milestone Summary



3.0 Conceptual Design

3.1 Discussion of Rules and Strategy

The mission profile of this year's competition was to design and construct an aircraft that can carry ten to twenty four softballs that can be easily removed and is able to fly without payload. The softballs must be a minimum of two directly abreast, having no staggered rows, and single height. Three different missions must be flown during the competition. During the first mission, "Position", the aircraft must fly two unloaded laps, complete a 360° turn in the direction opposite of the base, and final turns on the downwind leg of each lap. During the second mission, "Passenger Delivery", the aircraft must land and the ground grew must load the payload, take-off and fly two laps, completing the 360° turn as described above. During the third mission, "Return", the aircraft must land and the payload must be unloaded, the aircraft is then to fly two full laps of the course. For each mission the aircraft is required to take-off within 200ft. Figure 3.1 is a graphical representation of the mission profile. The following list highlights the competition rules:

- No payload may be carried internal to the wing proper
- Must be propeller driven and electric powered with an unmodified, over the counter model electric motor
- All motors must be from the Graupner or AstroFlight families of brushed electric motors
- Motors and batteries will be limited to a maximum of 40 Amp current draw
- Must use over the counter NiCad batteries
- Maximum battery pack weight is 5.0 lb
- Take-off gross weight with payload must be less than 55 lb

Based on the flight performance of the aircraft the single flight score is calculated using the following formula:

$$Single\ \ Flight\ \ Score = \frac{total\,\#\,laps\ \ flown\ +\ \#\,balls\ \ carried}{total\ \ mission\ \ time} \tag{3.1}$$

The best three of five flights are then added together to achieve the total flight score. This is then entered into the following equation to determine the team score:

$$SCORE = \frac{written\ \ report\ \ score\ *\ total\ \ flight\ \ score}{rated\ \ aircraft\ \ cost} \tag{3.2}$$

The rated aircraft cost, RAC, is penalty imposed on each team based on the simplicity of the aircraft. The RAC takes into account the number and size of the wings, tail, motors, batteries, servos, weight of the aircraft. Initial optimization of the aircraft was based on these rules and scoring methods.

3.2 Design Parameters Investigated

The team began the design of the 2002 entry by examining what worked for past WVU teams, other schools, as well as some new ideas, which were proposed by current members. Since several team members have been involved with this project for a long time and others have a great interest in RC aircraft a lot of time was saved on the simple principle that we have seen it work or we have seen it fail.

During the "back of the envelope" design phase, team members discussed the rules and everyone suggested their initial ideas to each other. From these discussions, three prominent aircraft configurations emerged. These ideas were further discussed and a consensus was formed that the following parameters were most import in the design of the aircraft: wing planform, tail configuration, payload and fuselage configuration, power plant, and landing gear. These parameters were then looked at based on their Figure of Merit, FOM. Using this approach early in the design stage allowed the team to quickly dismiss ideas that were either to difficult to construct or that the team felt would just not work. Table 3.1 Is the FOM breakdown of the individual design parameters.

3.3 Figures of Merit Breakdown

In the conceptual design stage, several generic baseline aircraft configurations were considered for further development. These configurations consisted of different wing planforms, tail configurations, payload size, and landing gear configurations. The number of each particular component was then evaluated based upon a set of Figures of Merit determined by the team, according to how the different numbers of each component would affect various aspects of the aircraft performance and RAC. If the number of components was favorable to a particular design variable, a value of 3 was assigned to that number of components. If the number of components was unfavorable, a value of 1 was assigned. If the number of components had a negligible effect on the design parameter, a 2 was assigned, the number 0 was assigned if it did not pertain to the parameter. The number of motors, wings, etc. with the highest FOM sum was selected for further evaluation.

Ease of construction

The competition requires each team to design, construct, and fly their aircraft. Approximately one school year was allotted for the team to complete everything. A weight multiplier of 3 was given to the construction of each component. The quicker the team could manufacture each component the more time could be spent optimizing and testing it.

Strength to Weight Ratio

Lowering the weight of the aircraft allows it to fly faster, carry more payload, and have a lower RAC. Each component of the aircraft must be built as light as possible yet withstand the loads that occur during the flight missions. Therefore, a weight multiplier of 2 was given to this category.

Handling

Due to the nature of the mission profile flight and ground handling qualities of the aircraft were discussed. During the "Position" and "Passenger Delivery" missions the aircraft must complete a 360o turn on each leg of the flight. Ground handling qualities of arise when the aircraft must taxi back to the starting line to begin the next mission.

Ease of Changing Payload

The total mission time is comprised of the time it takes the aircraft to navigate the course and the time it takes the ground crew to load and unload the payload. Different configurations of wing vertical placement, motor location, and tail type can affect the amount of time that is spent on the ground.

Rated Aircraft Cost

Different configurations and sizes of the aircraft components not only affect the flight characteristics of the aircraft the also affect the rated aircraft cost, RAC. Here the team examined how each component's parameters affected the RAC. The RAC is further discussed in section 3.6.

Cost

The cost of batteries, motors, servos, and other materials was a big concern for the team. Materials such as composites and epoxies were able to be purchased at a discount our received through donation. Other components such as batteries and motor needed to be purchased. Here the team looked at the components from previous year's aircraft that could be used over again and tried to minimize the amount of new components needed to be purchased

3.4 Figure of Merit Selection Process

3.4.1 Wing Planform

Elliptical (rejected)

Construction of an elliptical wing is too difficult. Cutting a foam core to shape could not be done. It therefore would have been shaped by hand and was deemed to difficult to construct.

Tapered (adopted)

A tapered wing is fairly easy to construct and has produced good results on past aircraft. It was therefore kept for further analysis.

Multi-break (adopted)

Multi-break planforms give similar characteristics to elliptical wings but are much easier to construct. The construction method is very similar to that of a tapered wing.

3.4.2 Wing Location

High (rejected)

The main benefit of a high wing is that it allows the fuselage to be placed lower to the ground and it has the lowest drag interference. After observing last year's competition, it was decided that the easiest access to the payload would be to load and unload from the top of the fuselage. A high wing was rejected since it would hinder the removable of the top half of the fuselage.

Mid (adopted)

Using a mid wing would give the advantage of tying the carry through spar structure in with the landing gear support and would also act as bulkhead in the fuselage. The team highly favored this approach due to its success on the 1997-1998 WVU entry.

Low (adopted)

Blending a low wing with the fuselage seemed like it would add to construction and frontal area. However, the low wing was kept since the final payload loading and unloading process was not yet decided.

Biplane (rejected)

Using a biplane would drastically raise the RAC and aircraft drag. Since there was no constraint on wingspan it was quickly rejected by the team.

3.4.3 Tail Configuration

Conventional (adopted)

The conventional tail is probably the simplest to construct and attach pushrods to servos. It is lighter than a T-tail since the vertical tail does not need to support the horizontal tail. However, the horizontal tail is in the wing wake and prop wash. Steering can be controlled with the same servo as the rudder and the elevator servo can be used to control the brakes.

V-tail (rejected)

The advantage of using a V-tail is reduced interference drag. The disadvantage is that the desirable handling qualities are not as stable as a T-tail or conventional tail. Using a V-tail would also raise the RAC by requiring another servo to control steering.

T-Tail (adopted)

The advantages of a T-tail are that the horizontal tail is lifted out of the wing wake and prop wash and that the vertical tail can be smaller due to end-plate effects. As with the conventional tail steering can be controlled with the same servo as the rudder and the elevator servo can be used to control the brakes.

Canard (rejected)

Canards are difficult to design so that they provide sufficient stability. More servos would have been required to control the aircraft raising the RAC. The team also felt that constructing a canard would add to the complexity of the aircraft and to the time spent constructing it.

3.4.5 Landing Gear

Tricycle (adopted) vs. Tail Dragger (rejected)

The payload can only be changed once the aircraft is at the starting line. Thus stable ground handling was important when selecting the landing gear. Since the location of the c.g. is behind the main gear an aircraft with a tail dragger configuration will tighten in a ground loop, resulting in poor ground handling qualities. Due to instability the tail dragger landing gear was rejected, tricycle landing gear does not have this problem. Stopping the aircraft once it reaches the starting line is also crucial. On a tail dragger, the brakes on the main gear would cause the aircraft to tip forward hitting the prop on the ground. With tricycle, gear the nose wheel prevents the aircraft from tipping forward.

3.4.6 Motors

One Motor (adopted) vs. Two Motors (rejected)

Several things factored into the selection of using only one motor. It was felt that AstroFlight had single motors capable of producing the required power. Second, the RAC with a second motor is much greater than that of just one motor.

3.5 Material Selection and Construction Techniques

The main goal in constructing any aircraft is to make each component as strong as needed and light as possible. Additionally, the team members had to take into account the required skill, cost, and

time to construct the aircraft using different construction techniques. This posed several problems. The team member's experience was varied. Because of this limitation, special "classes" were set up by the senior team members to increase everyone's knowledge in different construction techniques. Time was also a limitation. Each member was taking a full course load, so construction time was limited. The team discussed typical UAV construction techniques in an attempt to determine the appropriate method needed to construct each component. Balsa wood, foam build-up, and composite material construction were investigated for each component.

3.5.1 Balsa Wood

One of the cheaper methods investigated, balsa wood construction is typically used to fabricate kit radio-controlled aircraft. This method consists of cutting ribs, spars, and stringers individually, and then assembling them to construct the wing, tail, or fuselage. However, as was learned with previous entries, this method is tedious and time consuming. However, this method was not completely ruled out since some components require minimal strength. Several members had previous experience working on a previous entry that used a balsa wing and ruled out the use of that type of wing due to the difficulties mentioned above.

3.5.2 Foam Build-up

The cheapest and easiest construction method is foam build-up method. This method has typically been used to construct UAV aircraft wings and consists of making templates of the desired airfoil shape and then using a hot-wire to cut the component out of a block of foam. A surface such as a thin layer of balsa wood or fiberglass is then applied. Previous WVU entries successfully used this method to build the wings. Balsa wood or spruce spars can easily be added to increase the strength of this type of wing. Foam build-up was also discussed for construction of the tail.

3.5.3 Composite Materials

Last year's team used composite materials for the construction of the wing. This was a laborious process that had several drawbacks. First, the time spent during construction was for one wing. If the team discovered a flaw in the wing design the time spent constructing the plug and mold would have been wasted. Second, the weight of last year's wings was much greater than predicted. Lightweight composite materials such as 1k carbon fiber are expensive. Due to these reasons the team felt that constructing the wing out of composites was not the way to go. Since the fuselage needs to basically be a hollow shell to house the payload, propulsion system, and avionics composite materials were highly discussed as a method to construct the fuselage. Repeatability is another advantage of composites. Once the required mold has been constructed, it can be used several times to produce identical parts. Previous WVU entries have also utilized composite materials for the construction of the landing gear. The 1997-1998 entry used a simple mold for the hand lay-up and vacuum bagging process. This proved to be a quick method to construct the landing gear.

3.6 Discussion of Rated Aircraft Cost

The rated aircraft cost can basically be described as the simplicity of the aircraft. It is a way for the competition to reward a team who can design an aircraft to perform the same goal with either a smaller or lighter aircraft, an aircraft with less batteries, an aircraft with a smaller wing area, etc.

3.6.1 Breakdown of the Rated Aircraft Cost

The overall rated aircraft cost (RAC) is calculated by the equation:

$$RAC,(\$thousands) = \frac{A \bullet MEW + B \bullet REP + C \bullet MFHR}{1000} \tag{3.3}$$

In the above equation, A, B, and C represent values of multipliers used to convert aircraft characteristics into manufacturing hours. The Manufacturer's Empty Weight Multiplier (MEW), Rated Engine Power (REP), and the Manufacturing Man Hours (MFHR) are a breakdown of the different aircraft components to be converted to man-hours. A more detailed explanation of these parameters is described below.

3.6.2 Manufacturer's Empty Weight Multiplier

The manufacturer's empty weight multiplier, MEW, is comprised of the weight of the aircraft without payload. In equation 3.3 A represents the manufacturer's empty weight multiplier and was given a value of $100/lb.

3.6.3 Rated Engine Power

The rated engine power, REP, was calculated using the following equation:

$$REP = battery \quad weight\left(1 + 0.25 * \#motors\right) \tag{3.4}$$

In equation 3.3 B represents the rated engine power multiplier and was given a value of $1500.

3.6.4 Manufacturing Man Hours

The manufacturing man hours, MFHR, further breaks down the components of the aircraft into the work breakdown structure, (WBS), and takes into account the number of wings and their size, the number of fuselages/pods, and their size, the number of horizontal and vertical surfaces, the number of servos used, and the number of motors and propellers used. In equation, 3.3 C represents the manufacturing cost multiplier and was assigned a value of $20/hr.

3.6.5 WBS 1.0 Wing

The rules state that 8hrs/ft of wing span, 8hrs/ft of max exposed wing chord, and 3hrs per control surface are to be used in the calculation of WBS 1.0.

3.6.6 WBS 2.0 Fuselage

WBS 2.0 consists of the length of the fuselage. 10hr/ft are used to calculate WBS 2.0.

3.6.7 WBS 3.0 Empenage

WBS 3.0 is the number of vertical and horizontal surfaces. For any vertical surface (winglets, struts, end plates, etc.) a penalty of 5hr per surface is to be assessed. For any vertical surface with an active control 10hr per surface is assessed. Any horizontal surface that is less than 25% of the span of the greatest span is assessed 10hr per surface.

3.6.8 WBS 4.0 Flight System

WBS 4.0 is the number of servos and controllers used on the aircraft. 5hrs was assigned per servo or controller.

3.6.9 WBS 5.0 Propulsion System

WBS 5.0 is the number of motors and propellers or fans used on the aircraft. 5hrs per motor, propeller, and fan was assigned.

3.7 Features that Produced the Final Configuration

Several questions had to be answered before the final configuration of the aircraft was decided upon. First, do we think this will work and can we construct it? Answers to this question were very valuable since several of the team members are RC aircraft builders and pilots. How does this affect the RAC was also asked. As described above adding another servo or changing the aspect ratio of the wing greatly affects the RAC. Lastly, the key design parameters described in section 3.2 and the FOMs described in section 3.3 produced the final configuration. Below is an outline of the final configuration:

- Mono wing either tapered or multi-break planform
- Single fuselage
- Single motor
- Conventional or T-tail depending on wing location
- Tricycle landing gear with breaks
- Mid or low wing depending on how payload is loaded - unloaded

Table 3.1 Figure of Merit breakdown of design parameters

level of importance
3 favorable
2 neutral
1 unfavorable
0 does not pertain

		Ease of construction (x3)	Strength to Weight Ratio (x2)	Flight and Ground Handling (x1)	Ease of Changing Payload (1x)	RAC (x2)	Cost (x1)	Total FOM	Adopt (A) or Reject (R)
Wing Planform	Elliptical	1	2	3	2	3	3	21	R
	Tapered	*3*	*2*	*3*	*2*	*3*	*3*	*27*	*A*
	Multi-break	*3*	*2*	*3*	*2*	*3*	*3*	*27*	*A*
Wing Location in Fuselage	High	1	3	2	1	2	2	18	R
	Mid	*3*	*2*	*2*	*3*	*2*	*2*	*24*	*A*
	Low	*1*	*3*	*2*	*3*	*2*	*2*	*20*	*A*
	Bi-plane	0	1	3	1	1	1	9	R
Tail Configuration	*Conventional*	*3*	*2*	*3*	*0*	*3*	*0*	*22*	*A*
	V-Tail	2	2	3	0	1	0	15	R
	T-Tail	*3*	*1*	*3*	*0*	*3*	*0*	*20*	*A*
	Canard	1	2	1	0	1	0	10	R
Landing Gear	*Tricycle*	*3*	*2*	*3*	*0*	*0*	*0*	*16*	*A*
	Tail Dragger	3	2	1	0	0	0	14	R
Motors	*One motor*	*3*	*0*	*2*	*2*	*3*	*3*	*22*	*A*
	Two motors	2	0	2	1	1	1	12	R

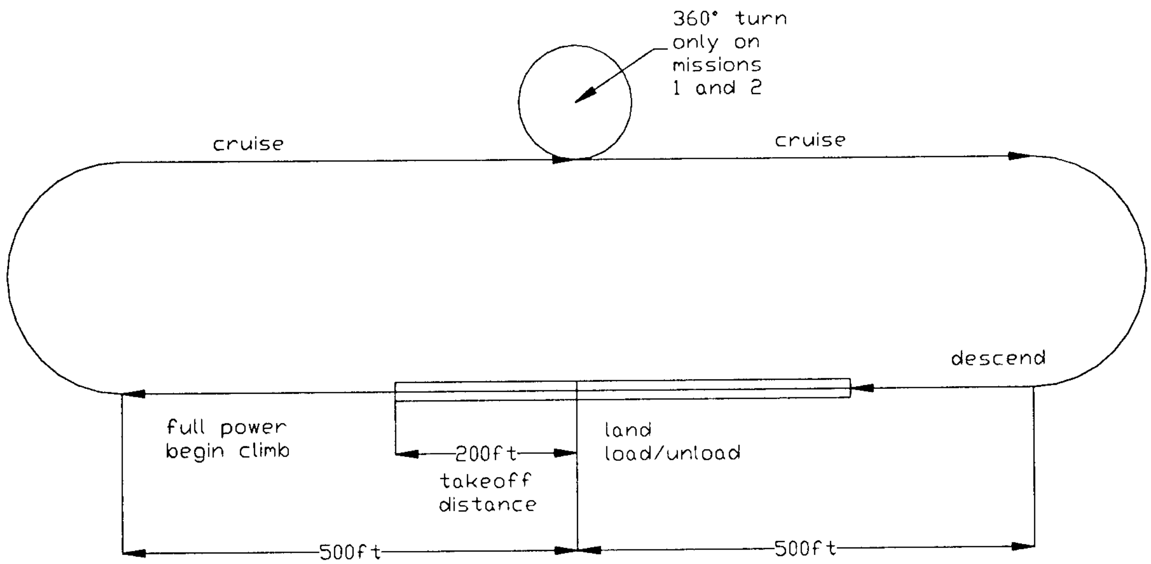


Figure 3.1 Mission Profile

4.0 Preliminary Design

4.1 Figures of Merit (FOM)

The "First Order FOM" that had the greatest impact on the overall score was identified by varying the parameters within the scoring formulas. They were as follows:

- Payload Capability
- Motor Power
- Battery Endurance
- Top Speed
- Battery Weight

The "Second Order FOM" which made a lesser impact to the overall score but were also viewed as important were the following:

- Fuselage cross-section and length
- Wingspan and maximum chord
- Empty weight

4.2 Baseline Configuration

The most important design parameter of the aircraft was payload. The number of balls was the starting point that allowed optimization of the other variables. To capture the effect of payload variation from 10 to 24 balls, three aircraft were examined and optimized to determine what area of payload range maximized the total score. These aircraft were 10, 18 and 24 balls with the 24 ball aircraft used as the baseline.

4.3 Assumptions

Based on the team's experience and judgment, a low drag baseline aircraft was chosen such that design trade studies could be performed. A high aspect ratio (10 or greater), mid-fuselage wing to provide a combination of low parasite drag at high speeds, low induced drag in the high G turns and easy access to the payload. Using 24 balls and a 5-lb. battery pack, the maximum gross weight (MGW) was estimated to be 24-lbs. and maximum empty weight (MEW) at 14 lbs. The entire mission was estimated to be completed in 5-5.5 minutes with motor run time of 4-4.5 minutes.

4.4 Simulation Program

To quantify the FOM #1, a simulation program was written in MATLAB to analyze the overall effects of the design parameters on the flight times and final scores. This program stepped the aircraft through a time increment till the mission phase was completed. The results were checked with the exact solution of the distance-time-velocity integral using the symbolic manipulator MAPLE. The results from MAPLE showed that 0.1-sec increment was reasonably accurate.

4.5 Propulsion System

Because of other projects in the WVU wind tunnel, testing of motor/battery/thrust combinations was not possible. In addition, thrust data from the 97/98 competition used brushless motors and so the team relied on ElectricCalc and MotoCalc as a virtual engine test stand. Previous competition results showed

that these programs were usually optimistic in their prediction of thrust and battery duration and so a knock down factor of 0.90 was used to determine thrust as a function of velocity.

4.5.1 Motor Selection

Using data from previous competition's winning aircraft, power loading of 50 watts/lb. was used to determine the approximate motor size. The estimated weight of the aircraft was 24 lbs., which showed the AstroFlight 661 and 691 fitting the requirements. Both of these motors had a maximum continuous current capability of 35 amps and maximum voltage equivalent to 40 cells. Graupner motors were not considered because the larger size motors were only available through overseas purchase and the team already possessed AstroFlight speed controllers.

4.5.2 Battery Selection

For the past few years of this competition, the 2400-mah cells have been most popular because of their high capacity density (Table 4.1) and low internal resistances. Some 1400 and 1700 mah cells also have high energy density (mah/oz) but also higher internal resistance. In the event that the maximum weight of batteries would not be needed for the mission, alternative cell/pack combinations were examined to see if they could provide sufficient power by increasing voltage and decreasing capacity as shown in Table 4.1.

Cell zapping is used for high current (60A) electric RC competitions like the F5B. At the current levels in this competition (max 40A), the manufacturer showed 8% increase in voltage over unzapped cells. Zapped cells would be preferred but would depend on the team's ability to raise funds.

4.5.3 Propeller Selection

To provide the highest thrust at high speed and efficiency, the natural choice is for the pitch to diameter ratio (P/D) to be 1.0. P/D values higher or lower than 1.0 show a drop in efficiency as confirmed by several iterations performed with ElectricCalc and MotoCalc. For brevity, only P/D of 1.0 results are shown here.

4.5.4 Motor/Prop Combinations

Figure 4.1 shows some of the best motor/prop combinations using 36 x RC-2400 cells running approximately 4.5 minutes. The 691 and 661 motors were available in direct drive (DD) or geared (G) configuration. Currently, only the 2.75:1 gearbox is available commercially and no custom gearboxes or ratios for older gearboxes were considered in this trade study. The 641 motor was examined but could not produce sufficient power without increasing the number of motors and propulsion RAC.

Surprisingly, direct drive combinations would not produce more thrust than the best geared combinations unless the velocity was higher than ~120 ft/sec. However, the highest aircraft speed was estimated to be no more than ~120 ft/sec. Furthermore, both direct drive combinations were operating at the continuous current limit of 35A. At this amperage, serious heating problems could occur. Although the 691G (22 x 22) had the largest torque output available in the AstroFlight family of motors, the 661G with the correct propeller (18 x 18) could provide slightly more power to the propeller for the 4.5 run time. For speeds higher than ~85 ft/sec, the 661G combination provided more thrust than the 691G. The

weight and integration of the large propellers into the airplane also favored the 661G. These factors are shown in Table 4.2 and led to the final decision to use the 661G with 18 x 18 prop.

4.6 Fuselage

4.6.1 Fuselage Shape

The fuselage had the competing requirements of short length, low drag and high payload capability. Three fuselage shapes were considered based on three different softball-packing configurations in Figure 4.2. Using the drag estimation techniques from [Anderson, Hoerner] the drag was estimated. In Table 4.3, the ranking shows that 2x12 is the best overall design. Length was not included as a FOM because tail moment arm drives the fuselage length requirements.

4.6.2 Fuselage Cross Section

A full-length fairing was incorporated into the fuselage cross-section to house the batteries. These batteries could be moved forward or aft as necessary to adjust CG position. (See Figure 4.3)

4.7 Wing

4.7.1 Airfoil

In the 97/98 competition, some notable aircraft reached speeds of 70 to 80 mph (100-117 ft/sec). With the motor and payload restrictions this year, it appeared reasonable to set 70 mph as an attainable top speed in the Position and Return mission segments. The heavy emphasis on speed drove the team to consider low to moderately cambered airfoils providing low drag at the predicted lift coefficient of 0.15-0.25 yet reasonably high Clmax of 1.1 or greater. The only airfoils considered were from the UIUC wind tunnel database and the top picks were the MH32, S7012, SD7003, RG-15, and E387. In general the drag polar (Figure 4.4) shows that these airfoils were optimized to perform at different lift coefficients. The lift polar shows that all the selected airfoils provide at least 1.1 Clmax. Although the SD7003 had the lowest drag in the target Cl, the stall behavior appeared very sharp in comparison to the other airfoils. The next lowest drag nearest the target Cl was S7012. Since it had a wider drag bucket and gentler stall characteristics, the S7012 was chosen as the baseline airfoil.

4.7.2 Aspect Ratio and Wingspan

High aspect ratio wings (10 or higher) provide lower induced drag but many issues pointed to the benefits of reducing aspect ratio. First, the RAC penalizes longer vs. shorter wingspans. Next, with very high aspect ratios, wing chords become smaller, reducing Reynolds number and increasing parasite drag. Finally, a thinner narrower wing is structurally less efficient. To help find the optimum wing configuration, the aspect ratio was varied in the simulation program. Figure 4.5 shows that the mission time did not vary strongly with aspect ratio despite the large increase of drag in the turns as a percent of the total drag (Figure 4.6, 4.7). The main driver on wingspan was sizing the tail span since it could only be 25% of the wingspan (see tail section). An aspect ratio of 8 was chosen to allow for reasonable tail span and the literature shows ARs 7 to 9 to be the best compromise.

4.7.3 Wing Area

Historical data showed that a wing loading limit of 50 oz/ft^2 could be used with a high lift device to allow slower landing speeds. Based on the initial weight estimate with payload of 24 lbs., a wing area of 8 ft^2 was used which resulted in a span of 8 ft using an AR = 8. This wing area allows the aircraft to just meet the takeoff field length limit with 196 ft in the Passenger configuration and 59 ft in the Position configuration with no headwind. Since strong headwinds are likely at the competition, it is probable that the takeoff field length, TOFL, requirement will be met.

4.8 Empennage

4.8.1 Horizontal Tail

From [Raymer] horizontal tail volume coefficients in the range of 0.5 to 0.8 and AR of 3 to 5 were desirable for this similar type aircraft. The long fuselage puts payload far from the CG so initially a volume of 0.80 was considered appropriate. The tail span had to be 25% or less of wingspan or else it would be considered a wing under the RAC. The impact on the RAC to add a second wing was very high and therefore a lower aspect tail or lower tail volume was considered. An AR of 2 tail was deemed as to provide too much drag and too difficult to integrate into the vertical tail, therefore the tail volume and AR were reduced to the low range of 0.5 and 3.

4.8.2 Vertical Tail

Typical values for vertical tail volume are between 0.03 and 0.08. Since the T-tail adds an endplate effect to the vertical, extra margin was believed to be available in this configuration. Integration of the vertical to the horizontal tail set the chord and taper ratio requirements. A tail volume of 0.04 was used.

4.9 Flight Profile

The planned course for the pilot to fly was designed to keep total time down and to be at a safe altitude of 30 feet in the event of an emergency maneuver. After takeoff, a pilot flies a slow climb out to 30 feet by the time the plane reaches the first turn. All turns are executed at the same bank angle for simplicity of simulation. When the plane is directly across the field from the pilot, the 360° turn is initiated. To slow the plane down for landing, the engine is shutoff at the end of the downwind leg to bleed off speed in the base turn and final approach. Moving the 360° turn to the end of the straightaway is one possibility to reduce speed even more but was not used in the Preliminary Design.

4.10 Turn Optimization

The simulation program showed that the tighter the turns, the lower the flying time. The limitation on turns then became Clmax and structural capability. Using the characteristics of the aircraft in the Position configuration and allowing 90% of CLmax, the maximum turn G was approximately 6Gs (80°-bank angle). For the Passenger Delivery configuration, the maximum turn G is approximately 3Gs (70°-bank angle) (Figure 4.8). If the G limits were exceeded, a stall would occur in the first turn when speed has not been built up and the CL requirement is high for both Passenger and Position mission segments. Lowering the flaps was a possibility in the turns but not evaluated in the preliminary design. Considering

the maximum structural bending stress capability of the wings, the planned maximum G capability of the aircraft was set at 4Gs in the full payload configuration to provide a safety margin of 1.33.

4.11 Results of Aircraft Optimization

After several iterations with the simulation program, the second most important variable next to the number of balls was the weight of the battery pack. Figure 4.9 shows pack combinations that result in run time from 2.5 to 4.5 minutes and weights from 2.4 to 5 lbs. Using the individual trade study results for motor/prop, wing, fuselage, and battery cell type, ten study aircraft were put into the simulation program to evaluate Overall Score performance. (Figure 4.10) The Total Mission Time only varied from 4.3 to 4.9 minutes for all the aircraft studied and was not a strong driver for the Overall Score. In other words, extra speed from being smaller and lighter (10 balls) was good, but not good enough to overcome the fewer balls carried, even if the battery was very small.

An intermediate sized plane (18 balls) could be competitive but it would have to be very, very stingy on energy usage. The score of 153 (shown in dashed lines) with the 40 cell 1100-mah pack is pushing the pack to 100% usage and risking the 3-minute penalty for incomplete laps. In no case could it be shown to be advantageous to receive this penalty. The next best 18-ball configuration had a score of 143.

The aircraft with 24 balls provided much more flexibility. Configured with 40-cell 1400-mah pack, it would receive the highest score and also be pushing the pack limits, but this time only to 90% of capacity. A more conservative approach would be to use the 40 cell 1700 mah pack which would receive the same score as the best 18 ball aircraft while still having the option to improve upon that score.

4.12 Preliminary Design Summary

After trade studies on the motor/prop, cell type/pack size, wing/fuselage characteristics, flight profile, the major features of the aircraft based on the "First Order Drivers" were the following:

- Maximum payload for best scoring and propulsion flexibility (24 balls)
- Highest Thrust on expected speed range (661G motor, 18 x 18 prop)
- Maximum cell count available for motor for maximum power (40 cells)
- Cell capacity adjusted for mission duration and to reduce battery pack wt. (approx. 1700 mah)

The "Second Order Drivers", which affected the FOM but to a lesser extent, were:

- 2 x 12-ball fuselage for reduced drag, adequate tail arms.
- High G capable wing to reduce overall course length (approx. 4G)
- Sufficient wingspan to allow H-tail sizing (AR=8)

Table 4.1 Cell and Pack Characteristics for candidate battery cells

	Cell Characteristics					Pack Characteristics		
Cell Name	Capacity [Mah]	Wt/Cell [oz]	Int. Res. [Mohm]	Mah/oz	Mah/oz /Mohm	Max # Cell by Wt or Volt.	Pack Wt [oz]	Pack Energy [ft-lbs]
KR-1100AEL	1100	0.95	10.5	1,158	110	40	38.0	142,560
KR-1400AE	1400	1.05	11.5	1,333	116	40	42.0	181,440
CP-1700	1700	1.62	5.5	1,049	191	40	64.8	220,320
KR-1700AE	1700	1.48	8.5	1,149	135	40	59.2	220,320
RC-2000	2000	1.98	7	1,010	144	39	78.0	255,273
RC-2400	2355	2.15	4.9	1,095	224	36	78.0	276,817
CP-2400	2355	2.15	4.5	1,095	243	36	78.0	276,817

Table 4.2 FOM for motor/prop selection

Motor	Prop	Thrust	Heating	Prop Integration	Weight	Total
691G	22 x 22	3	3	1	1	8
661G	*18 x 18*	*3*	*3*	*2*	*2*	*10*
691DD	14 x 14	1	1	3	2	7
661G	12 x 12	1	1	3	3	8

Table 4.3 FOM for selection of payload configuration

Balls	Drag	Ease of Construction	Flexibility to Change	Total
2 x 12	2	3	3	8
3 x 8	1	3	3	7
2 x 12, 60% laminar	3	1	1	5

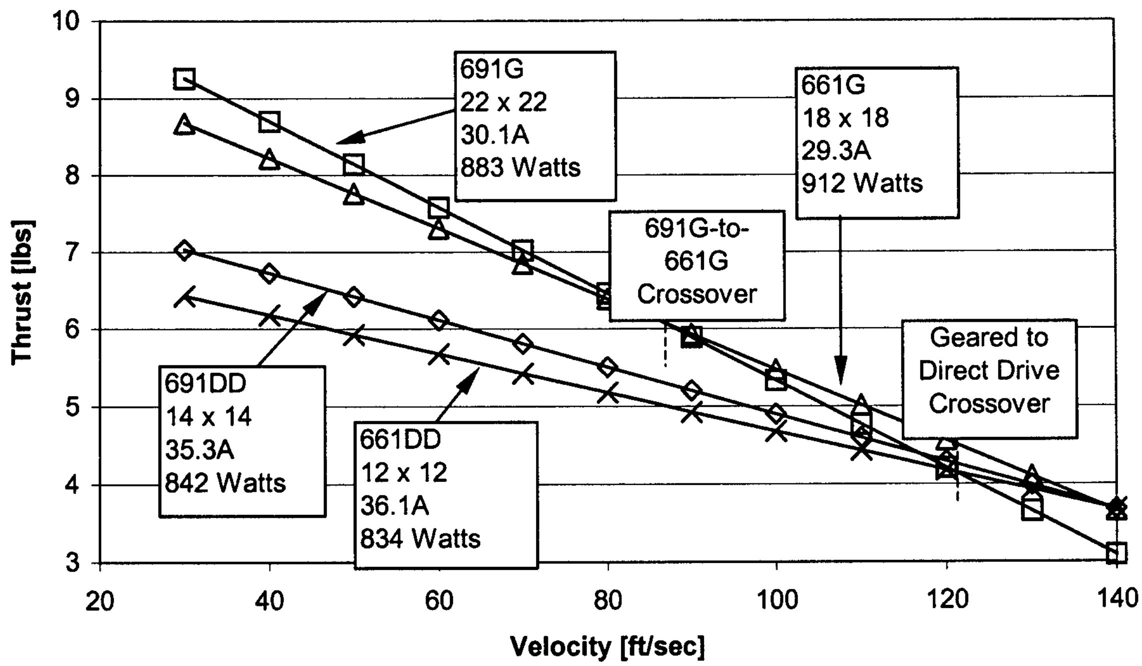


Figure 4.1 Best Motor/Prop combinations on 36 x RC-2400 with 4.5 minute duration

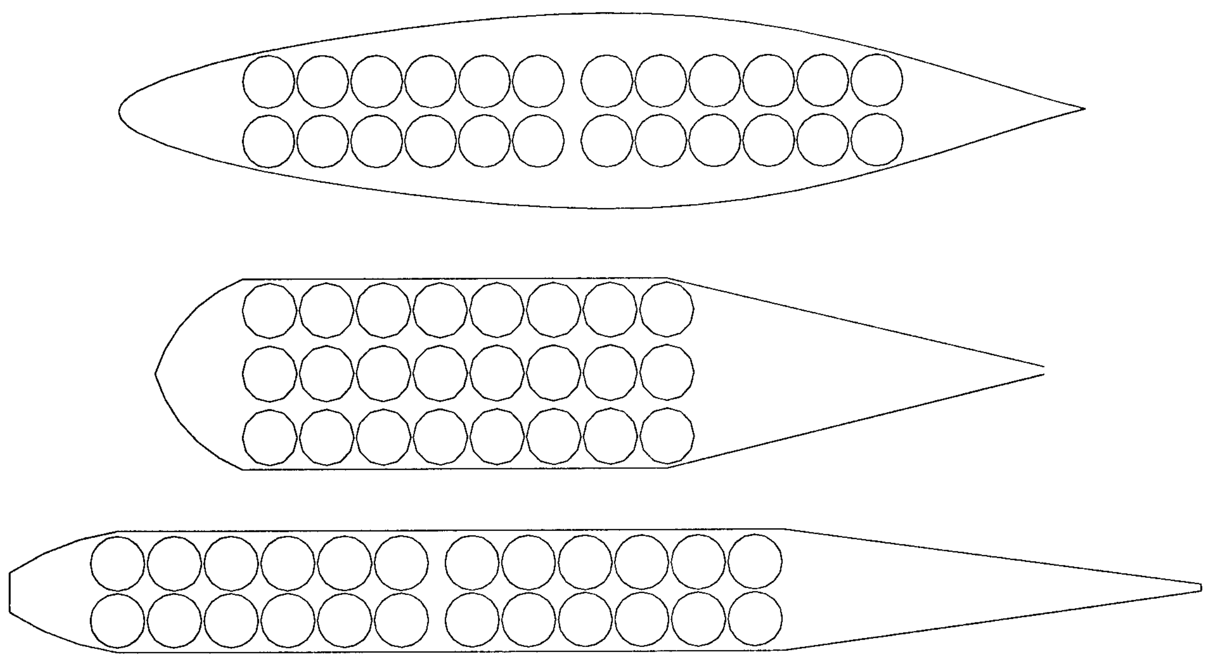


Figure 4.2 Fuselage payload configurations, 2 x 12 60% laminar, 3 x 8, and 2 x 12

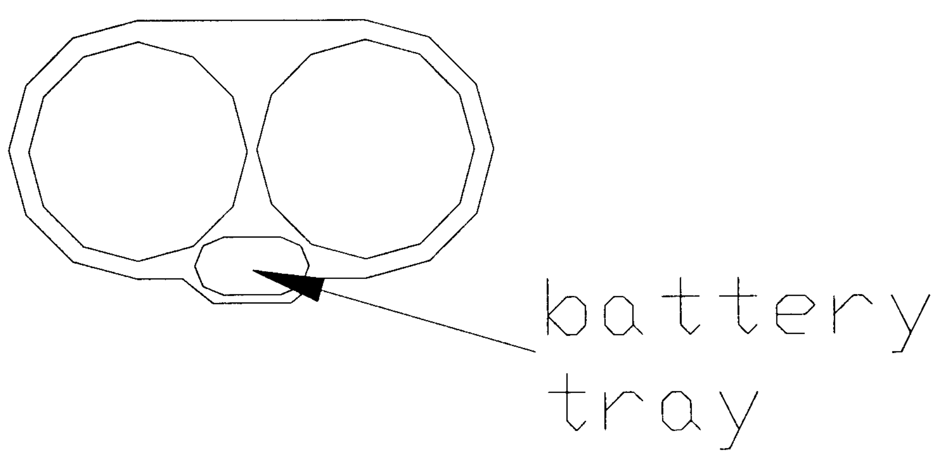


Figure 4.3 Fuselage cross-section showing adjustment of CG by moving battery pack

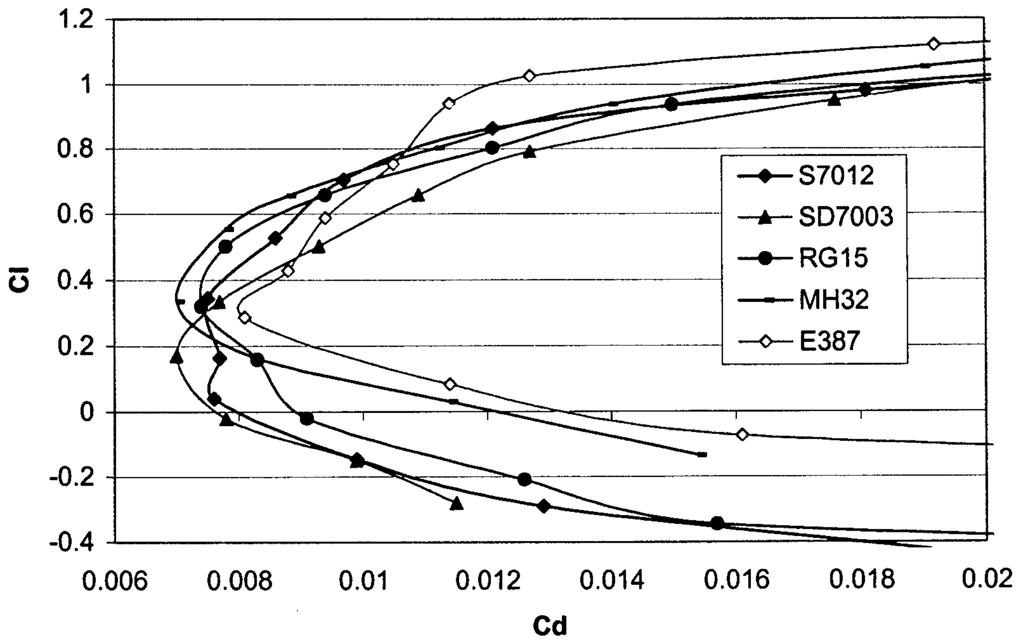


Figure 4.4 Airfoil comparison at Re = 300K

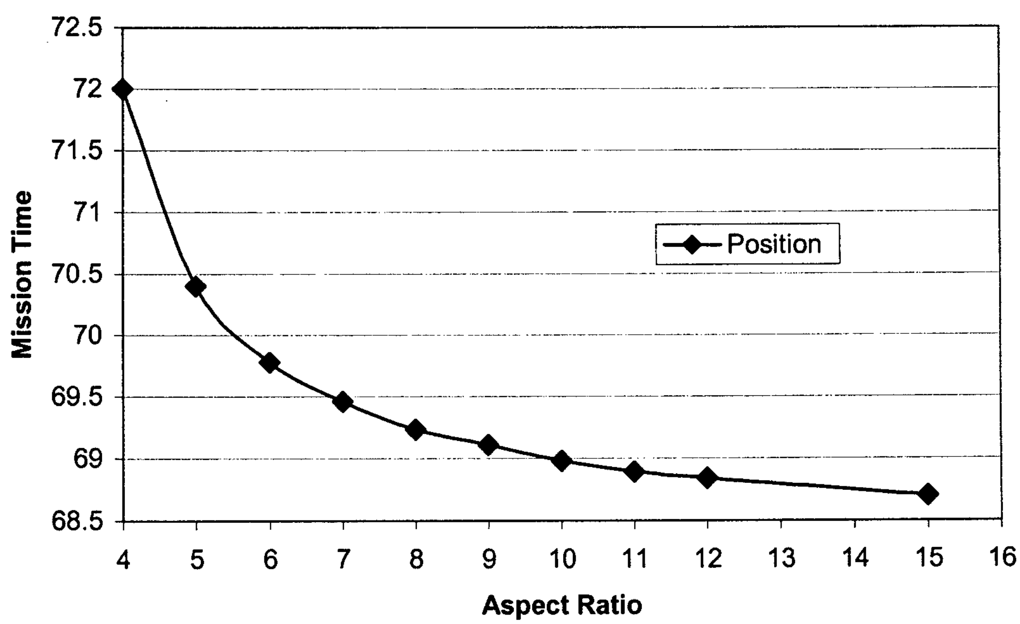


Figure 4.5 Reduction in Position Mission time with increasing AR

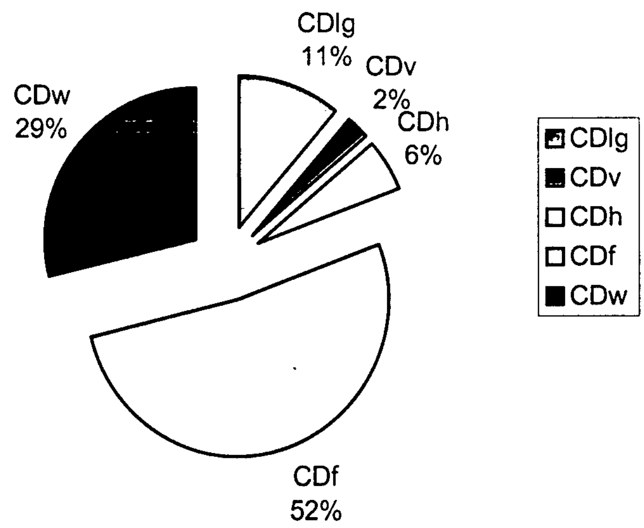


Figure 4.6 Drag breakdown at end of 1000 ft straightaway for passenger delivery

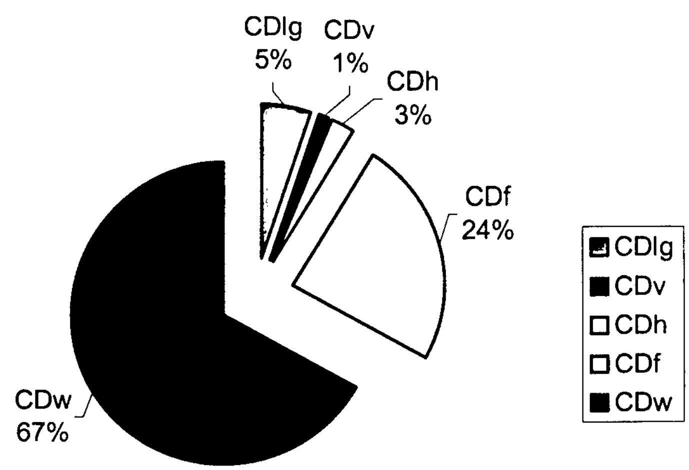


Figure 4.7 Drag breakdown at end of 3rd 180 deg for passenger delivery

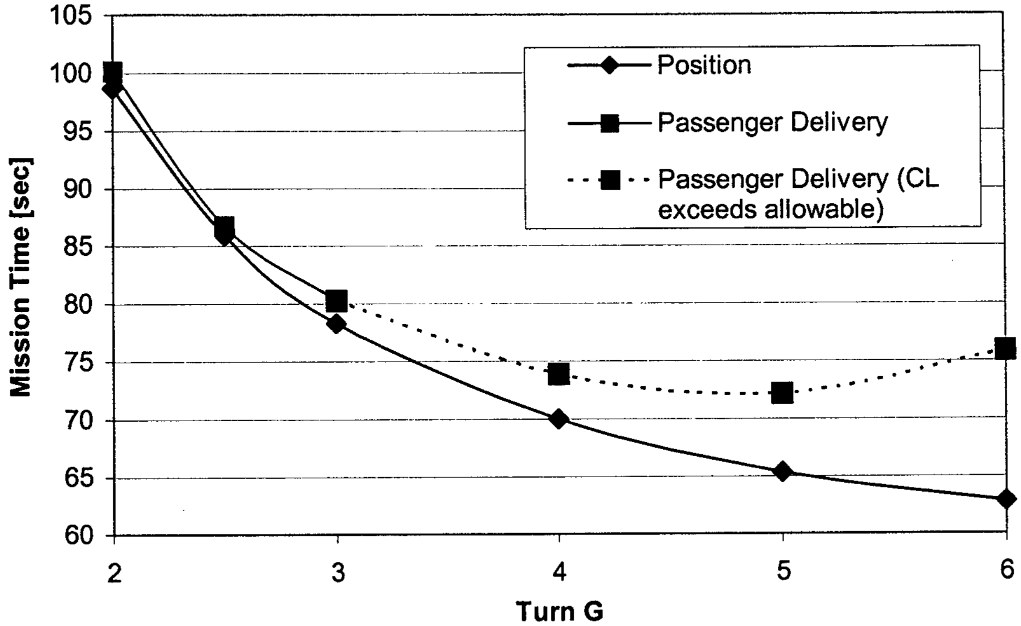


Figure 4.8 Limits on Turn G to reduce Mission Time

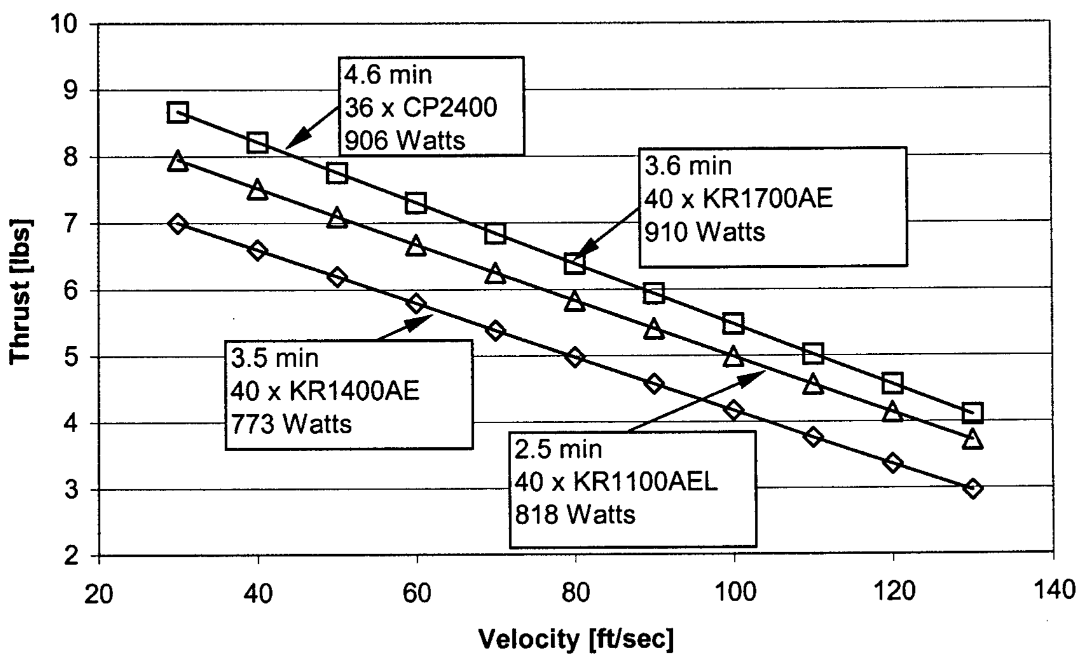


Figure 4.9 Battery pack combinations for a range of mission times

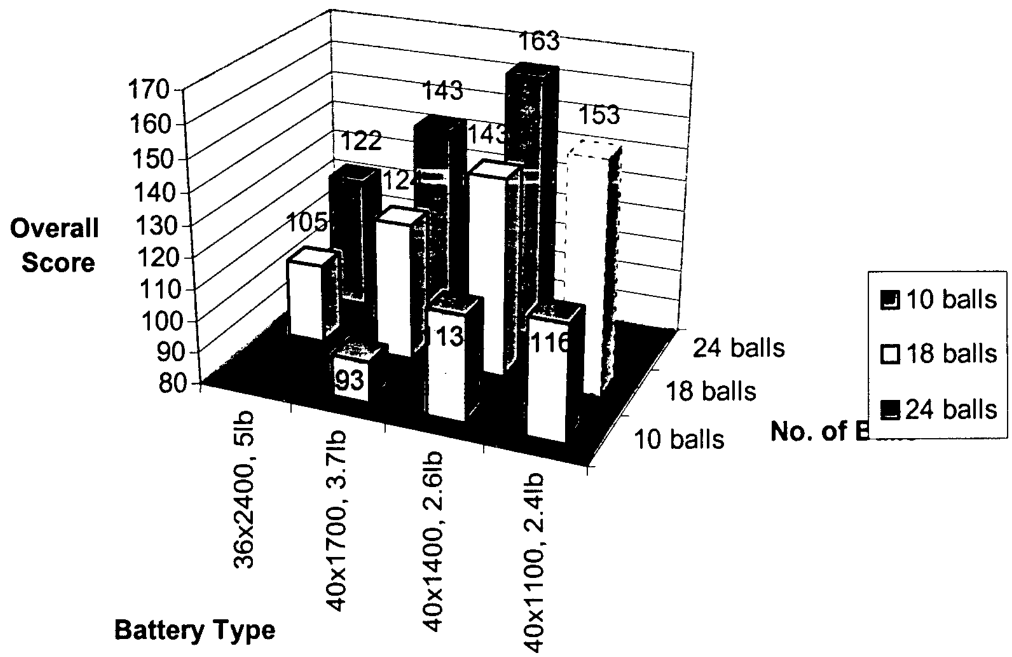


Figure 4.10 Overall Score for 10, 18, 24 ball airplane optimization

5.0 Detail Design

5.1 Weight Increase Items

A detailed look at the estimated weight showed that it required an increase of 2 to 3 pounds over the Preliminary Design weight estimate. This was mainly caused by the following:

- Decision to construct the wing in two halves required weight for the connection tube
- Structural reinforcement of fuselage for ground crew handling
- Actual weight of braking system higher than estimated weight
- T-tail stiffening required to reduce deflections
- Wing strengthening for 4G load with payload

Because these items would have increased the weight of the 10 and 18 ball aircraft, this change did not affect the relative strength of the 24 ball aircraft over the two study options, and therefore the payload configuration remained the same.

5.2 Wing Refinements

5.2.1 Wing Area

Because of the increase in weight, an 8.5% increase in wing area was made to keep wing loading to slightly below 50 oz/ft^2 and keep the TOFL within 200 ft. Keeping the same Aspect Ratio, the simulation program showed the wing area increase to have a minor effect on flight time. The RAC did increase slightly because of the increase in wingspan and max chord, but these changes were also very small.

5.2.2 Wing Planform

In the 97/98 DBF competition, the WVU entrant "Raptor" suffered from tip stall during high-G turns. The team used an untwisted RG15 airfoil with single taper planform at taper ratio 0.4 to reduce induced drag. This taper ratio was based on [Anderson], which showed the minimum induced drag occurs roughly at 0.3 for aspect ratios 4 to 10.

This year, the team wanted to design a plane with much better handling qualities in the turn, yet low overall induced drag. Based on information from the literature, many successful RC sailplane planforms used a constant local Cl from the root to 80% of semi-span and then dropped Cl off rapidly to the tip. In addition, these planforms used a combination of geometric and aerodynamic twist to provide protection against tip stall and insured that the entire wing passed through the zero lift line at the same angle of attack as the fuselage.

A program using the vortex lattice method outlined in [Anderson] was used to create and evaluate candidate planforms with the above guidelines in mind, as well as contribution to RAC. All planforms used an AR of 8 as determined from the previous study. To make a very flat Cl distribution from the root to 80% of semi-span, a large Cr was required, similar to an elliptical planform. Because of the penalty to the RAC for large Cr and span, a compromise position was taken with reduced Cr and almost constant Cl to 70% of semi-span. (Figure 5.1)

5.2.3 Final Airfoil Selection

As further protection against stall and for structural efficiency, a slightly thicker and cambered airfoil than the root S7012 was used on the tip. The S7032, SA7036, and SD7037 were evaluated using [UIUC wind tunnel] data (Figure 5.2). The SA7036 (9.2% t/c, 2.79% camber) was chosen because it provided lower drag compared to the other airfoils at low Cls, and provided 10% higher Clmax than the root S7012 over an alpha range of 11 to 15 degrees. In the planform, the S7012 went from root to 60% of semi-span and then transitioned to SA7036 at 92% of semi-span. In this section the wing is twisted nose down by the difference in the zero lift angles, or 1.9 degrees.

5.3 Handling Qualities

5.3.1 Horizontal-Tail

The low value for horizontal tail volume of 0.5 was kept in the detail design. The team had difficulty finding good data to approximate the static margin. Values for X_{AC} and CL_{α} on the wing-body were estimated from historical data but were very rough at best. The final value of the static margin was –0.18. The target value the team had expected was between -.20 and -.25 for reasonable handling qualities. The flight-testing program will be very conservative to determine if design changes are necessary.

Historical data shows elevator areas are typically 30 to 40% of total horizontal tail area. A value of 33% was chosen since it was felt that in a T-tail configuration, the tail is out of the downwash of the wing and its effectiveness is improved.

5.3.2 Vertical-Tail

To integrate well with the elevator, the rudder chord was made equal to the elevator. This corresponded to a rudder at 28% of total vertical tail area. This percentage is also typical for vertical tails.

5.3.3 Wing Control Surfaces

Because of the varied lift requirements of this competition, flaperons were deemed essential. The most obvious reasons for flaperons are to slow landing speeds and shorten TOFL during the heavy "Passenger Delivery" mission segment. Flaperons could also be mixed with elevator input to augment pitch up authority. Lastly, flaperons could be spoilerons and dump lift or slow the aircraft after touch down.

From historical data, flaps are typically sized at 20% of the chord and extend less than 50% of semi-span. In this design, flaperons were set at 20% chord and 60% of semi-span. The flaperons were purposely extended into the twisted region of the wing to assist in roll control in the event of stall caused by flaperon deflection.

5.4 Structural Design

5.4.1 V-n Diagram

The minimum and maximum load factors, n_{neg} and n_{pos}, express the maneuvering ability of an aircraft as a multiple of the acceleration due to gravity. Based on the limit load factors of the aircraft while turning, the limit load factors were determined to be n_{pos} = 4.0 and n_{neg} = -1.0. The V-n diagram (Figure

5.3) depicts the limit load factors as a function of airspeed. The positive and negative lift limit curves were calculated using the following equation:

$$n = \frac{0.5\rho_{alt}V^2 S_W C_{L_{max}}}{W_{TO}} \tag{5.1}$$

5.4.2 Wing G Capability

The maximum 4G lift along the span was calculated using an elliptical lift approximation. This was possible since the wing was tapered and twisted to achieve a nearly elliptical lift distribution. Next, the distributed load of the wing was calculated along the span, refer to Figure 5.4. Using a trapezoidal approximation to calculate the area under the curve, the shear diagrams and moment diagrams were as in Figure 5.5 and Figure 5.6.

To handle the bending loads a carbon fiber wing spar tube extended from the root rib to 60% of semi-span. At the joint location in the fuselage, the same size carbon tube was used. A 6061-T6 .030" wall tube fit into the wing spar and fuselage carbon tubes and carried the bending loads between the three components. To carry bending loads past 60% of semi-span, two full span spruce spars ran from 30% of root chord to 30% of tip chord. The upper spruce spar was 3/8"x3/16" and the lower spruce spar 3/16"x3/16".

The wing G capability was then analyzed at two critical locations. The first location was at the wing root where there was the highest moment. The compressive stress was 17,500 psi, which provided a 1.9 safety factor from compressive yield at 34,000 psi. The second location was at 60% semi-span, where the carbon tube stopped and the spruce spars carried the bending. Using typical values of compression strength for spruce (3500 psi), the spruce/balsa section showed the spruce at 1150 psi, or a safety factor of 3.0 from yield.

5.5 Systems Architecture

The layout of motor, battery, receiver, control rods, servos, brakes, and payload in the aircraft are shown in the drawing package.

5.5.1 Servos

The minimum number of servos, including electronic speed controller, ESC, to have a functional aircraft was four - 2 aileron, 1 elevator, 1 rudder, and 1 ESC. Two functions which where deemed necessary were wheel braking and flaperons for camber changing. The elevator servo was used to make contact with the brake valve eliminating the need for an extra servo. Hitec HS-225 servos were chosen because of their high torque (56.3 oz-in), low weight (1.13 oz) and slim profile compared to standard servos.

5.5.2 Brakes

Over the counter pneumatic brakes were purchased from BVM. These brakes are typically used on RC jet aircraft and were placed only on the main gear to permit responsive nose gear steering while braking. The tank is pressurized to 100 psi before flight.

5.5.3 Motor/Battery Cooling

Motor and battery cooling is absolutely necessary at 30A continuous. Inlet cooling holes were incorporated in the front of the fuselage and an air path from the nose to the tail provides cooling for motor and batteries.

5.6 Performance

5.6.1 Takeoff

To check the worst case, the simulation program uses 1.1 V_{stall} with no flaperon deflection and no headwind to calculate a wheels up takeoff field length. For the 24-ball payload case, the TOFL is 194 ft and for the no payload case, the TOFL is 68 ft.

5.6.2 Climb Capability

The simulation program showed the maximum climb rate in the Passenger Delivery configuration was 9.64 ft/sec. (Figure 5.7) The minimum climb rate deemed acceptable was 5 ft/sec.

5.6.3 Mission Performance

The effect of a 20 mph headwind is included in the calculation by extending the actual length of the course by the relationship shown below. This method was used successfully by the 2000/2001 team and adds approximately 12% to the course length.

$$dist_{eff} = dist\left(\frac{1}{1-\left(\frac{V_{wind}}{V_{flight}}\right)}-1\right) \tag{5.2}$$

Refinements were made to the estimated thrust and the final mission performance is shown in Figures 5.7, 5.8 and Table 5.1. Figure 5.8 describes the points on the course for the Position mission segment.

5.7 Aircraft Weight and Balance

The weight and balance spreadsheet is shown in Table 5.2. The final aircraft weight was 25.9 lbs and the CG was located 33.3 inches aft of the motor bulkhead datum line. This position is 28% of MAC. The payload fraction was 37%.

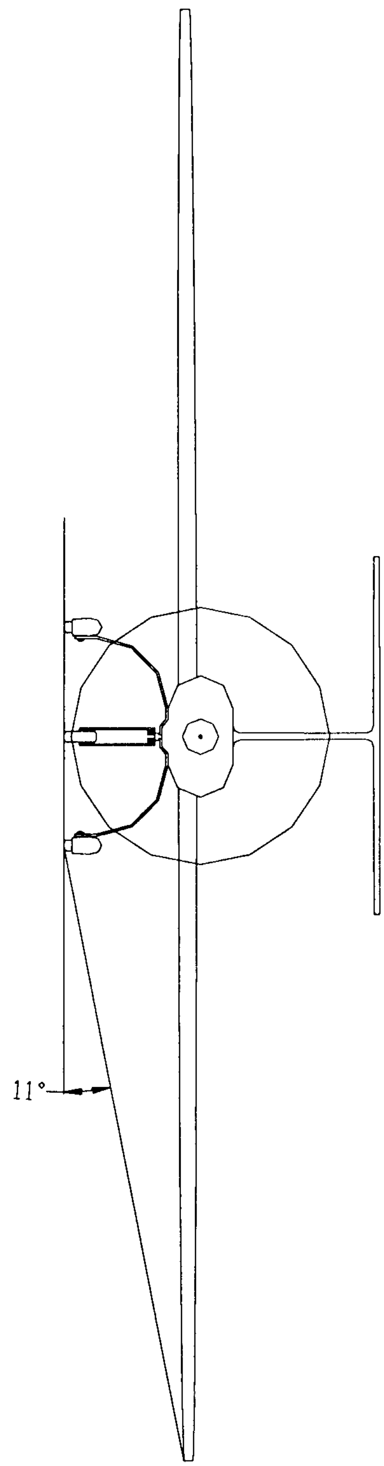
5.8 Rated Aircraft Cost Worksheet

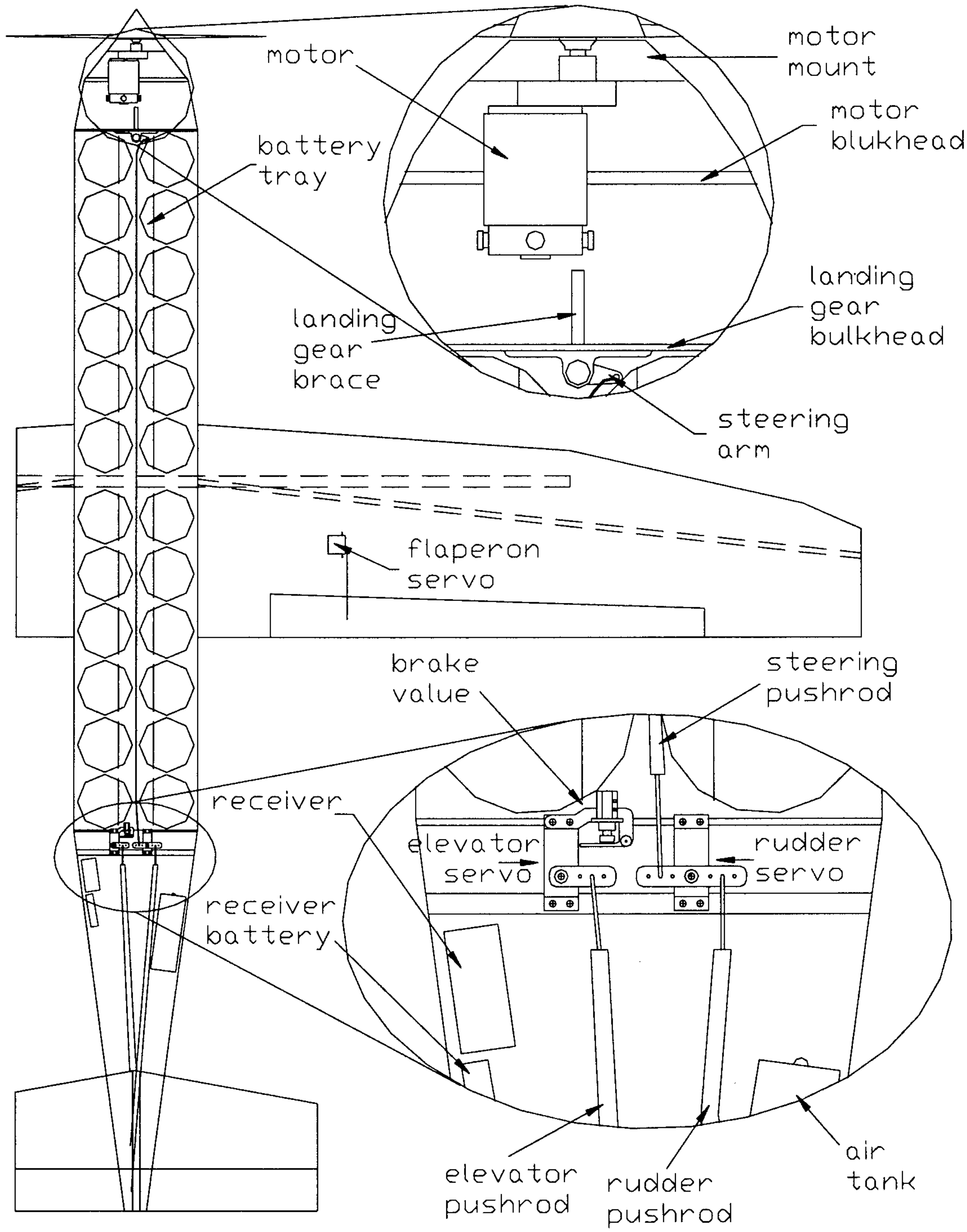
The rated aircraft cost was calculated using the parameters set forth in the competition rules. A breakdown of the RAC is shown in Table 5.3 And Figure 5.10. The RAC for phastball was calculated to be 11.56.

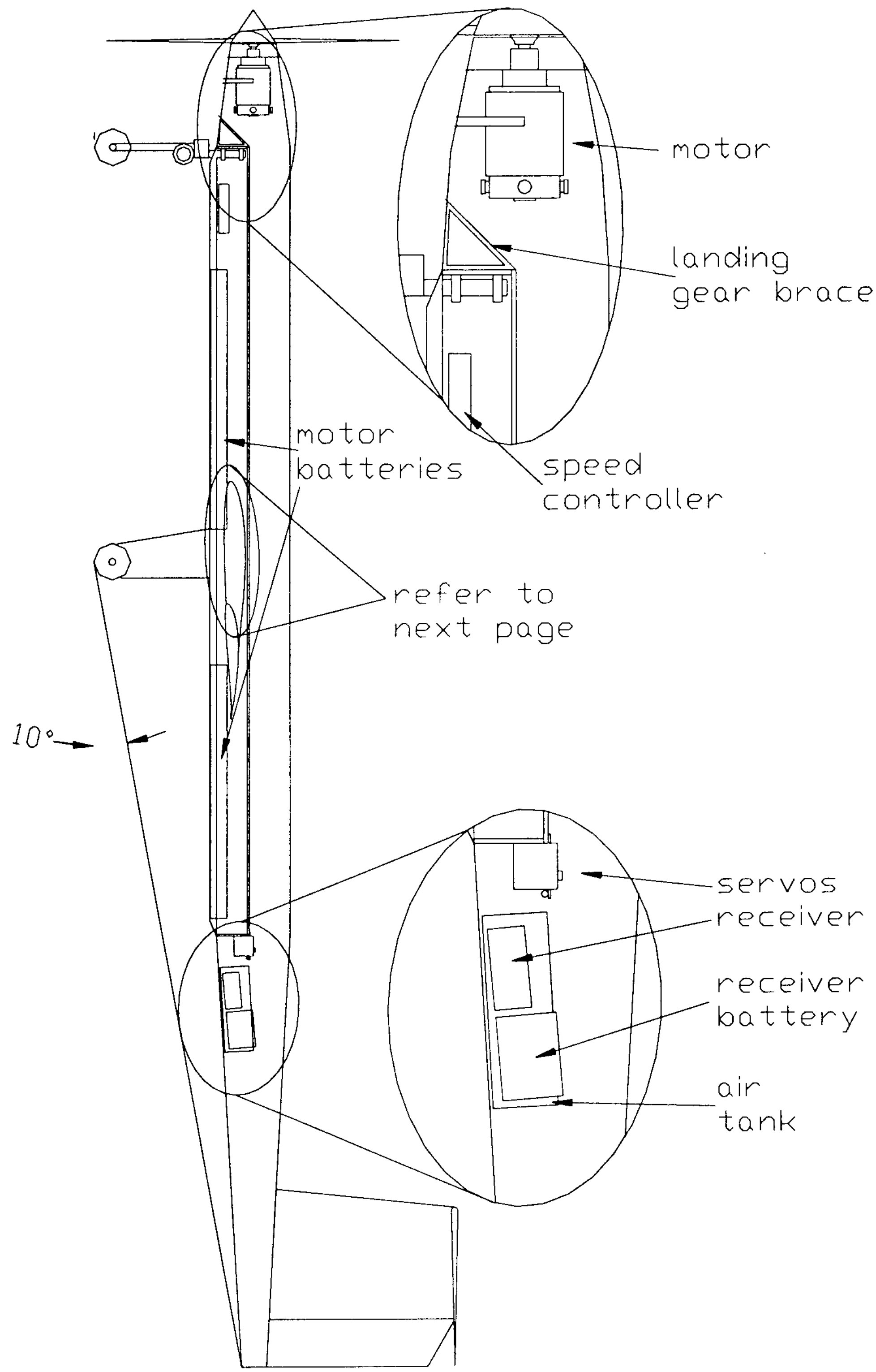
5.9 Final Configuration

The final aircraft configuration is listed below and is shown in detail in the drawing package.

Wing Airfoil Panel 1	*S7012*
Wing Airfoil Panel 2	*S7012 to SD7036*
Wing Airfoil Panel 3	*SD7036*
Wing Aspect Ratio	*8*
Panel 1 Span	*2.5 ft*
Panel 2 Span	*1.33 ft*
Panel 3 Span	*0.33 ft*
Root Chord	*1.24 ft*
Break 1 Chord	*1.07 ft*
Break 2 Chord	*0.77 ft*
Tip Chord	*0.60 ft*
Panel 1 Area	*2.89 ft^2*
Panel 2 Area	*1.23 ft^2*
Panel 3 Area	*0.23 ft^2*
Horizontal and Vertical Tail Airfoil	*NACA 0005*
Wing Span	*8.33 ft*
Wing Area	*8.68 ft^2*
Horizontal Tail Area	*1.48 ft^2*
Vertical Tail Area	*0.792 ft^2*
Aileron Area per Wing	*0.52 ft^2*
Wing Root Incidence	*1.9°*
Maximum Gross Weight	*25.9 lb*
Softball Capacity	*24*
Landing Gear Type	*tricycle*
Landing Gear Span	*1.33 ft*
Motor	*AstroFlight 661*
Gear Ratio	*2.75 to 1*
Propeller	*18 x 18*
Predicted RAC	*11.56*


11°


motor
motor mount
motor blukhead
battery tray
landing gear bulkhead
landing gear brace
steering arm
flaperon servo
brake value
steering pushrod
receiver
elevator servo
rudder servo
receiver battery
elevator pushrod
rudder pushrod
air tank


motor
landing
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speed
controller
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next page
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receiver
receiver
battery
air
tank

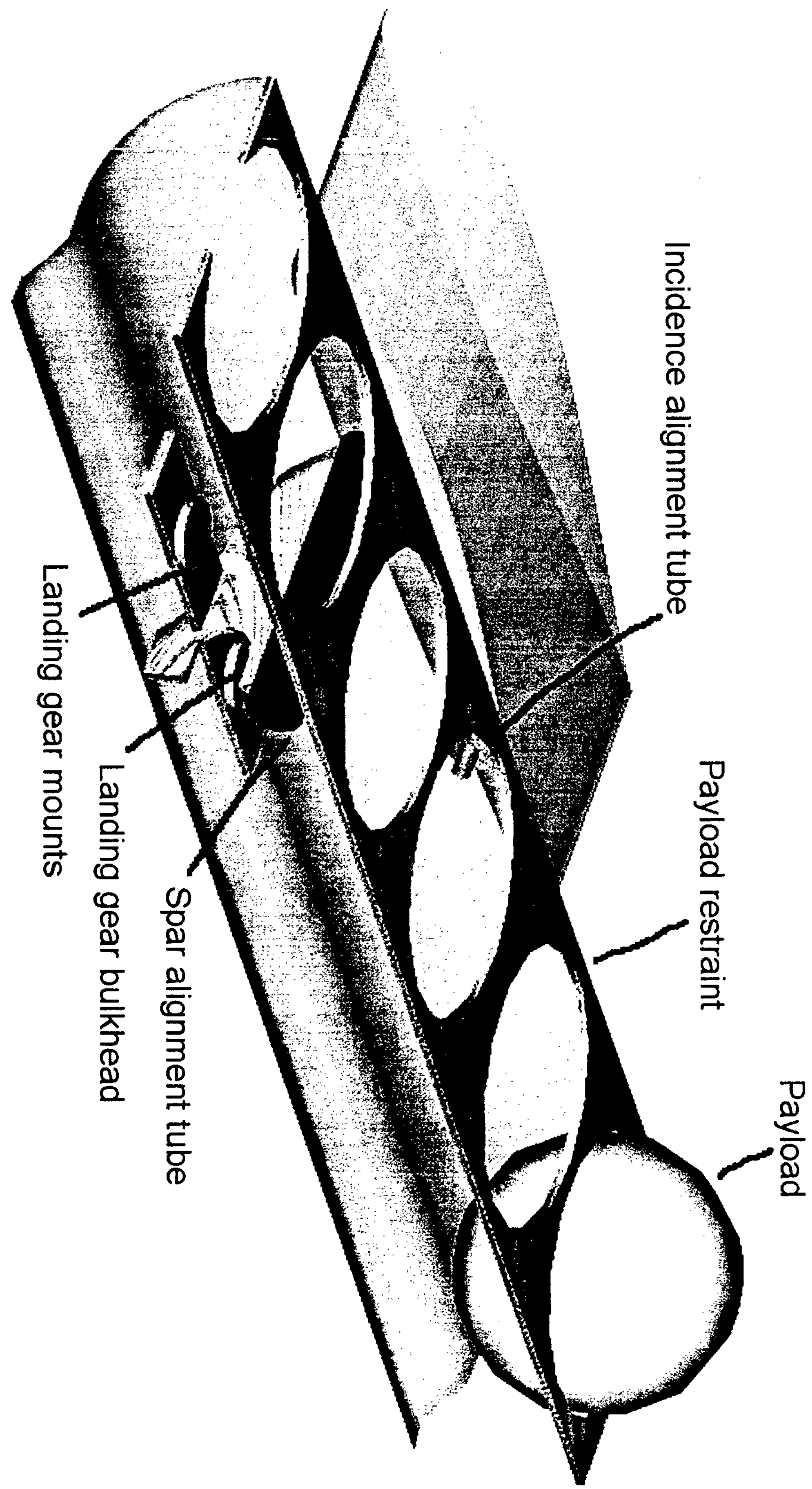
Incidence alignment tube
Payload restraint
Payload
Landing gear mounts
Landing gear bulkhead
Spar alignment tube

Table 5.1 Summary of Mission Performance

	Position	Passenger Delivery	Return
TOFL [ft]	68	194	68
Flight Time [sec]	73.9	89.1	62.9
Motor Run Time [sec]	60.8	74.6	49.8
Pit Time [sec]	40	40	0
Max Speed [ft/sec]	112	108	112
Max Turn Radius [ft]	101	152	101
Total Mission Time [sec]	266		
Total Mission Time [min]	4.4		

Table 5.2 Weight and Balance Spreadsheet

Component	Dist. [in]	Wt [oz.]	Wt [lbs]	% of MTOGW	Moment [oz-in]
Motor	3.0	26.8	1.7	6%	80
ESC	7.3	2.0	0.1	0.5%	14
Prop	-0.5	5.9	0.4	1%	-3
spinner	-0.5	2.9	0.2	1%	-1.5
nose gear	9.1	4.4	0.3	1%	40
main gear	34.5	13.1	0.8	3%	451
tail servos	57.8	3.0	0.2	1%	173
RX	61.0	1.2	0.1	0.3%	70
RX Battery	61.0	3.4	0.2	1%	206
air tank	61.4	1.2	0.1	0.3%	75
horizontal tail	77.5	4.2	0.3	1%	322
vertical tail	75.5	4.9	0.3	1%	372
wing	35.0	64.0	4.0	15%	2240
fuse	37.3	64.0	4.0	15%	2384
battery 1	41.0	30.0	1.9	7%	1230
battery 2	34.5	30.0	1.9	7%	1035
MEW	33.3	260.9	16.3	63%	8689
Payload	33.3	153.6	9.6	37%	5115
MTOGW	33.3	414.5	25.9	100%	13804

Table 5.3 RAC Worksheet

Description		Aircraft Parameter	Value ($ thousands)
MEW Manufacturers Empty Weight	aircraft empty weight	16.3lbs	
			1.63
REP Rated Engine Power	# of motors	1	
	total battery weight	3.86lbs	
			5.78
MFHR Manufacturing Man Hours	wing span	8.33ft	1.33
	max exposed wing chord	1.22ft	0.20
	# of control surfaces	2	0.12
	length of fuselage	7ft	1.40
	# of vertical surfaces	0	0.00
	# of vertical surfaces with control surfaces	1	0.20
	# of horizontal surfaces	1	0.20
	# of servos or motor controllers	5	0.50
	# of motors	1	0.10
	# of propellers of fans	1	0.10
		total RAC:	***11.56***

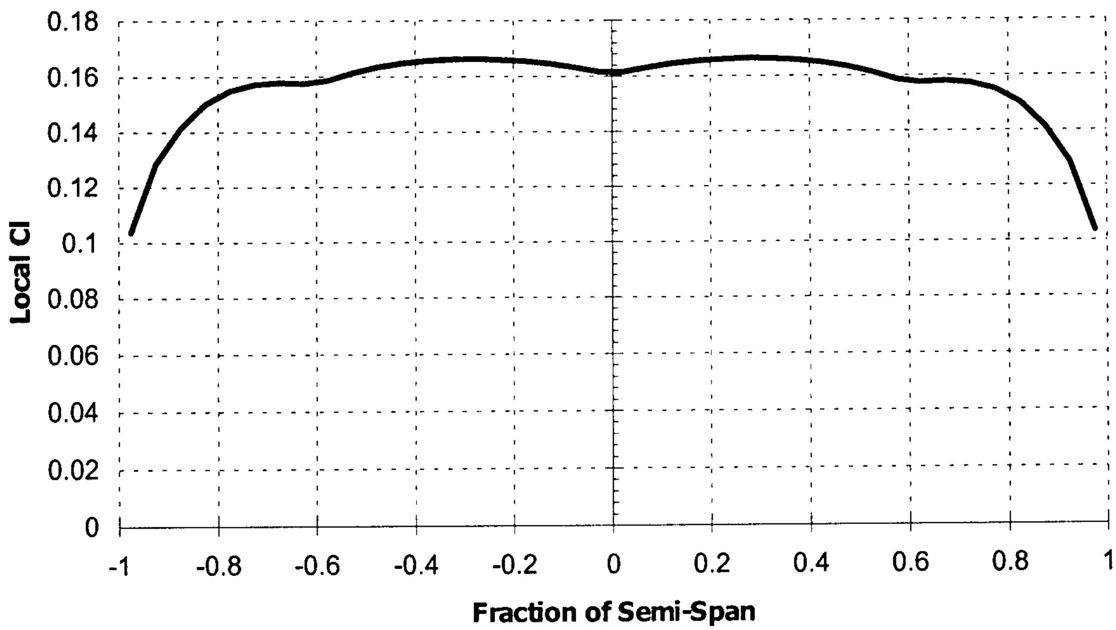

Figure 5.1 Span wise Cl distribution for final wing planform

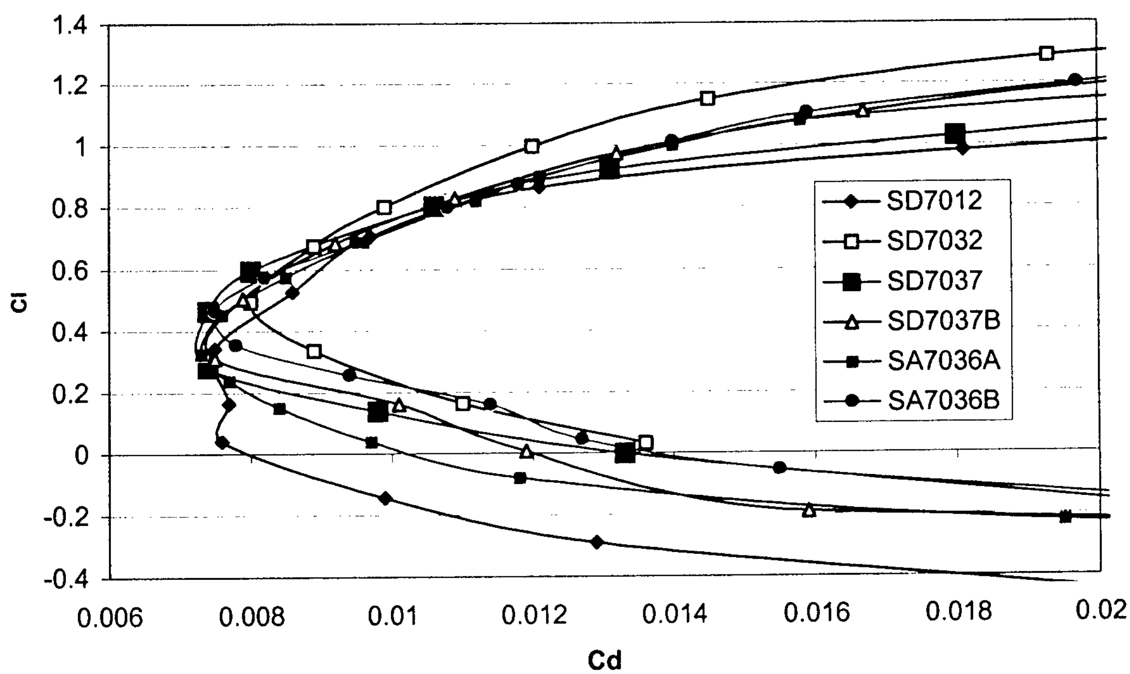

Figure 5.2 Airfoils evaluated for wing tip at Re=300K

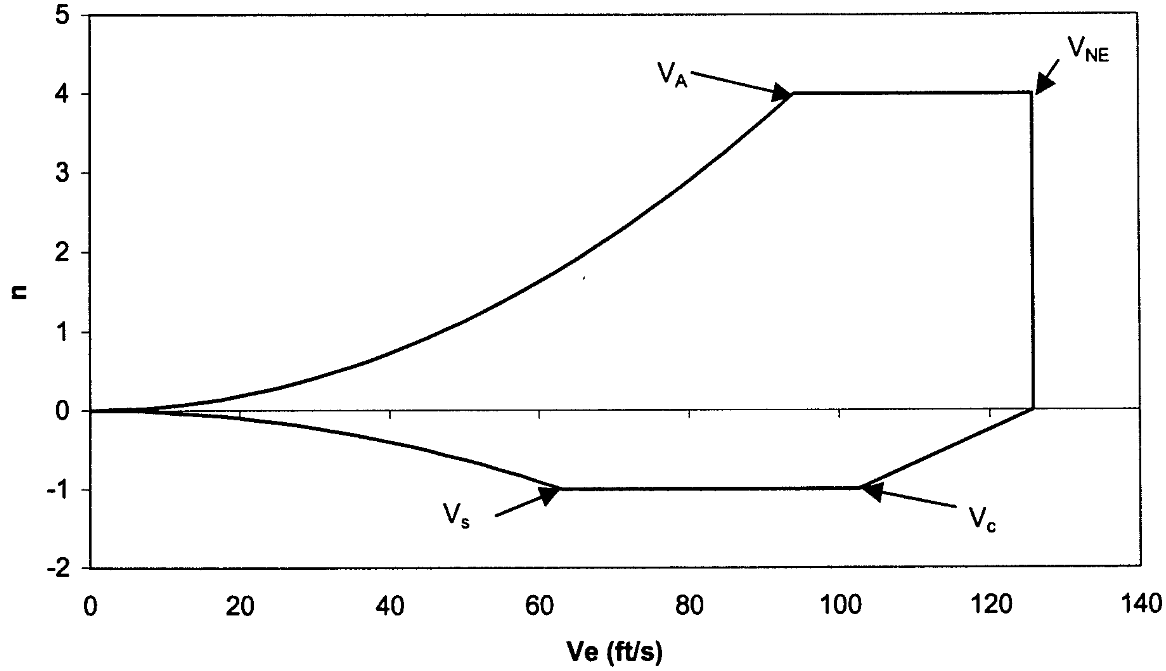

Figure 5.3 V-n diagram – Defines the flight speed envelope which must be adhered to for ensured structural integrity

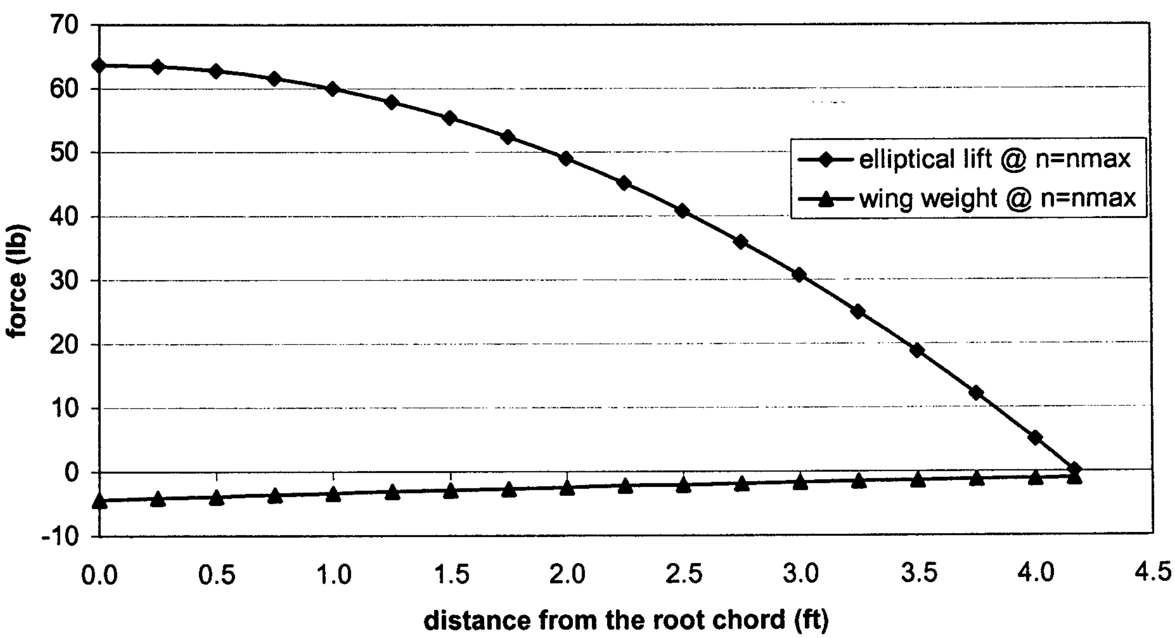

Figure 5.4 Flight load diagram due to lift and weight at load factor $n=n_{max}$

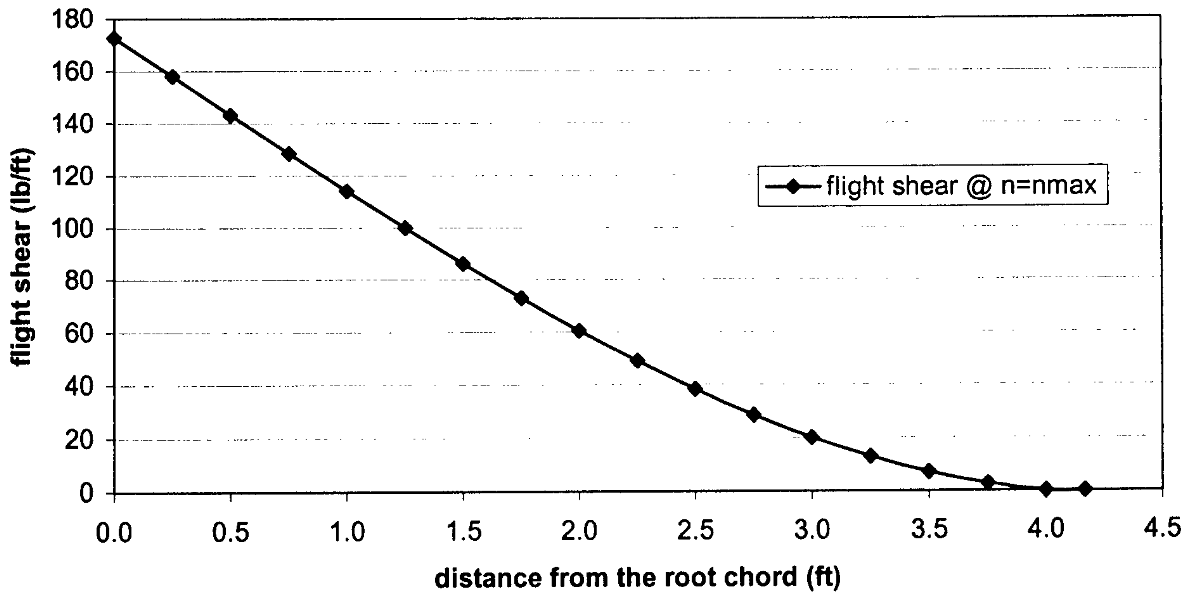

Figure 5.5 Flight shear diagram due to lift and weight at load factor n=n_{max}

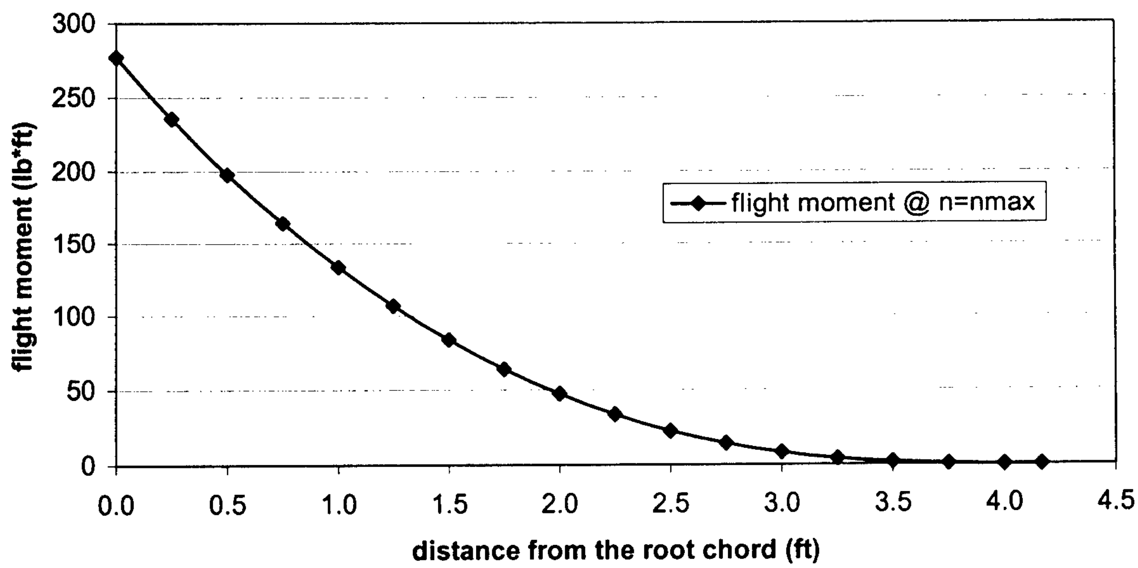

Figure 5.6 Flight bending moment diagram due to lift and weight at load factor n=n_{max}

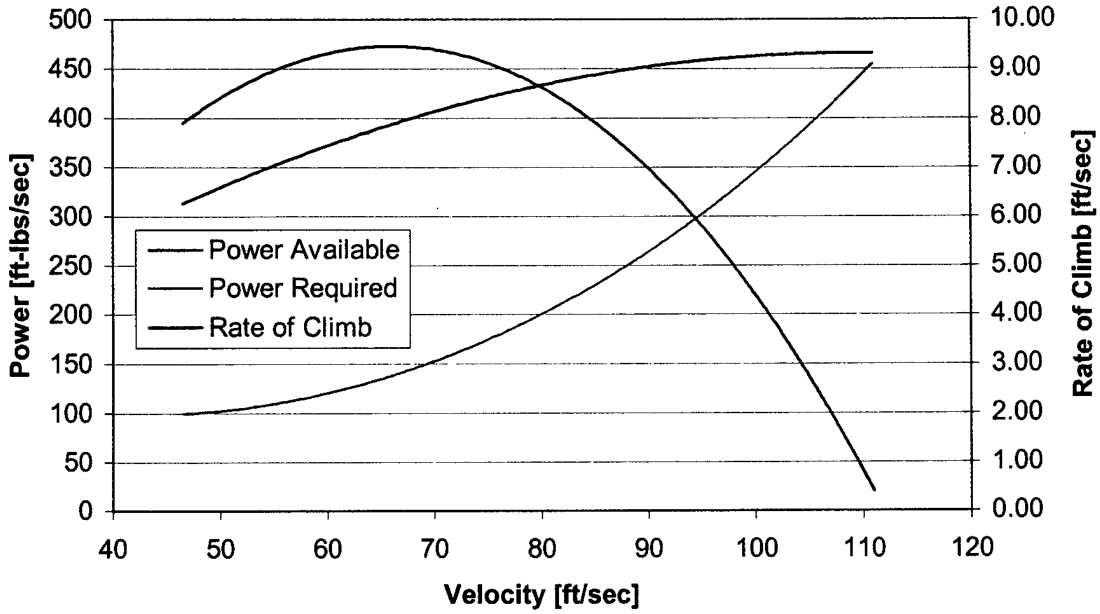

Figure 5.7 Power and Climb Performance in Passenger Delivery Configuration

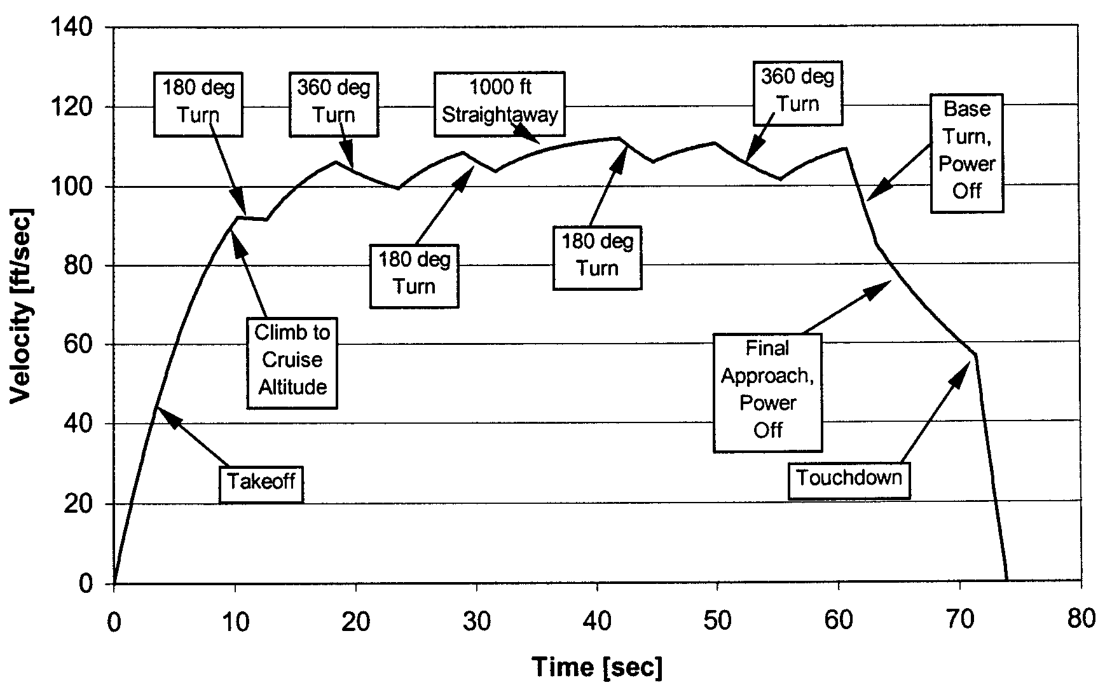

Figure 5.8 Velocity Profile during Position Mission Segment

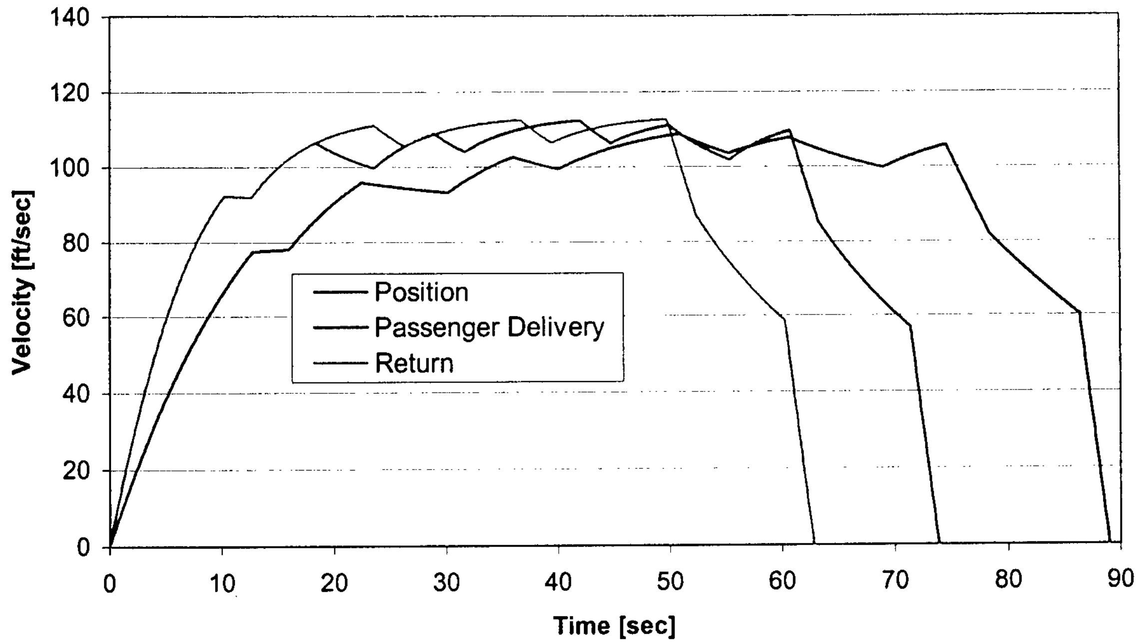

Figure 5.9 Velocity Profiles for all mission segments

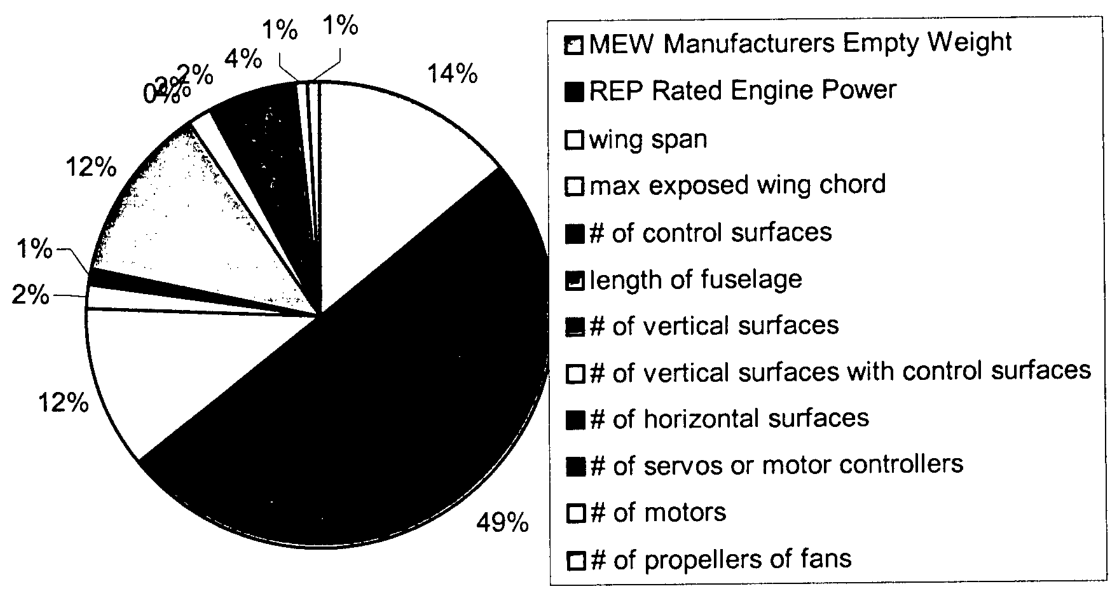

Figure 5.10 Breakdown of Rated Aircraft Cost

6.0 Manufacturing Plan

6.1 Manufacturing Processes Investigated

The fabrication and assembly of the aircraft components needed to be done quickly and within the limitations of the teams skill and available equipment at the University. During the construction of the aircraft, time was a nemesis to help keep the team on pace to finish the construction in time for the competition a manufacturing milestone schedule was created, refer to Table 6.1. To help manage the time requirements of the project while attaining the highest production and results, the manufacturing plan was evaluated using the following FOM shown in Table 6.1.

6.2 Figures of Merit Breakdown

The purpose of the figures of merit is essentially to help determine the pros and cons of each construction phase of the project. Using this process to guide the team in construction alternatives helped maximize production and efficiency. The figures of merit that were considered to be most influential in the final design were simplicity, time required, cost, strength to weight, and availability.

Simplicity

It was determined that there had to be some limitations as to the overall complexity of the aircraft to make it both producible and maintainable. If the construction of a single part was easily done, it was given a 5. If it was tedious or took skills not available by team members, it was given a 1.

Time required

Time was a large contributing factor as to the feasibility of construction for the aircraft. In order to complete construction on all the components of the aircraft, the team had little forgiveness in missed deadlines. For each process, the time required to produce the part was compared to the overall advantage of making the part. If a part was time consuming, it was given a 1, if it could be completed quickly, it was given a 5.

Cost

The cost of the project was a very significant parameter that was non-negotiable. An estimation of materials needed and salvageable materials from past projects was done from conception to minimize new purchases and thus overall aircraft cost. With a limited budget, the selected parts and materials were first compared on a pro-con basis for optimal results. Parts and materials with a low cost or donations were given a 5, high cost parts and materials were given a 1.

Weight

As in all aircraft, weight is a critical factor. It was the intention of this year's team to build a lightweight aircraft with a low profile. The weight consideration of this year's plane was one of the most critical aspects and most all decisions were based from it. Bulky and heavy parts and materials were given a 1, whereas light and slim parts and materials were given a 5.

Availability

The availability of materials needed to complete the manufacturing process was an important concern. If the material was donated or easily obtained, it was given a 1. If the material was expensive or needed special ordering and obtaining procedures, it was given a 5.

6.3 Process Selected for Major Component Manufacture

6.3.1 Fuselage

Hand lay-up of composite materials was used to construct the fuselage. The construction of the fuselage began by first producing a male plug that was identical to the final fuselage shape. The plug was constructed out of foam that was cut to shape using a hot wire. Once the foam was cut it was then covered with three layers of fiberglass. The fiberglass gave strength to the plug and was able to be sanded to a smooth finish. Several coats of body filler and light sanding were applied to achieve the desired finish. Next, a female mold was produced from the plug. The mold was made in two halves. Construction of the mold was done by first building a dam around the perimeter of the plug, then applying a release agent to the parts. Next, a 1/8" coating of surface coat was applied to the plug half and surrounding dam. Twenty layers of 6oz fiberglass were then applied to strengthen the mold.

Once the mold was constructed, the fuselage shells were constructed. A release agent was applied to each mold half before the lay-up process began. Several composite shells were manufactured to determine the final laminate to be used. These shells were comprised of combinations of fiberglass, Kevlar, and carbon fiber. Using this approach allowed the team members to develop the proper techniques for manufacturing composite components and produced spare fuselages. The laminate decided upon for the final fuselage was one layer of 3k x 1k carbon fiber for the lower half and two layers of 1.9oz Kevlar for the upper shell. Additional layers of material were added in strategic areas such as around the motor mount and landing gear mount. Plywood bulkheads were then added in areas of importance as well. The payload holder was fabricated out of 1/8" plywood. 3.75" holes were drilled so that the softballs could sit in the fuselage. Figure 6.1 is a picture of the construction of the fuselage.

6.3.2 Wing and Tail

The wing and tail were constructed out of 1lb density Styrofoam that was then sheeted with 1/16" balsa wood. First, blocks of Styrofoam were cut to the wing planform. Next, the spar and tube holes were cut into the entire wing. The wing was then cut into sections corresponding to the breaks in the wing planform. Next, templates were made for each section. The airfoil shape was then cut of each section using a hot wire. Each section was then glued together with epoxy and the entire wing was sanded smooth. Next, the spars and alignment tubes were glued into place. Then 4" x 48" x 1/16" sheets of balsa wood were then glued together with CA glue to form large panels. These panels were then adhered to the foam core using Elmer's Pro Bond Polyurethane Adhesive. The wings panels were then trimmed to match that of the foam core and leading edge stock was applied. Next the wings were sanded smooth. Flaperons were then cut out of the wing and the hinges were installed and the flaperons were

reattached. The wings were then covered in monocote. Figure 6.2 is a picture of one of the wing panels before it was covered with monocote. Horizontal and vertical tail surfaces followed the same procedure.

6.3.3 Landing Gear

The landing gear was constructed out of carbon fiber. Previous year's entries were used as a starting point to determine the laminate needed to withstand landing. Several of the past WVU entries used a carbon landing gear. Their weights and number of layers were examined and scaled accordingly to this year's aircraft. Using this approach allowed the team to spend more time on the construction phase of the landing gear rather than attempting to design the landing gear using FEM. The landing gear was constructed using the following laminate $[\pm 45/0_2/\pm 45/0_4/\pm 45]_S$. The mold for the landing gear was constructed by cutting the shape out of foam, which was then laminated with a thin layer of Mylar. Hand lay-up and vacuum bagging techniques were then used to produce the final part.

6.3.4 Payload Restraint

Payload is restrained vertically when the fuse hatch is installed on the aircraft. Balls are restrained laterally and longitudinally with plywood structure built into the lower fuse half with hole diameters equal to the ball diameter. Refer to drawing package.

6.3.5 Other Components

Other aircraft components such as bulkheads, servo mounts, landing gear mounts, etc. were constructed out of plywood. Figure 6.4 is a picture of the servos used to control the rudder, elevator, steering, and brakes.

Table 6.1 Manufacturing milestones

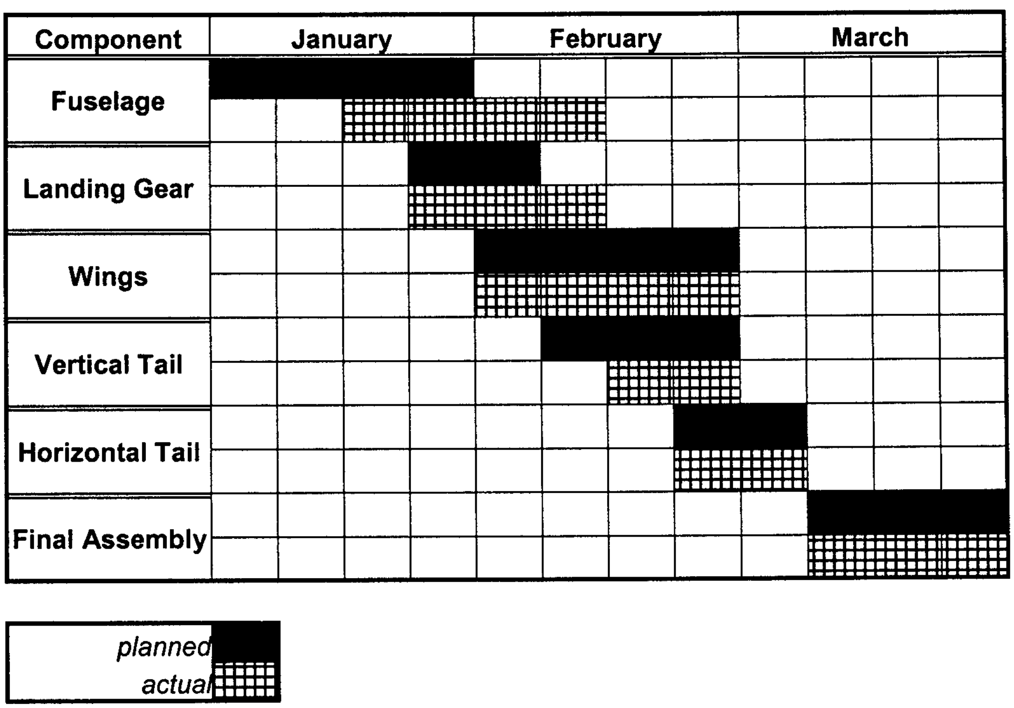

Table 6.2 Figure of Merit breakdown of manufacturing processes

		Simplicity (x5)	Cost (x1)	Strength to Weight (x6)	Time Required (x1)	Availability (x5)	Results
Wings and Empenage	Build up (balsa wood ribs, hardwood spars)	3	4	3	2	4	59
	Foam core with balsa wood sheeting	4	4	4	5	4	73
	All Composite	2	1	5	2	2	53
Tail	Build up (balsa wood ribs, hardwood spars)	3	5	4	3	4	67
	Foam core with balsa wood sheeting	4	4	4	5	4	73
	All Composite	2	1	5	2	2	53
Fuselage	Build up (balsa ribs, hardwood stringers)	3	3	3	3	4	59
	All Composite	2	1	6	2	4	69
Landing Gear	Purchase	5	5	3	5	2	63
	Aluminum	3	3	3	3	4	59
	All Composite	4	3	5	3	4	76

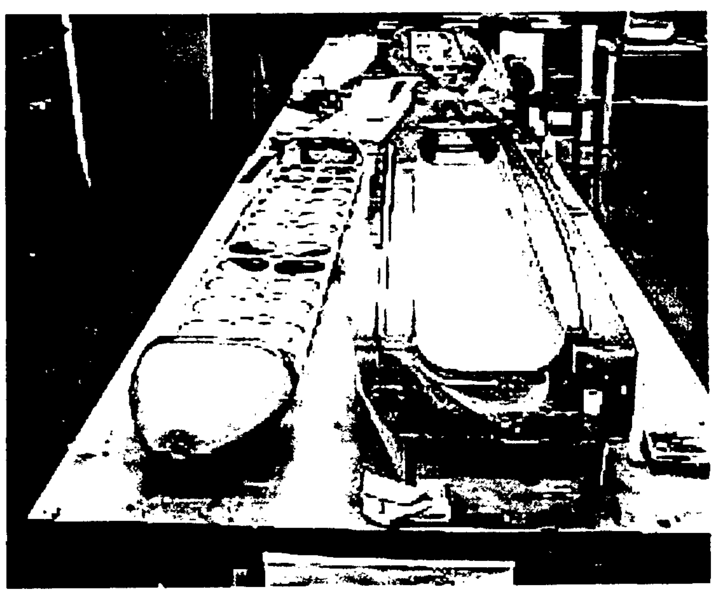

Figure 6.1 Picture of the construction of the fuselage

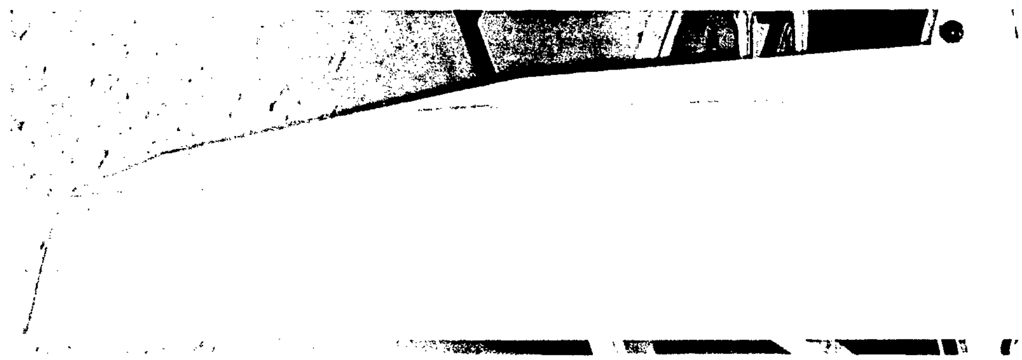

Figure 6.2 Foam wing sheeted with balsa

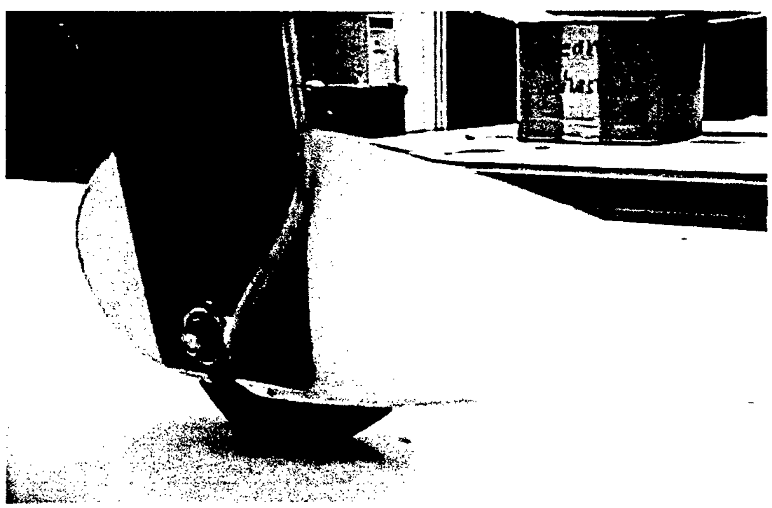

Figure 6.3 Carbon fiber landing gear with brake, wheel, and wheel pant

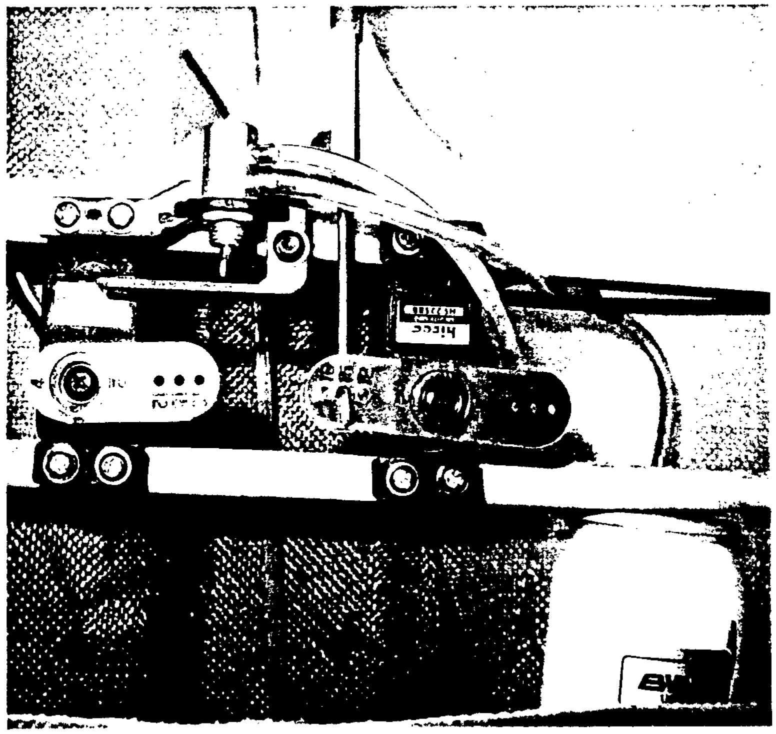

Figure 6.4 Rudder, elevator, steering, and brake servos

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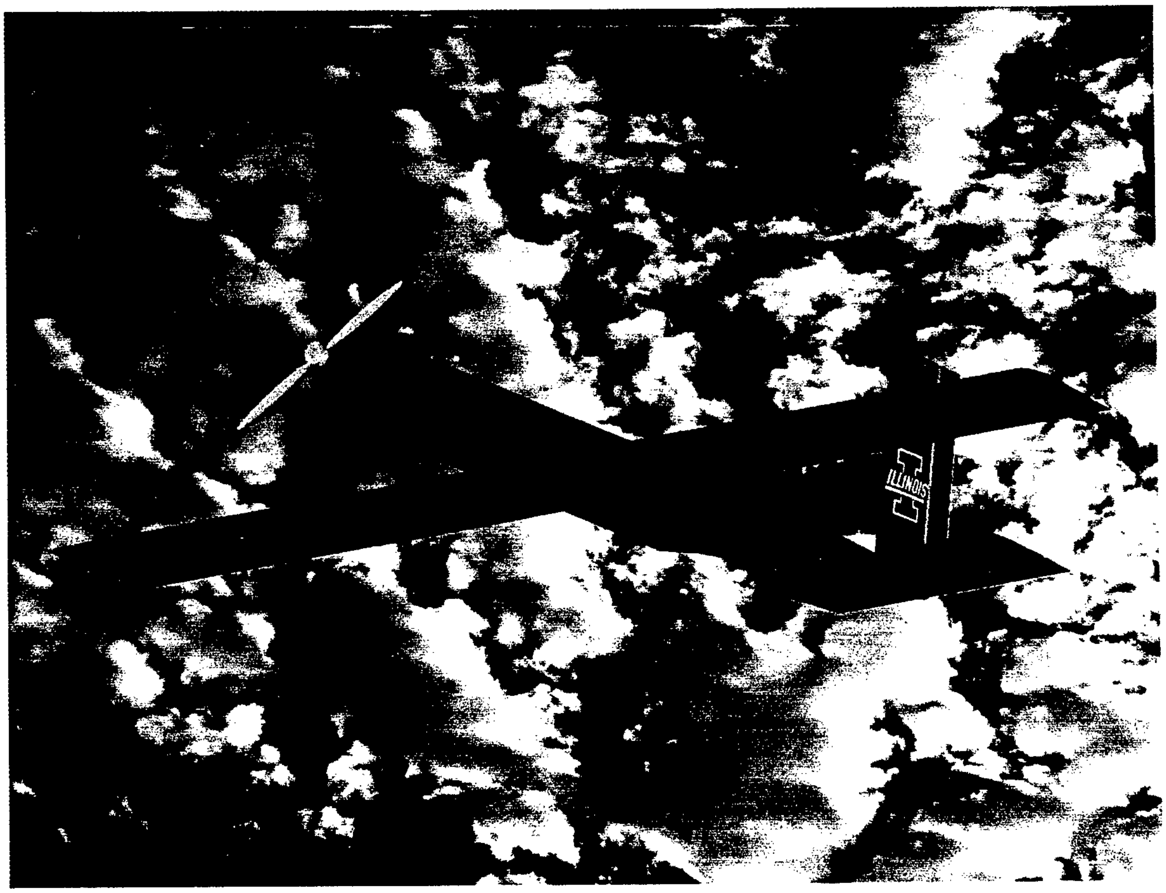
Illiniwek I

Final Design Report



AIAA Student Design/Build/Fly Competition

Cessna Field Flight Test Facility
Wichita, Kansas

Submitted by

Department of Aeronautical and Astronautical Engineering
University of Illinois at Urbana-Champaign

on

March 12, 2002

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Acknowledgements

Several sources contributed to the success of this year's team and we would like to acknowledge their help and express our appreciation. We thank Jim Schmidt and Hobbico, Inc. for their continued and incomparable support through their generous donation of building materials, tools, supplies, and pilots. Thanks goes to Mike and AnnMarie Cross for their donation of time and their skills. We could not be continually successful without these great pilots. We would also like to thank Professor Michael Bragg and the Department of Aeronautical and Astronautical Engineering for providing space for a workshop, the use of a low-speed wind tunnel for testing, and monetary support. We express thanks to the College of Engineering at the University of Illinois at Urbana-Champaign, the Illinois section of the AIAA, and Region III for their continued financial support. We especially thank James Kessler, an alumnus of the University, for his continued support of the team. We are indebted to our advisors for their encouragement, suggestions, and assistance in operating smoothly. We thank Jason Merret for taking the time to assist different subgroups in their work and for helping to modify the performance code that he originally wrote years ago. We would like to thank Dr. Andy Broeren for also assisting the different subgroup heads in their work and for supervising our wind tunnel testing. He was instrumental in reserving us a tunnel entry that provided much needed propulsion data. Finally, we would like to thank Professor Kenneth Sivier for his continued advisement of the team. Our deepest gratitude goes to him for his continued guidance and endless encouragement.

Table of Contents

1.0 Executive Summary

This report presents the design process and results for the University of Illinois' entry into the sixth annual AIAA Student Design/Build/Fly Competition. The final design, *Illiniwek I*, is the result of detailed aerodynamic, performance, and structural analysis. The processes discussed in this report are based on experimental data, acquired through structural and wind tunnel testing, and analytical and numerical modeling. The aircraft is designed to satisfy all of the competition requirements as well as minimizing mission time, maximizing payload carried, and minimizing Rated Aircraft Cost.

The design process began with a consideration of several different configurations. Using mission requirements and a cost model, these configurations were narrowed to a flying wing, a canard, and a conventional configuration. The flying wing was eliminated using a Figure of Merit system. The results of the conceptual design process indicated that the performance of the canard and conventional configurations were nearly the same. After several team deliberations, the canard configuration was eliminated for aerodynamic reasons. It was found that the lift on the canard would have to be high for the trim condition. In this case, the canard would have to be designed with heavier structure than the tail of the conventional design. The conventional design was chosen because it could be trimmed with less tail-loading. This configuration also offered the team more predictable flight qualities and the design techniques involved were well known.

Structural and wind tunnel testing were the focus of the preliminary design phase. A performance code was also used to optimize the configuration. The loads the wing configuration would experience were modeled and simulated for structures testing. The goal was to find what materials and structural system would provide a strong and light airplane. Propeller and battery combinations were tested in a wind tunnel. After baseline data were collected, simulated contest missions were performed. It was determined that the total mission would use approximately 1500 mAh of energy compared to the 2400 mAh theoretically available from the proposed battery pack. The performance code was used to optimize wing geometry by minimizing the quantity of Mission Time*Rated Aircraft Cost.

The detail design consisted of choosing the airfoil, sizing the control surfaces, choosing a propeller, selecting building materials, and determining how components would be built. During this phase, a final estimation of overall mission performance was also determined.

Illiniwek I satisfies all of the requirements put forth in the 2001-2002 Rules and Vehicle Design Specifications. The aircraft was designed for maximum payload capacity of 24 softballs. The team expects to complete one mission consisting of three sorties in a time of approximately six minutes.

The University of Illinois at Urbana-Champaign is proud to submit this design to the sponsors of the AIAA Design/Build/Fly Competition.

2.0 Management Summary

The organization of the design team was influenced by the architecture of previous teams and the systems engineering design philosophy taught at the University of Illinois. A core group met during the summer to discuss the new contest rules and to establish a test matrix for propulsion data. Basic configurations were also discussed and recorded. It was decided not to begin the design phase over the summer. That allowed the team to concentrate on propulsion research during that time. It also ensured that the incoming students were present for the entire design process, which built a foundation for future years.

When the fall semester began, it was decided by the returning members that the focus would be on design and its process. The design areas that the team wished to investigate were established and subgroups were created accordingly. Returning members volunteered to head the subgroups and new members were assigned to these subgroups based on previous experience, coursework, and class. Subgroups and their respective participants are given in Table 2.1. It should be noted that many team members worked in more than one area. This enhanced team communication and allowed the new members to develop a broader knowledge base for future teams.

The project head was responsible for organizing meetings, procuring materials, and tending to administrative concerns. The assistant project head aided in the administrative aspects, but mainly focused on leading the testing of new building techniques and the developing the team's design process. The project heads worked closely with the advisors to ensure a smooth design process and to make sure each subgroup remained productive.

Each week the entire team met to update the members on work that had been accomplished during the week. Concerns were also voiced at this time. Another weekly meeting was held with only the project heads, advisors, and subgroup heads. This meeting focused on major policy decisions and changes. The subgroup heads controlled the subgroup's schedule and planned the subgroup's meetings. Additional meetings were held on the weekends to accomplish manufacturing and, later, flight testing.

A project milestone chart is shown in Figure 2.1. The chart was established in order to adhere the team to a basic schedule. The chart set reasonable goals such as finishing the preliminary design before winter final exams and beginning construction after the semester break. Some deviation from this schedule occurred due to limited participation, but also due to input from team advisors and suggestions from subgroup heads.

Table 2.1: University of Illinois Design/Build/Fly Participants

Project Head	Michael Rynne
Assistant Project Head	Chris LaMarre
Faculty Advisor	Prof. Kenneth Sivier
Staff Advisor	Dr. Andy Broeren
Graduate Advisor	Jason Merret
Pilot / Advisor	Mike Cross
	AnnMarie Cross

Aerodynamics	Joe Zimmerman*	**Stability and Control**	Tracy Kidd*
	Jeff Lazzaro		Scot Campbell
	Justin Ford		Luke Vortman
	Sung Hyun Kim		
	Stephen Treharne		
Funding	Aaron Dufrene*	**Structures and Weights**	Carey Lunsford*
Manufacturing	Aaron Dufrene*	**Testing and Data Collection**	Chris LaMarre*
	Jeff Lazzaro*		Carey Lunsford
	Tom Krenzke		Joe Zimmerman
	Geoff Bower		Karl Klingebiel
			Michael Rynne
Performance	Ed Whalen*	**Writing and Editing**	Michael Rynne*
	Geoff Bower		Joe Zimmerman
Propulsion	Karl Klingebiel*		Chris LaMarre
	Tom Krenzke		Karl Klingebiel
			Tom Krenzke

* Denotes subgroup head

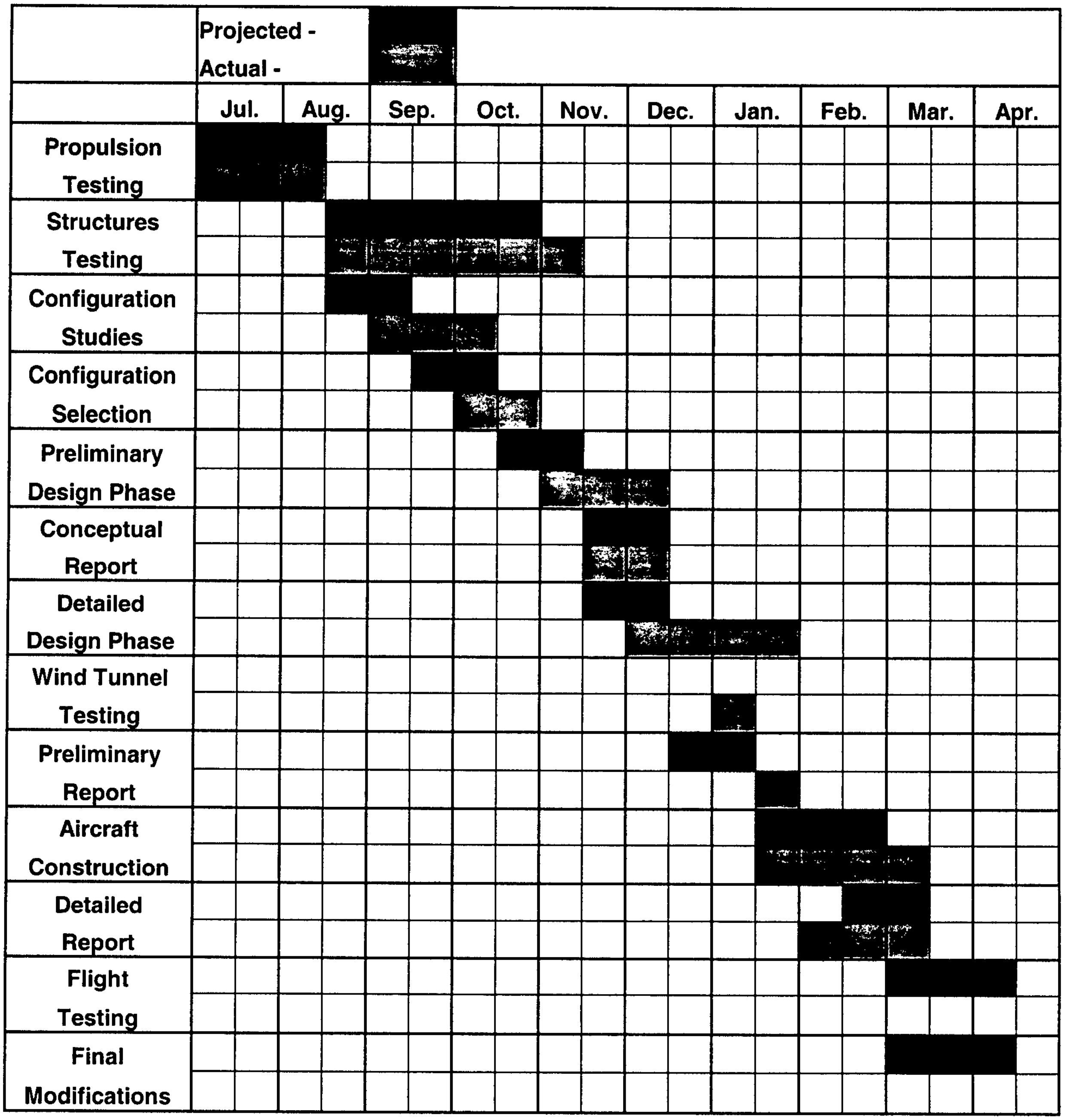


Figure 2.1: Milestone Chart

3.0 Conceptual Design

The goal of the conceptual design phase was to generate candidate aircraft configurations for the competition, and to select the best configuration for the preliminary design work. In order to meet this goal, a subgroup of four team members was formed, where each member had the initial responsibility of proposing two concepts for evaluation. From that point, this small group worked together to determine which of those concepts possessed the qualities necessary for an optimized aircraft and proposed their ideas to the rest of the team. The analysis and results of the conceptual design phase are summarized below.

3.1 System Requirements and Constraints

Prior to generating candidate configurations, a list of quantitative and qualitative design requirements was compiled. This list both constituted the proposed concepts and offered criterion by which the resulting configurations could be judged. The sources of this list of requirements were the Design/Build/Fly 2001-2002 Rules and Vehicle Design document and the experience of team members who had participated in previous contests. The design requirements were:

- Payload of 10 to 24 softballs
- Maximum of 5 lbs. of batteries
- 200 ft. takeoff distance limit
- Favorable flying qualities
- Quick payload changes
- Ease of manufacture

The first three of these requirements stem directly from the aircraft's required mission while the last three were based on prior experience. Favorable flying qualities and quick payload changing ability promoted a competition-worthy aircraft and increased ease of manufacture allowed for the production of the aircraft to move along at a quicker pace, leaving more time for flight testing and preparation for the contest.

In addition to these design requirements, the design objectives imposed by the competition were also recognized. These were low aircraft cost, high flight score, and low energy consumption. Throughout design work, these three areas were a major concern. During the conceptual design phase, candidate configurations were evaluated on both these design objectives and their qualitative performance with respect to the mission design requirements.

3.2 Assumptions

In order to make comparisons among the proposed concepts, it was necessary to develop a set of assumptions that would affect the analysis of the concepts. These assumptions pertain to the weight, size, and performance of the candidate configurations. The assumptions were:

-All candidate configurations must:

- carry the same 24-softball payload
- use the same electric motor

- have the same reference area

-In addition:

- fuselage weight and length were proportional at 1.2 lbs./ft. and landing gear was ignored
- wing weight was proportional to span at 0.35 lbs./ft^2.
- battery weight was constant at 5 lbs.
- aircraft mission flight time decreased as aircraft C_{D_0} decreased

Incorporation of these assumptions allowed each candidate configuration to be evaluated at a basic level. Some of these assumptions, however, require a brief explanation.

The payload of 24 softballs was selected based on its influence on the total score versus the flight time and the rated aircraft cost. The flight score equation was used to compare two aircraft, with one carrying half the payload of the other. The aircraft with half the payload would have to complete the course in half of the first aircraft's time, have half the rated cost of the first aircraft, or have a combination of a faster time and a lower cost. Because power is related to the cube of velocity, as the aircraft achieves a higher speed, the power required increases rapidly. Thus, it is unlikely that an aircraft carrying a smaller payload could achieve a fast enough flight time to counter the detriment in score due to decreased payload. This can be seen easily through investigation of the flight score equation:

$$\text{Flight Score} = (\#\text{ Laps} \times \#\text{ Softballs}) \times (\text{Total Mission Time})^{-1}$$

A common propulsion system was assumed anticipating that all of the aircraft would be of comparable weight and size. The characteristics of the AstroFlight Cobalt 60 were used because the team already had two of these motors and had collected a large amount of data on them for use in last year's Design/Build/Fly entry. At this stage, actual motor performance was not an issue.

The assumption of mission flight time decreasing with decrease in C_{D0} is a first-order approximation. Considering that the estimated drag of each configuration would be based on the same reference area, the direct comparison of parasite drag is possible. In general, an aircraft possessing lower parasite drag will be able to fly faster and be more energy efficient than an aircraft of higher drag with the same power and payload weight. The values given for C_{D0} were estimated using the empirical subsonic parasite drag equations given in Raymer[1].

Once these assumptions were made, the geometric details of each candidate configuration were developed. Based on these details, the stated design requirements, and the given assumptions, a down-selection process was carried out.

3.3 Configuration Down-Selection Process

3.3.1 First-Level Configuration Down-Selection via Weight and Parasite Drag

Initially, eight conceptual configurations were examined. Each of these configurations is described in Table 3.3.1. The primary differences in these concepts are wing configuration, tail configuration, and fuselage geometry. At this stage, each concept was evaluated based on weight and C_{D0}. From this analysis, the lowest estimated weight and parasite drag was achieved by the flying-wing concept, due to its simple and compact geometry. The conventional and canard configurations

possessed relatively low drag, while the wide-body configurations possessed relatively low estimated weight. Given the results of this first analysis, the dual-fuselage concepts were removed, while the remaining concepts were refined for the next level of down-selection.

3.3.2 Second-Level Configuration Down-Selection via Rated Cost

The six configurations remaining after the first down-selection were the conventional, the two wide-bodied conventional, the canard, the flying-wing, and the blended-wing-body configurations. However, it was decided that this list of candidates could be reworked into a more compact form. According to the aerodynamic and weight estimations being used, the canard and conventional concepts were virtually identical. Therefore, the canard and conventional concepts, with the two wide by 12 long (2Wx12L) payload arrangements were grouped together into one concept referred to as "conventional." In a similar manner, the two wide-body configurations were grouped together as an aircraft having a 3Wx8L payload configuration and referred to as "conventional wide-body". This resulted in the four concepts shown in Figure 3.3.1.

Because the drag of these configurations had already been estimated in the previous down-selection, aircraft rated cost was investigated during the second-level down-selection process. As detailed in Table 3.3.2, the best cost was achieved with the blended-wing-body, while the conventional concept with a 2Wx12L payload configuration had the highest rated cost. This makes sense considering that rated aircraft cost increases as length is added to the aircraft. Given this comparison, the tailless configurations showed the greatest potential because they possessed both low drag and low cost. The conventional/canard configuration with a 2Wx12L also appeared to be a good choice because it possessed a low parasite drag coefficient, despite its 23% increase in rated aircraft cost. Given the results of the second-level analysis, the wide-body configuration was dropped due to having the highest values in parasite drag and a relatively high value in rated cost. The configurations that remained at this stage were the conventional and canard concepts with 2Wx12L payload configurations, the flying-wing, and the blended-wing-body. In the next level of down-selection, these configurations were investigated using a Figure of Merit (FOM) system.

3.3.3 Third-Level Down-Selection via Figures-of-Merit (FOMs)

At this level of the down-selection process, the group of remaining conceptual configurations was again refined. The conventional/canard concept was now split into two separate concepts while the blended-wing-body and flying-wing concepts were combined into one concept labeled "flying-wing." These two changes were made due to the qualitative nature of using a Figure of Merit-based numerical rating system. So far in the down-selection process, the flying-wing and blended-wing-body had exhibited similar behavior because they possessed similar qualities. The canard and conventional configurations had already been grouped together for a similar reason. However, this analysis level called for the canard and conventional concepts to be investigated separately. Therefore, the three general concepts compared in this level of down-selection were the conventional, canard, and flying wing.

The Figure-of-Merit system used at this stage was based on the mission requirements, the design objectives, and results from previous levels of the down-selection process. The six FOMs used were:

- Manufacturability: The ease of manufacturing a particular structural design
- Ground Operations: The work required in loading/unloading the payload and ground handling
- Performance: The qualitative aerodynamic performance
- Cost: The rated aircraft cost (RAC)
- Durability: The robustness of a concept's structure with respect to aircraft loads
- Stability: The qualitative performance with respect to stability and control

The rating system applied to these FOMs is described in Table 3.3.3. All of these FOMs were assumed to carry equal weights, and the total FOM score of a particular configuration is simply the sum of each FOM rating. The breakdown of each concept's FOM score is provided in Table 3.3.4. The canard and conventional concepts tied, while the flying-wing scored only a few points lower. Some of the reasoning behind this scoring is discussed below.

With respect to manufacturability, the conventional concept was given the highest score of 5 because the current team had prior building experience with conventional designs. The canard was given a 4 because it was similar to the conventional layout, but would require slight modifications. The flying-wing scored lowest with a 2 because it would require a very complicated wing and payload bay design in order to produce a competitive aircraft. The ground operations scoring showed similar trends, due to the idea that 2Wx12L payload configurations would be easier to load, and that a flying-wing design would be awkward to handle if it needed to be hand-carried to the flight line.

The highest durability score was given to the flying-wing considering that it consisted primarily of a robust wing structure with minimal structural load-path concerns. The conventional and canard concepts scored lower in this category due to the increased possibility of failure as flight and landing loads were transferred through a more complex load-path. In a configuration with a conventional fuselage, the fuselage must be designed such that the loads from the wing, tail, and landing gear are carried through the structure properly. However, in a flying-wing type configuration, the loads from the wing and landing gear can be carried by the same structures, allowing for a less complicated structure.

The aerodynamic performance of each design was judged taking estimated parasite drag and theoretical induced drag into consideration. The canard and flying-wing designs both scored highest due to theoretical advantages compared to a conventional, tail-aft configuration. The flying-wing offers lower parasite drag, while a properly designed canard should have less induced drag due to the reduced loading on the wing because of the positive trim lift load on the canard. The stability FOM score, in contrast, highlighted the disadvantages of the canard and flying-wing concepts. The team members responsible for conceptual design were confident that a controllable conventional aircraft could be produced, but the flying-wing and canard concepts were unfamiliar. Primarily, the team was concerned with the difficulty of producing an aerodynamically efficient, stable, and controllable design in the cases that the flying-wing or canard concepts were selected.

The Figure of Merit rating system produced a tie between the canard and conventional concepts. Therefore, it was necessary to create a definitive tiebreaker between the two concepts. The final decision was based on a more detailed investigation of the aerodynamics of the two remaining configurations.

3.3.4 Final-Level Down-Selection via Induced Drag Analysis

Because the Figure of Merit rating resulted in a tie, it was decided that the remaining canard and conventional configurations would be compared aerodynamically. Until this point, the two concepts were assumed to have the same parasite drag, which assumes that they would have virtually the same geometry. However, it was important to investigate these two concepts with respect to induced drag, which showed differences. Obviously, the concept with the most favorable aerodynamic performance was selected.

In order to evaluate the two concepts, the lifting-line program LinAir^2 was used to compare the induced drag of the designs. The description of the details of this program will be delayed until the preliminary design discussion, where it was a major analytical tool. A description of the two compared configurations is given in Table 3.3.5. The assumptions made in this comparison were as follows.

- The wings and horizontal stabilizers of each concept are identical.
- The aerodynamics centers of the wings and horizontal stabilizers are in the same horizontal plane (i.e. no vertical gap).
- The distances between the aerodynamic centers of the wing and horizontal stabilizer are identical in each design.
- The center-of-gravity of each concept is placed such that a desirable static margin is achieved.
- Only the effects of the wing and horizontal tail are considered.

The untrimmed drag polar of each concept is shown in Figure 3.3.2 while a lift-curve for each is given in Figure 3.3.3. According to these results, the major difference between these two configurations is the zero-lift angle of attack. The canard aircraft achieves a higher aircraft lift coefficient for a given angle of attack. The drag polars, which are the major concern, were very similar. The two configurations had a similar L/D ratio, and were virtually the same at aircraft lift coefficients near 0.3, where the aircraft was expected to cruise. Therefore, from comparison of the lift-curves and drag polars the two concepts were virtually identical.

Figure 3.3.4, however, offers an important argument, showing a comparison between the trimmed canard configuration and the trimmed tail-aft conventional configuration. The two configurations had very different span-wise lift distributions on both lifting surfaces. For the lifting surfaces in the conventional configuration, the distribution was fairly smooth, trailing off near the wing tips. With the canard configuration, the wing lift distribution was discontinuous near the y-location of the canard's wing tip. As expected, the trailing vortex from the canard produced upwash on the outboard wing and downwash on the inboard wing. This discontinuity in wing loading called for a more robust and heavier wing structure. In addition to this, the loading on the canard was much larger than on the conventional configuration tail,

due to the stabilizer lift required to balance the large moment arm of the canard's wing lift about the center-of-gravity. This means that the horizontal stabilizer of the canard design also required heavier structure.

The conclusion that can be drawn from this analysis was that the conventional concept is superior to the canard concept in that it has the potential to be designed with a lighter airframe. Otherwise, there was no true advantage in using either configuration. Based on this analysis, the conventional configuration was chosen.

3.4 Conceptual Design Summary

The concept selected for further analysis was the conventional, tail-aft concept with a 2Wx12L payload layout. This selection was made taking into consideration estimated aerodynamic performance, aircraft rated cost, and qualitative performance with respect to the design requirements. This configuration was ideal because it exceeded all of the other proposed designs in the team-determined criterion necessary for a competition-worthy aircraft.

Table 3.3.1: Estimated Weight and Parasite Drag of Proposed Configurations

Concept	Conventional	Conventional Wide-Body 1	Conventional Wide-Body 2	Dual-Fuselage Single Engine	Dual-Fuselage Twin Engine	Canard	Flying-Wing	Blended-Wing-Body
Fuselage Shape	Rectangular	Teardrop	Teardrop	Circular	Circular	Rectangular Cross-Section	Integrated into wing	Faired with airfoil
Payload Configuration	OO OO OO OO OO OO OO OO OO OO OO OO	OO OO OOOO OOOO OOOO OOOO OO OO	OO OOO OOO OOOO OOOO OOO OOO OO	O O O O O O O O O O O O O O O O O O O O O O O O	O O O O O O O O O O O O O O O O O O O O O O O O	OO OO OO OO OO OO OO OO OO OO OO OO	OOOO OOOO OOOO OOOO OOOO OOOO	OOOO OOOO OOOO OOOO OOOO OOOO
Estimated C_{D0}	0.0182	0.0213	0.0207	0.0196	0.0196	0.0182	0.0139	0.0142
Estimated Weight (lbs.)	15.8	13.3	13.3	16.2	18.2	15.8	12.4	12.4

Table 3.3.2: Rated Aircraft Cost of Second-Level Concepts

Concept	Blended-Wing-Body	Flying-Wing	Conventional Wide-Body	Conventional (Canard)
Payload Configuration	4W x 6L	4W x 6L	3W x 8L	2W x 12L
Rated Aircraft Cost	11.7	12.0	13.3	14.0

Table 3.3.3: Figure of Merit Rating System

FOM	1	3	5
Manufacturability	Difficult to build	Average to build	Easy to build
Ground Operations	Slow loading	Moderate loading	Fast loading
Performance	Poor	Satisfactory	Excellent
Cost	Expensive	Satisfactory	Inexpensive
Durability	Fragile	Satisfactory	Durable
Stability	Unstable	Neutral	Stable

Table 3.3.4: Figure of Merit Scoring of Third-Level Concepts

FOM	Canard	Flying-Wing	Conventional
Manufacturability	4	2	5
Ground Operations	5	3	5
Performance	5	5	3
Cost	4	5	3
Durability	3	5	4
Stability	4	2	5
Total	25	22	25

Table 3.3.5: Input Information for Aerodynamic Comparison of Canard and Conventional Configurations

Configuration	Tail-Aft	Canard
Wing Area	10 ft^2	10 ft^2
Wing Span	10 ft	10 ft
Wing Incidence Angle	0 degrees	0 degrees
Wing Airfoil	ClarkY	ClarkY
Horizontal Stabilizer Area	1.5 ft^2	1.5 ft^2
Horizontal Stabilizer Span	2.5 ft	2.5 ft
Horizontal Stabilizer Incidence Angle to Trim at 4 degrees Angle of Attack	-1.05 degrees	8.49 degrees
Horizontal Stabilizer Airfoil	NACA 0009	NACA 0009
Distance Between Wing and Stabilizer Aerodynamic Centers	4 ft	4 ft
Location of Configuration Aerodynamic Center (measured from wing quarter-chord)	0.313 ft (aft)	0.453 ft (forward)
Location of Center-of-Gravity (measured from wing quarter-chord)	0.163 ft (aft)	.603 ft (forward)
Static Margin	0.15	0.15

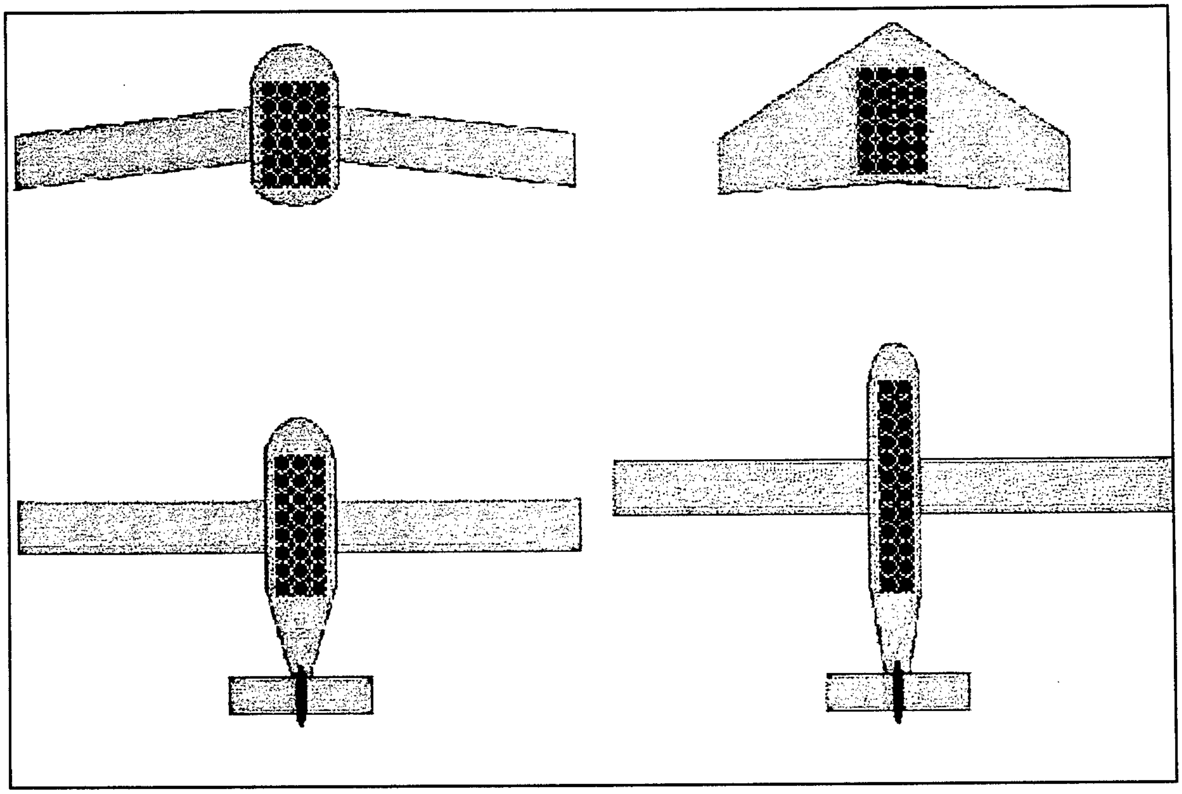


Figure 3.3.1: Layouts of Second-Level Concepts (clockwise from upper-left: blended-wing-body, flying-wing, conventional wide-body, conventional/canard)

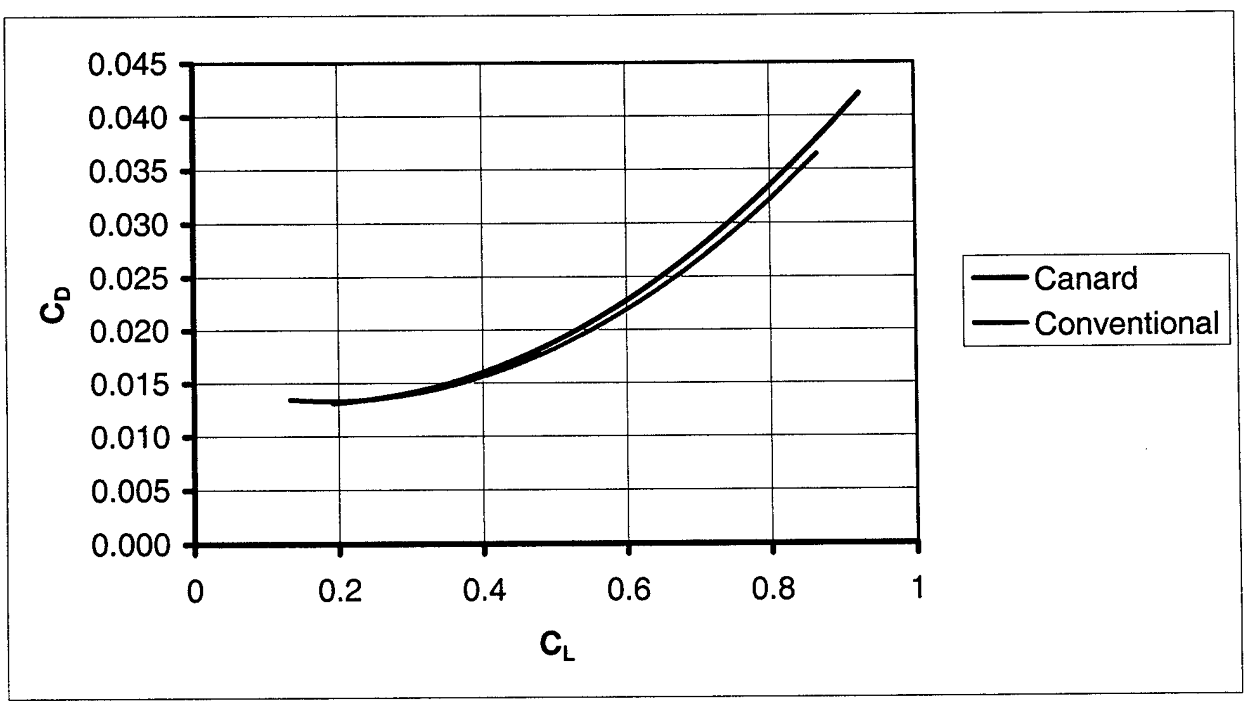


Figure 3.3.2: Comparison of Canard and Conventional Concepts: Untrimmed Drag Polars

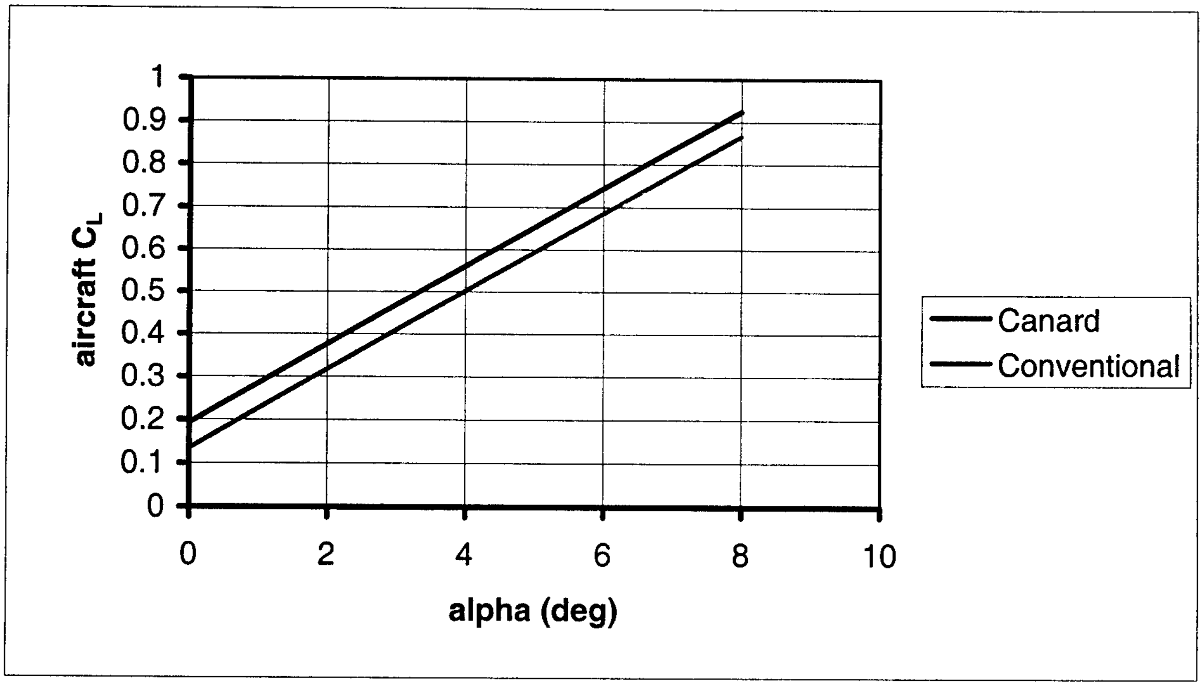


Figure 3.3.3: Comparison of Canard and Conventional Concepts: Untrimmed Lift-Curve

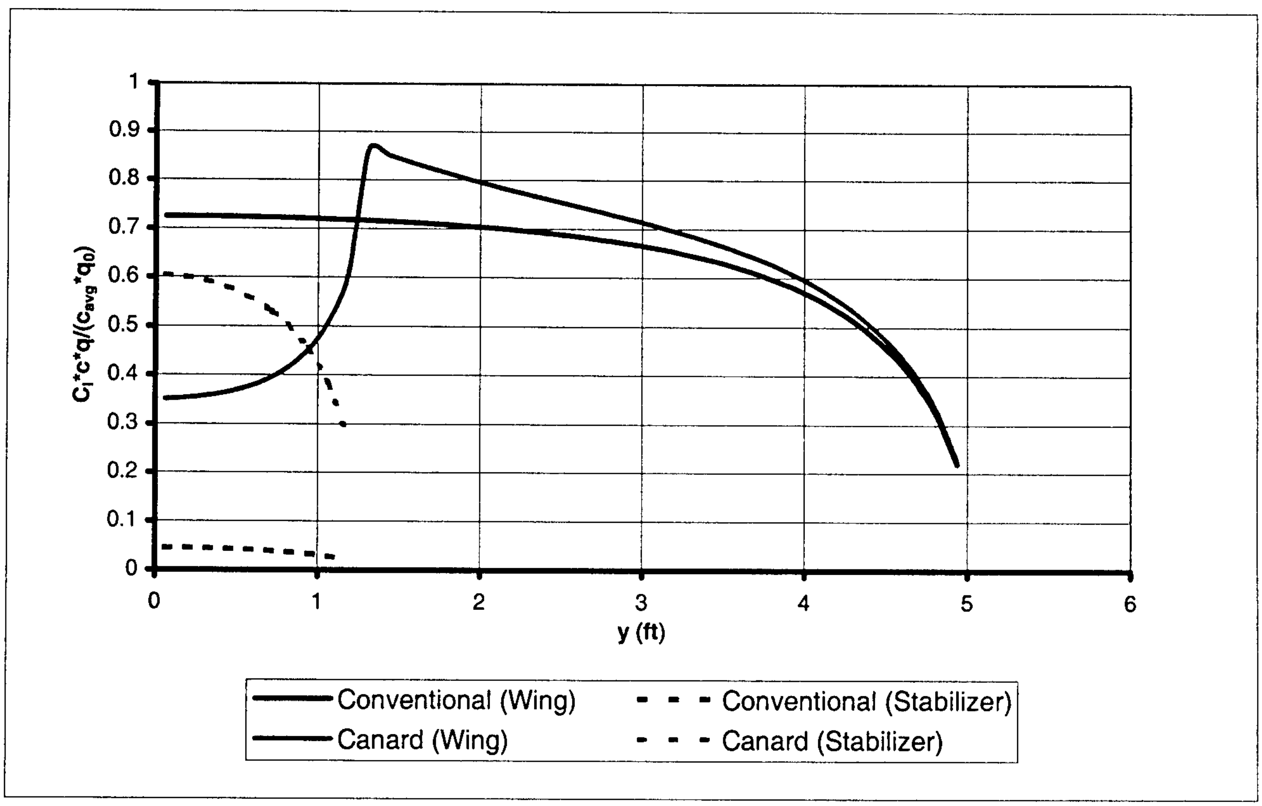


Figure 3.3.4: Comparison of Canard and Conventional Concepts: Span-wise Lift Distribution, Trimmed at 4 degrees angle of attack

4.0 Preliminary Design

4.1 Strategy

Over previous years, the team has developed and modified its performance code. The performance code required that accurate aerodynamic, propulsion, and weight models were available in order to find an optimal design. The first task was to construct these models. Once the models had been developed they could be used by the performance group to find the optimal design through a series of trade studies. The goal of these trade studies was to maximize the overall flight score. Since overall score is inversely proportional to Mission Time*rated aircraft cost, minimizing this quantity would maximize the flight score.

4.2 Analysis Tools

4.2.1 Aerodynamic Analysis

During the preliminary design work, the drag polars of several different planform configurations were constructed for use in performance analyses. The matrix of planforms studied varied in aspect ratio from 7 to 12, and in area from 9 to 10 ft^2. The Clark Y airfoil was used as a baseline wing airfoil while the NACA 0009 was used as a baseline for the tail planes. The drag characteristics of the fuselage and landing gear were also incorporated. The analysis performed was a mixture of both empirical and computational methods and showed considerable advancement over the aerodynamic analyses performed during conceptual design.

The major analytical tool used in the preliminary aerodynamic analysis was Desktop Aeronautics' LinAir 1.4. LinAir is a user-friendly program used in computing the aerodynamic characteristics of multi-element lifting surfaces. The program is based on Weissenger's discrete vortex method, which is also known as "extended lifting line theory" [Linair]. Although the program solves only for the inviscid, irrotational, and subsonic case, it allows the input of airfoil drag polar characteristics. In the case of preliminary design work, LinAir was used to investigate the aerodynamic properties of the wing and tail of each planform in the configuration matrix. The horizontal tail was sized by assuming that its span was 25% of the wingspan and selecting its area based on tail-volume coefficients [Raymer]. Once the dimensions of a particular planform configuration were known, a LinAir data file was created using the baseline airfoils for each surface. A sample LinAir input file display is shown in Fig. 4.2.1.

After being analyzed in LinAir, each configuration in the matrix was grouped with a component drag for the fuselage, landing gear, and an appropriately sized vertical tail. The vertical tail was sized based on volume coefficients while the parasite drag of the landing gear, fuselage, and vertical tail and interference drag were estimated from equations given in Raymer.[1] The drag polar for each configuration was then plotted and fit to a quadratic equation. The coefficients of this curve fit were used directly in the performance analysis. A comparison of the estimated untrimmed drag polars for varying planforms is provided in Fig. 4.2.2.

4.2.2 Propulsion

4.2.2.1 MotoCalc™ Data

The primary tool for the selection of an initial propulsion system was the commercially available software MotoCalc™.[3] The program allows the user to configure a propulsion system and then estimate useful parameters such as thrust and current drawn by the particular setup. It is able to create predictions for both static and in-flight situations. MotoCalc™ did not predict actual propulsion performance accurately, however, it did show general trends well. It was a useful tool for narrowing the configuration possibilities to a field small enough for more thorough testing in the time before the competition.

4.2.2.2 Preliminary Wind Tunnel Data

Wind tunnel data gathered by the team for the 1999-2000 contest season was used as a tool to verify the predictions of MotoCalc™. These data contain tests performed with propellers of sizes 16x10, 16x14 and 18x8 and gear ratios of 5:1 and 2.72:1 on an Astroflight™ Cobalt 60 motor. These propeller and gearbox combinations were tested over a range of velocities between 0 and 80 ft/sec.

In comparing the 1999-2000 wind tunnel data with the MotoCalc™ predictions, one general trend was noticed that could significantly change the energy usage predictions. The MotoCalc™ predictions showed that the current drawn by the motor decreased as the velocity of the aircraft increased. This decrease in current was as much as 67% at an airspeed of 80 ft/sec. The wind tunnel tests, however, showed only a 13% decrease in current as airspeed increased to 80 ft/sec. In past years, the flight duration has fallen short of the performance code predictions. It was believed that this was due, at least in part, to the use of MotoCalc™ predicted currents in calculating battery runtime. In an attempt to improve the accuracy of the energy usage predictions this year, the initial energy calculations were made using current as a function of throttle setting only, as opposed to the past method of using current as a function of both throttle setting and airspeed. This represents a conservative approach and should ensure that adequate energy will be available for the competition flights.

4.2.2.3 Programs

In order to create predictions of the performance of the aircraft as a whole, the propulsion system data output by MotoCalc™ or gathered in the wind tunnel needs to be prepared for integration with parameters from other aspects of the aircraft. This was accomplished through a team-written MATLAB™[4] code. It also included the ability to estimate current as a function of throttle setting alone or both throttle setting and airspeed. The code read text files containing wind-tunnel data or MotoCalc™ output data for a given configuration to obtain vectors containing airspeed, current and thrust. Current was then described as a function of thrust through by fitting a third-order polynomial. Next, the thrust was described as a function of airspeed, also by fitting a third-order polynomial to the data. These two curve fits resulted in a set of 16 coefficients that modeled current drawn as a function of airspeed. One last third-order polynomial fit was performed on the data to give maximum thrust as a function of airspeed,

which resulted in four more coefficients. Finally, an output file was created containing the 20 coefficients. This output file provided exactly the propulsion system data required by the team's performance code.

4.2.3 Stability and Control

For initial sizing of the tail, volume coefficients were used. The volume coefficients were determined by calculating the coefficients from previous competition aircraft. They were then averaged and used to estimate stability and control characteristics of the aircraft. Tail geometries corresponding to various wing areas, aspect ratios and lever arms were calculated using this method. The results can be found in Table 4.2.1. A common symmetrical airfoil, the NACA 0009, was chosen for the tail.

After the different tails had been sized, an analysis of the static margin for each tail size was performed using the method outlined in Raymer[1]. The center of gravity was assumed to be at 33% of the wing chord. It was found that all static margins fell within the desired range of 10-14%. The selected final wing dimensions were an area of nine ft^2 with a nine-foot span. Using that information, the corresponding tail that had already been calculated was used in preliminary design.

4.2.4 Structures

4.2.4.1 Structural Tests

A series of wing structure tests were performed to study such properties as manufacturing technique, structural weight, and the maximum load at which failure occurs. The test rig consisted of two fixed support points, 35 inches apart. A clamping mechanism was used at the support points to prevent the test section from twisting or translating while being loaded. Weight was applied to the center of the test section in one-pound increments until failure of the test section occurred. The sections tested and their dimensions are given in Table 4.2.2. Test number, sections tested, and results are summarized in Table 4.2.3.

Test TS-1 compared two favored spar-manufacturing techniques. Section WS-1 was constructed using basswood for a two-piece, top and bottom, main spar located at the quarter chord. Section WS-2 was constructed using an I-beam main spar located at one third of the chord. The spar was constructed using carbon fiber caps and a balsa wood shear web with the grain oriented parallel to the span. Both sections used a NACA 4415 airfoil with a solid balsa leading edge and sheet balsa trailing edge. They were covered in an identical fashion using Monokote™. Results from test TS-1 led to the selection of the I-beam, main spar construction used in section WS-2 for further testing.

Test TS-2 was a study of failure load verses spar thickness. Sections IS-1, IS-2, and IS-3 were I-beam main spars corresponding to 10%, 12%, and 15% thickness ratio airfoils respectively, assuming a wing chord of 12 inches. They were constructed with carbon fiber caps and a balsa shear web with the grain oriented parallel to the span. Test TS-3 was a study of failure load verses shear web construction. Sections IS-4, IS-5, and IS-6 were I-beam main spars identical to sections IS-1, IS-2, and IS-3 but with the balsa shear web-oriented perpendicular to the span. This grain orientation significantly increased the I-beam failure load.

4.2.4.2 Weight Model

Results from structure tests TS-1 through TS-3, in addition to historical data, were used to create a weight model. Table 4.2.4 shows historical wing data for competition aircraft dating back two years in addition to section WS-2 from test TS-1. Span was plotted against wing weight for these three cases and a second-order polynomial trend line was fitted to the data points shown in Figure 4.2.1. The equation for the trend line was used to find wing weight as a function of wing area and aspect ratio. A plot of wing weight verses wing area for a range of constant aspect ratios was then constructed. This is presented in Figure 4.2.2. In addition, trend lines were fitted to each of the constant aspect ratio lines. The equations for the lines were of the form $Y=AX^{0.5}$ with A dependent on aspect ratio. In order to estimate wing weight for any aspect ratio, the coefficient A was plotted verses aspect ratio. By using this graph, any aspect ratio could be selected and a corresponding coefficient A could be found. The coefficient A could then be used in the equation $Y=AX^{0.5}$ to determine wing weight from wing area, assuming similar building techniques. The fuselage weight was estimated from the length of fuselage based on historical competition aircraft data. Thus, a total aircraft weight could be estimated based on aspect ratio, wing area and fuselage length.

4.2.5 Performance

A team-written code was developed to analyze the performance of candidate aircraft configurations. The code applies a fourth-order Runge-Kutta method to solve Newton's Second Law of Motion as applied to the aircraft. This code gave the design team the ability to accurately model the aircraft's performance in an actual competition mission. Inputs to this code include: aerodynamics, geometry, weight, motor performance and mission description. The primary outputs of this code were mission time and electrical energy used. Using this tool, the performance team evaluated many variations of the configuration that had resulted from conceptual design work.

Some assumptions were made to simplify the code. First, constant altitude turns were assumed. Second, the throttle was defined as a percentage of maximum thrust. In practice, the aircraft throttle does not operate exactly that way, but it was the most convenient way to define throttle in the model. Last, the rolling friction coefficient was determined empirically during previous flight tests to be 0.1. All of these assumptions were reasonable, such that the accuracy of the code was a function of the quality of the aerodynamic, propulsion and weight models.

The performance code produces a wide range of output data. The flight profile is a history that shows how much time the aircraft spends in each segment (turn, straight, takeoff, etc) as well as its speed, g-loading and energy usage. Takeoff distance is also computed and displayed with the flight profile. The time history of parameters such as: C_L, C_D, V, n and T were added to the output to gain insight into the operating ranges of the aircraft.

The goal of the performance analysis was to create an airplane that could win the 2002 DBF Competition. It was obvious that, due to the total score equation, the only ways to maximize the score would be to minimize Mission Time*RAC, maximize report score and maximize the payload. However,

since Mission Time*RAC and payload were related, more attention needed to be directed there. With that in mind, the performance analysis group set out to develop an optimal aircraft.

4.3 Trade Studies

Trade studies are a critical part of any preliminary design effort. Using trade studies, the design team was able to develop an aircraft that would optimally perform the mission laid out in the contest rules. The trade studies carried out in preliminary design included: parameter sensitivity, geometry variations, airfoils and g loading. The final values of the several aircraft parameters were finalized as part of the detail design work discussed in Section 5.0.

Sensitivity studies were used to determine which parameters had the greatest effect on performance objectives such as takeoff distance and total time. In order to conduct these studies, a baseline aircraft was established from the previous preliminary sizing work. These studies yielded many interesting results. Figure 4.3.1 presents the sensitivity of maximum (fully loaded) takeoff to various parameters. The "non-dimensional parameter" is a linear interpolation equal to zero when the parameter was a minimum and equal to one when it was a maximum. Figure 4.3.1 shows that to minimize takeoff distance a large wing and a light aircraft was best. This was an obvious result and it showed that weight had the largest influence. Figure 4.3.2 presents the sensitivity of total mission time to the same parameters. The "non-dimensional parameter" was defined as in Figure 4.3.1. It showed that C_{D0} and weight had the largest effect in minimizing total mission time. Again, the obvious result was that minimizing them resulted in the best outcome. Wing chord also played a prominent role in both areas, but had an opposite effect in each. For takeoff, the results suggested a large chord while for total time they suggested a small chord.

Energy usage was an important concern for this design. Not carrying unnecessary battery weight would greatly improve the overall aircraft score. However, getting the most speed out of the aircraft required that the propulsion system operate at a high throttle throughout the mission. Considering that, a study was conducted to evaluate the penalty on total score due to flying the mission at less than full throttle. It was shown that reducing the throttle from full to 80% resulted in approximately a 5% increase in mission time times Rated Aircraft Cost (RAC), see Figure 4.3.3. In addition, it was found that any reduction in the number of cells resulted in an unacceptable loss of maximum thrust for takeoff (Table 4.3.1). Therefore, it was decided that the missions would be flown at full throttle as much as possible in order to take advantage of the batteries being carried.

Payload size and configuration determined the overall aircraft size and shape. The first thing considered was determining whether or not it was advantageous to carry less than the maximum number of softballs as payload. This was considered in an attempt to reduce weight as well as drag. Through some simple calculations and the use of the code, it became evident that carrying less than the maximum was not advantageous. Although the aircraft became lighter and smaller, the effect of the reduction in payload outpaced the reduction in time and the total score fell. The next step was to determine what the best payload layout would be. Many payload configurations were considered including a 2Wx12L, a

3Wx8L, and several elaborate diamond-like layouts. The 2Wx12L layout resulted in lower drag, but was slightly heavier than the 3Wx8L layout. The reduction in drag though, outweighed the weight increase in terms of added performance. Since the sensitivity study demonstrated the importance of C_{D0} and weight with respect to mission time, it was an easy choice. Selecting the payload configuration made it possible to accurately estimate the empty weight of the configuration from structural data. Now that the fuselage had a general size, shape and weight, it was possible to move on to other areas.

The next step in the preliminary design process was to optimize the wing geometry. Aerodynamic data, for a variety of wing planforms and the Clark-Y baseline airfoil, were acquired from aerodynamics. The performance code was then used to generate takeoff distance, mission time and energy usage. The driving constraint of the mission was found to be takeoff distance. The total energy usage estimates for the mission fell well below the estimated effective battery energy output of approximately 1800 mAh. Figure 4.3.4 shows the variation of takeoff distance with wing area for constant aspect ratio. The yellow buffer zone was established as a takeoff limit to ensure that the aircraft would be able to takeoff within the maximum allowable distance of 200ft.

The data were also calculated taking into account a thrust loss, observed during previous competitions, that occurs as the battery pack is drained. In addition, a manufacturing weight gain of 10% was added to the estimated weight to account for weight gains observed in the past. These adjustments caused the original propeller, an 18x18, to no longer provide sufficient thrust for takeoff. A 20x16 propeller generated enough thrust for takeoff. The starred-point at an area of 10 ft^2 and an aspect ratio (AR) of 12 was initially selected based on Figure 4.3.4 and Figure 4.3.5. It shows the variation of Mission Time*RAC versus wing area for constant values of AR. This showed that the starred point minimized Mission Time*RAC as desired. Note that the difference in Mission Time*RAC between the AR of 12 point and the AR of 7 point was just over 1%.

4.4 Preliminary Design Summary

After extensive testing and conceptual design analysis, the following design parameters were chosen. In order to minimize C_{D0} and thus minimize the parameter Mission Time*RAC, a payload configuration of two softballs wide by 12 softballs long was chosen. The performance optimized wing geometry was determined to have an area of 10 ft^2 and an aspect ratio of 12. It was also determined that the tail would utilize the NACA 0009 airfoil. Choosing parameters such as the wing airfoil, propeller, and building materials now began.

Table 4.2.1: Wing and Tail Parameters

Main Wing						
AR	S_{ref} (ft^2)	b_{ref} (ft)	c (ft)	diameter of fuselage	S_{wetted} (ft^2)	$\lambda = c_{tip}/c_{root}$
7	9	7.937	1.134	0.833	8.167	1
8.5	9.3	8.891	1.046	0.833	8.428	1
10.5	9.7	10.092	0.961	0.833	8.899	1
12	10	10.954	0.913	0.833	9.239	1
Horizontal Tail						
AR	S_{ref} (ft^2)	b_{ref} (ft)	c (ft)	diameter of fuselage at tail attachment	S_{wetted} (ft^2)	$\lambda = c_{tip}/c_{root}$
3	2.24	2.592	0.864	0.083	2.168	1
3	2.12	2.522	0.841	0.083	2.050	1
3	2.04	2.474	0.825	0.083	1.971	1
3	1.996	2.447	0.816	0.083	1.928	1

Table 4.2.2: Test Section Description

Test Section	**Span (inches)**	**Thickness (inches)**	**Chord (inches)**	**Area ($inches^2$)**	**Weight (pounds)**	**Description**
WS-1	35.875	1.8	12	430.5	0.625	Complete Wing, Basswood Spar
WS-2	35.875	1.8	12	430.5	0.8125	Complete Wing, Carbon Fiber I-beam Spar
IS-1	35.875	1.2	-------	-------		I-beam Spar, Parallel Grain
IS-2	35.875	1.44	-------	-------		I-beam Spar, Parallel Grain
IS-3	35.875	1.8	-------	-------		I-beam Spar, Parallel Grain
IS-4	35.875	1.2	-------	-------		I-beam Spar, Perpendicular Grain
IS-5	35.875	1.44	-------	-------		I-beam Spar, Perpendicular Grain
IS-6	35.875	1.8	-------	-------		I-beam Spar, Perpendicular Grain

Table 4.2.3: Test Results

Test Number	Test Section	Load at Failure (pounds)	Notes
TS-1	WS-1	75.50	
	WS-2	101.00	
TS-2	IS-1	32.11	
	IS-2	40.81	
	IS-3	75.96	
TS-3	IS-4	75.04	
	IS-5	47.19	1
	IS-6	103.00	
Note 1:	I-beam cap not properly attached to shear web resulted in premature failure		

Table 4.2.4: Historical Wing Data

	Span (inches)	Wing Area (sq. inches)	Aspect Ratio	Wing Weight (pounds)
Accipiter	108.00	1944.00	6.00	3.83
Hobbes	84.00	1512.00	4.67	3.21
WS-2	35.88	385.66	3.34	0.81

Table 4.3.1: Battery Cell Trade Study

Number of Cells	Maximum Thrust (lb)	Takeoff Distance (ft)
36	11.375	178.2
34	10.475	202.4
32	9.56	226.5

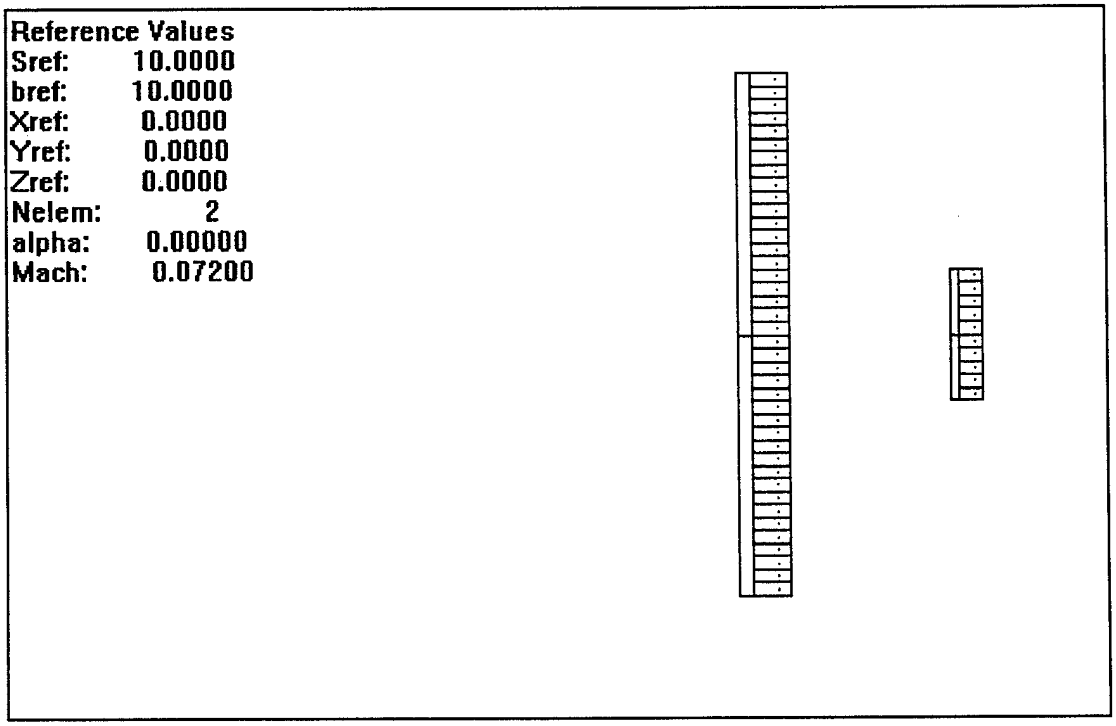


Figure 4.2.1 Sample LinAir Input Display

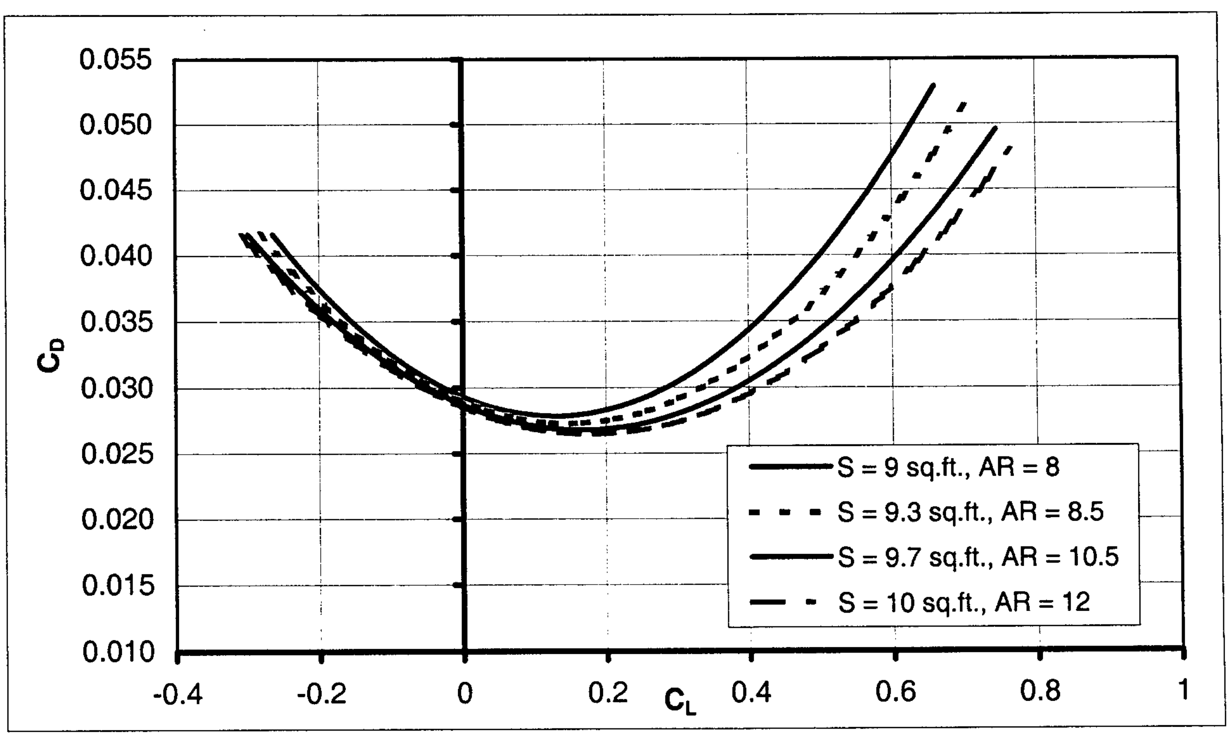


Figure 4.2.2 Untrimmed Aircraft Drag Polar for Various Planforms (ClarkY wing, NACA 0009 tail)

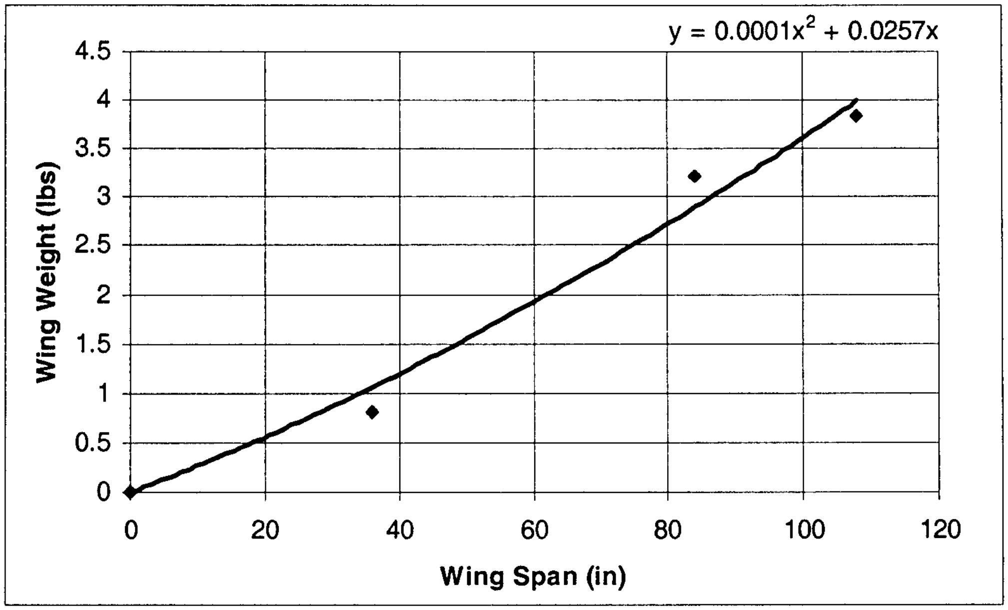


Figure 4.2.1: Second Order Curve Fit of Wing Weight Versus Wing Span

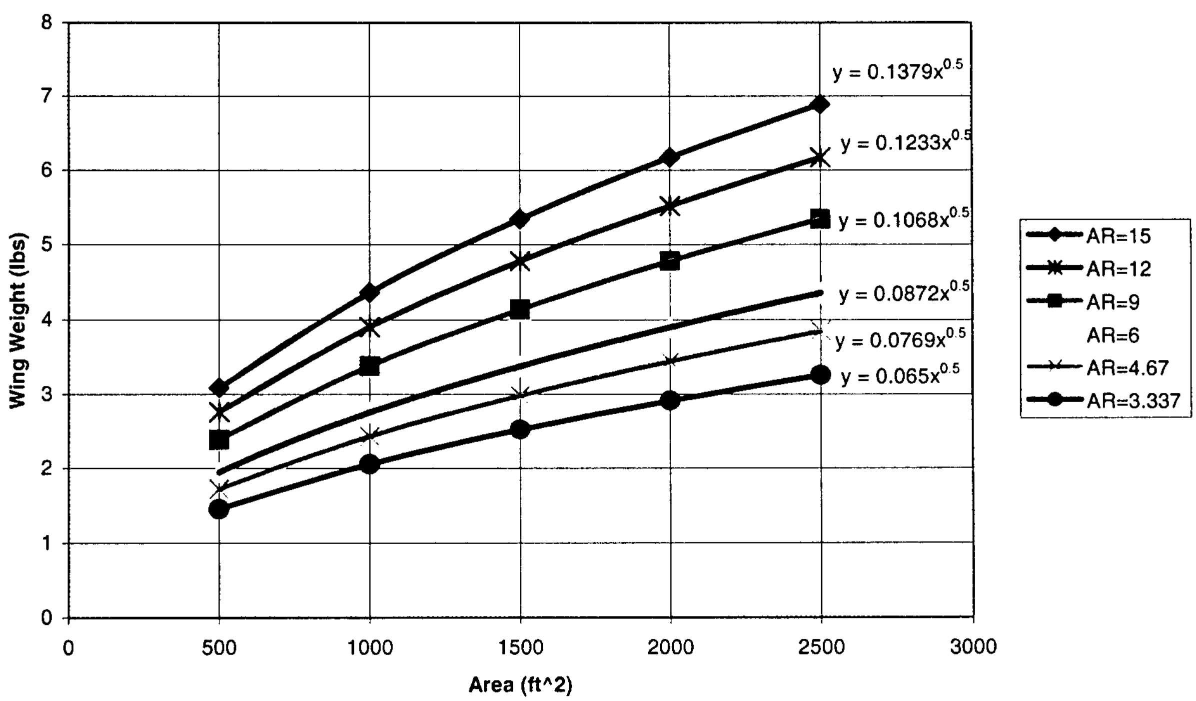


Figure 4.2.2: Wing Weight as a Function of Wing Area

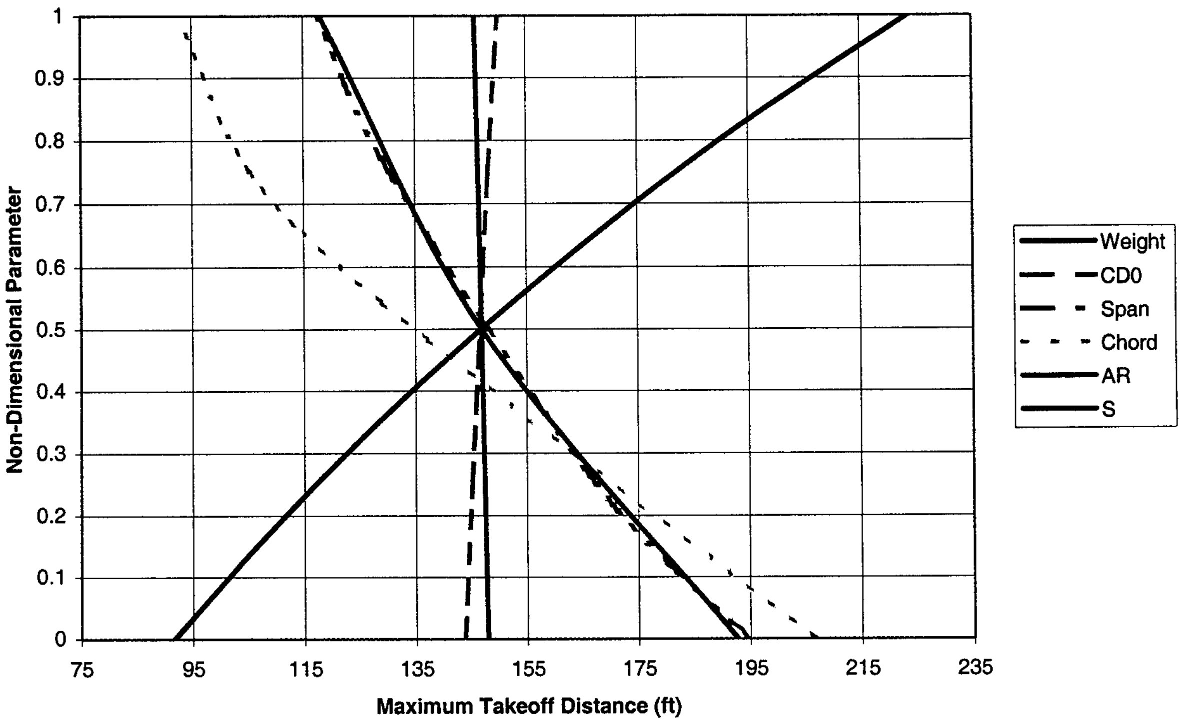


Figure 4.3.1: Sensitivity of Maximum Takeoff Distance

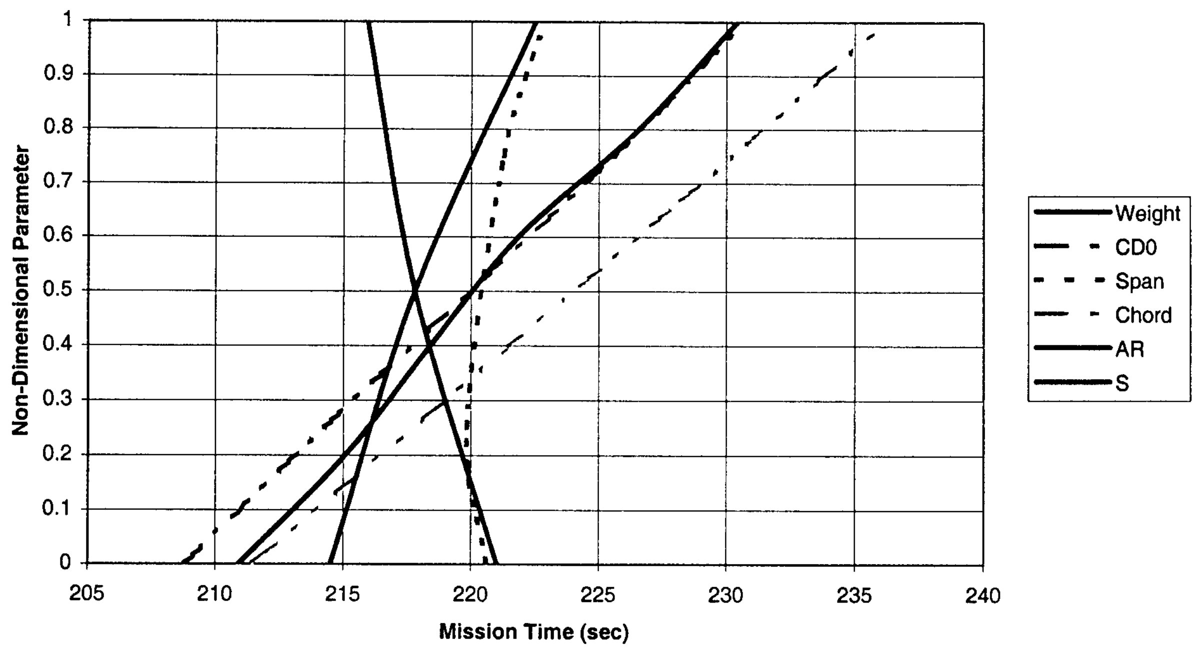


Figure 4.3.2: Sensitivity of Total Mission Time

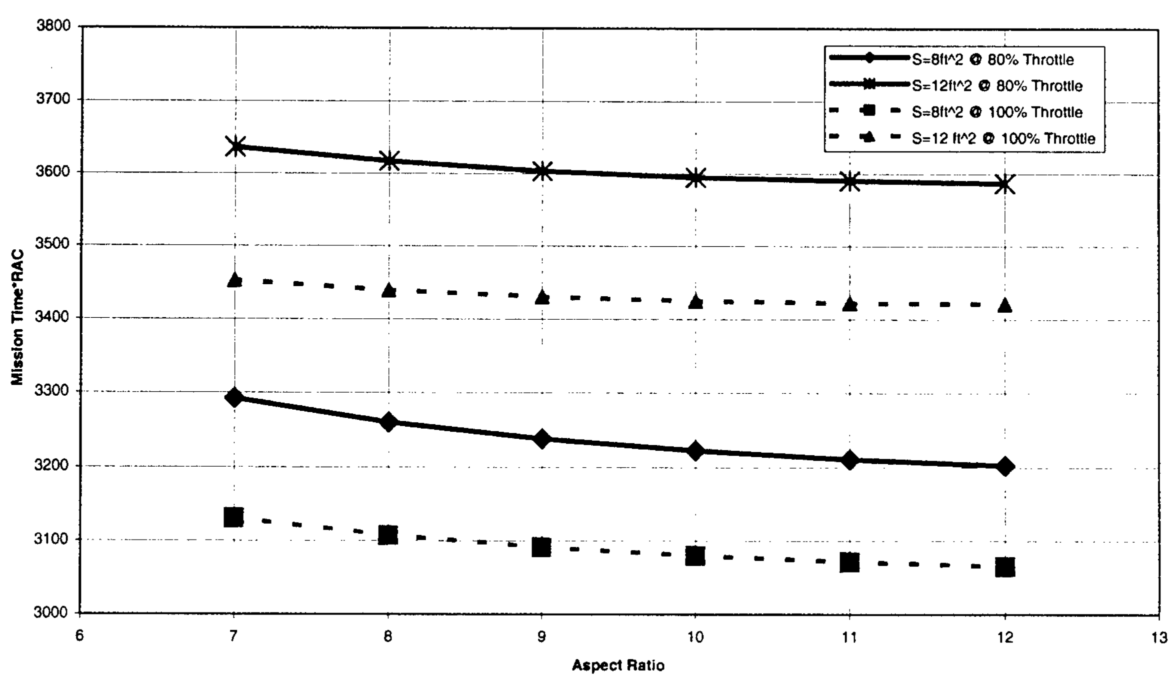


Figure 4.3.3: Throttle Setting Study

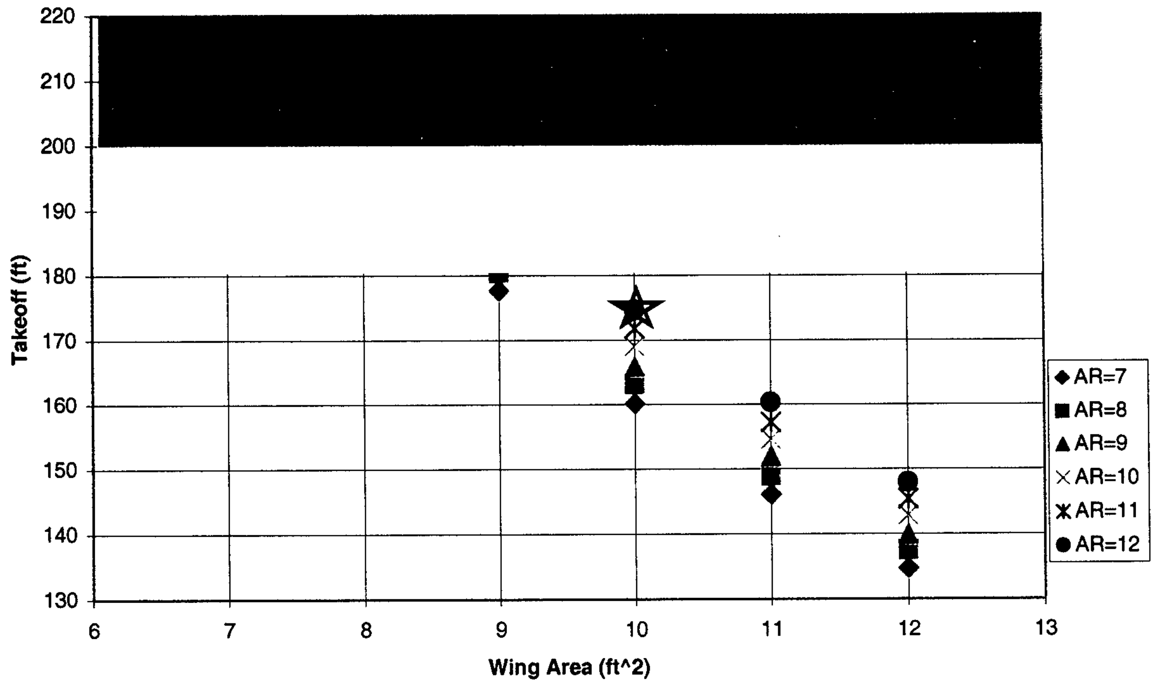


Figure 4.3.4: Takeoff Distance vs. Wing Area

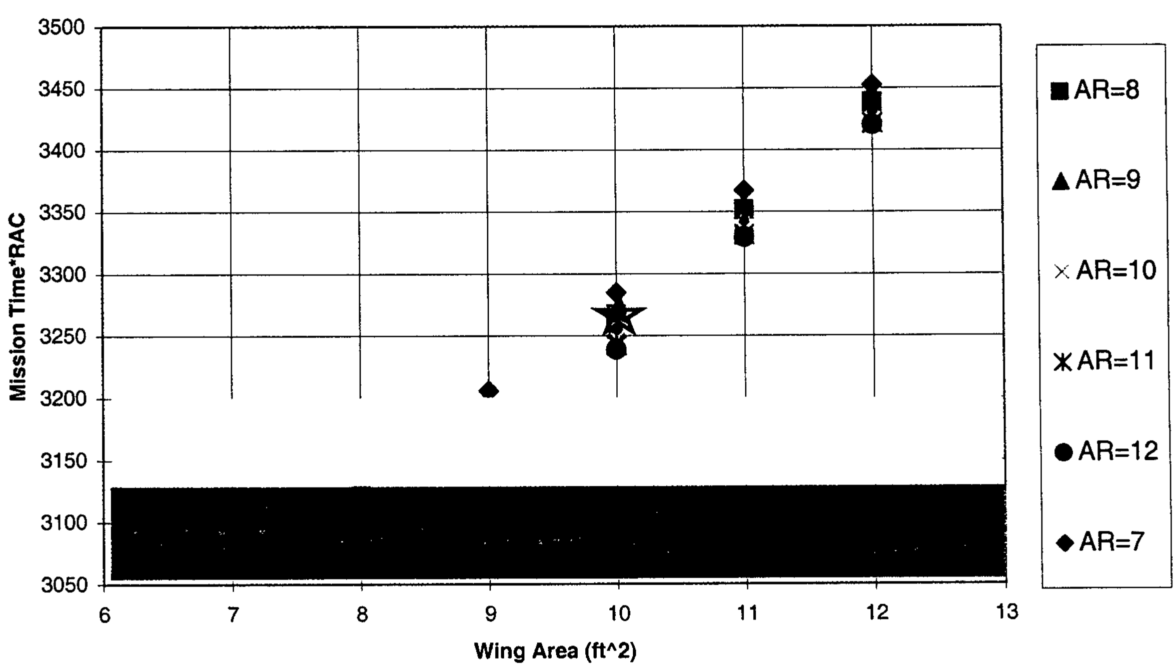


Figure 4.3.5: Mission Time*RAC vs. Wing Area

5.0 Detail Design

This section includes the optimization of components designed in the preliminary stage such as planform finalization, fuselage design, airfoil selection, propeller selection, and control surface sizing. In addition, this section covers other aspects of the aircraft such as the landing gear and the engine cowling.

5.1 Aerodynamic Analyses

The primary result of preliminary design work was a performance-optimized sizing of the aircraft's planform. Other parameters describing the aircraft's geometry had been based primarily on the payload configuration determined during the conceptual design studies. At the detail design level, it was necessary to perform some aerodynamic trade studies to finalize the details of the configuration. Two trade studies were performed. They involved airfoil selection and the refinement of the fuselage geometry. In addition to these studies, the design of the cowling and battery cooling system were developed. The aerodynamics of the final configuration were then calculated and used to refine the estimation of the competition aircraft's performance. The details of the final aerodynamic design work are summarized in the following sections

5.1.1 Airfoil Studies

The primary trade study performed for detailed aerodynamic design work was the selection of the wing airfoil. This study was performed in a fashion similar to the one used to select the planform during preliminary design. Given a wing and horizontal tail planform configuration, a separate LinAir[2] model was developed to determine the drag polar of these components for each airfoil studied. Once these drag polars were constructed, the parasite and interference drags of other components were added to each polar. These drag polars were then fitted with a quadratic polynomial and the coefficients of this polynomial were used directly in the performance estimation code.

The four wing airfoils studied were the Clark Y, RG14, S8036, and SG6041. These airfoils were chosen because they represented a wide variation in aerodynamic characteristics. Data about each of these airfoils is available through the UIUC Low Speed Airfoil Test[5] website. The objective of the study was to determine the best airfoil through an investigation of the effects on aircraft performance. Airfoil lift-curves and drag polar data for the airfoils studied are shown in Figures 5.1.1 and 5.1.2, respectively. The study began by determining a sectional drag polar for each, fitting a polynomial to each polar and using the coefficients of that polynomial in the LinAir program to develop the three-dimensional drag polars of the wing and horizontal tail; a NACA 0009 tail airfoil section was assumed. Then the parasite drags of the other aircraft components were added to the drag polars and the effects of airfoils on performance were determined using the analysis described in Section 5.3 below. A comparison of the resulting drag polars is provided in Figure 5.1.3.

5.1.2 Fuselage Aerodynamic Studies

It was also necessary to study the fuselage configuration to determine the best layout from an aerodynamic standpoint. During the conceptual design phase, the payload configuration and the general aircraft layout had been selected. However, the actual fuselage dimensions had not been set. In

particular, the fuselage layout had not been produced with respect to the placements of batteries around the softball payload. The best fuselage layout was selected through comparison of both the fuselage drag, and the impact of fuselage length on cost.

Three different fuselage layouts were proposed. The cross-sections are shown in Figure 5.1.4. Two of these configurations had the battery pack aligned along the length of the fuselage while the other placed the batteries in a compartment similar to the compartments used to hold the softballs. Aligning the batteries along the length increases the frontal area while grouping them in a compartment increases the fuselage length. It was important to determine whether or not the additional length and reduced frontal area (i.e. an increase in fineness ratio) of the first configuration benefited the aerodynamics enough to justify the additional aircraft length. The results of this study are tabulated in Table 5.1.1. It is obvious from this study that the drag count available with the first design is not sufficient to warrant the increase in aircraft length and rated cost. However, the effect of these configurations on performance was also studied (to be discussed in Section 5.3). The third configuration was finally chosen because it had lower drag than the second configuration, it would allow for better battery cooling, and it would be easier to construct with respect to the proposed I-beam fuselage (to be discussed in Section 5.4).

5.1.3 Cowl Design

The design of the cowl was produced given the rules-of-thumb provided in Raymer[1] for the design of piston-engine cooling systems. The requirement of this design was to provide sufficient cooling for both the motor and the battery pack. In order to ensure that these components received sufficient cooling airflow, it was decided that each should have a separate ducting system. The ducting system for the battery consists of a rectangular intake with roughly the same dimensions as the battery compartment and an outlet of the same size at the rear of the battery compartment. The ducting system for the motor has a semicircular intake sized to 50% of the motor's frontal area, placed on the side of the cowl where the motor is located. The flow into this intake is passed over the motor and is ejected through an outlet in the side of the cowl opposite the motor. This cowl outlet has an area that is approximately 75% of the motor's cross-sectional area. This design is based primarily on ducting air from areas of high pressure at the aircraft nose to areas of low pressure at the rear of the cowling or battery compartment. A drawing of the cowl is presented in Fig. 5.1.5.

5.1.4 Drag Build-Up of the Final Configuration

Once the entire aircraft configuration had been determined (See Section 5.5), the total drag polar of the competition aircraft could be calculated. This calculation involved two steps: developing a LinAir[2] model to determine the total (parasite plus induced) drag of the wing and tail, and determining the parasite drag of the fuselage, the vertical tail, and the landing gear. The LinAir model was developed using the final geometry of the wing and horizontal tail and the drag polars for their respective airfoils. The parasite drags of non-lifting components were developed using the empirical equations provided in Raymer[1]. The results of the final parasite drag build-up are provided in Table 5.1.2. The parasite drags of the fuselage, landing gear, and vertical tail were then added to the drag polar for the wing and

horizontal tail produced by LinAir. The final untrimmed aircraft drag polar is provided in Fig. 5.1.6. These drag data were used in the final calculations of the mission performance.

Use of LinAir to develop a drag polar also produced an untrimmed lift-curve for the airplane. The resulting aircraft lift curve is shown in Fig. 5.1.7. Due to the limitations of LinAir, this only predicts the linear portion of the lift-curve. The C_{Lmax} of the aircraft can be estimated as 90% of the airfoil C_{lmax} according to Raymer[1]. For a wing using the S8036, this corresponds to a C_{Lmax} of approximately 1.16.

5.2 Propulsion

5.2.1 Wind Tunnel Testing

The team was able to use a University wind tunnel for a brief time this year. The testing plan called for three categories of tests to be performed. These tests were:

- Thrust and current characteristics testing
- Battery voltage characteristics testing
- Competition flight simulation testing

In each test type, the parameters recorded were: throttle setting, battery voltage, battery current, battery charge, battery output power, motor voltage, motor amperage, motor power, propeller RPM, thrust and the conditions in the wind tunnel (velocity, temperature, and pressure). Prior to the testing, drag measurements were obtained for the motor mount apparatus inside the wind tunnel over the range of velocities used. This was used in calculating the true thrust produced by the motor. The velocity correction for this experiment was 1% or less according to the propeller test correction equation in Pope.[6] Therefore, the tunnel boundary correction was considered to be acceptable. All tests were performed using an AstroFlight ™ Cobalt 60 motor, an AstroFlight ™ 2.72:1 gearbox and an AstroFlight ™ 204D electronic speed control.

5.2.1.1 Thrust and Current Characteristics Testing

This test was used to characterize the thrust produced and current drawn by various propeller, motor, and battery combinations at all throttle settings over velocities ranging from zero to 100 ft/s. Data were gathered over a set of six throttle settings (0, 25, 50, 66, 84 and 100%) and five velocities (0, 25, 50, 75 and 100 ft/sec). Third-order polynomials were fit to the data to estimate parameter values at any point in the range.

From the data obtained in this test, input files containing functions to describe current draw and maximum thrust, both as functions of airspeed, were generated for use with the team's performance code. This facilitated a revision of the mission performance prediction using experimental data. Team experience suggested that this prediction should be closer to the actual performance of the aircraft than the prediction based on MotoCalc™ output.

These data also were used in a comparison to determine the level of accuracy of MotoCalc™ output. Unfortunately, the experimental data are not consistent with the output from MotoCalc™. Figure 5.2.1 shows MotoCalc's™ thrust predictions with at most a 10% error over the entire velocity range for the APC™ 18x18 propeller, but this error increases to 30% and above for the APC™ 20x14. Although

MotoCalc's™ thrust predictions agree with experimental data in the 25 ft/sec to 75 ft/sec velocity range for all three of the propellers tested, a final conclusion can not be made regarding its accuracy without further wind tunnel testing. From the current data, MotoCalc™ predictions are not a reasonable substitute for experimental propulsion data to model aircraft performance with high accuracy. However, MotoCalc™ predicts general trends well and is still useful as a preliminary design tool.

Current draw results from previous years' wind tunnel tests were replicated in this recent test. It can be seen from the plots in Figure 5.2.2 that in the wind tunnel, current did not decrease significantly with increasing airspeed. MotoCalc™ predicted a large drop for both the APC™ 20x14 and APC™ 20x16 propellers. Over the range of propellers tested, it appears that the assumption of constant current at constant throttle setting is a conservative one.

5.2.1.2 Battery Voltage Characteristics

The objective of the second test was to describe the voltage characteristics of the battery packs as their energy was depleted. These tests indicated how much energy can be used from the battery packs before a significant voltage drop occurs, resulting in thrust loss. In this test, a constant throttle and velocity was set and the propulsion system was run until the battery voltage dropped to a predetermined level. This test was performed for a set of four throttle settings (50, 66, 84 and 100%) and four velocities (25, 50, 75 and 100 ft/sec). One 36-cell pack of SR™ 2400 Max batteries and one 36-cell pack of Sanyo™ RC-2400 batteries were used during wind tunnel testing. These particular cells were chosen through a comparison of specifications on available cells from three major manufacturers. This comparison is shown in Table 5.3.1. The SR and Sanyo 2400 mAh cells were chosen for more detailed study due to their higher capacity-to-weight ratio.

Despite the fact that the cells in both packs were rated at a capacity of 2400 mAh, the pack voltage dropped off significantly before 2400 mAh had been used, as Figure 5.2.3 shows. The Sanyo™ RC-2400 cells began a sharp drop in voltage after discharging just 1500-1750 mAh. The SR Batteries™ 2400 Max cells began to drop sharply in voltage after 1750-2000 mAh were discharged. Not only did the SR™ cells hold their voltage through a larger percent of their total capacity than the Sanyo cells, but they also have a higher energy density.

5.2.1.3 Competition Flight Simulation

The third test was designed to simulate a competition flight and gather energy usage data. These tests were performed by varying throttle and tunnel velocity settings according to data output from the performance code.

The average total energy used, for five simulated competition flights, was 1501.2 mAh. The performance code produced a different and lower value. This difference is attributed to the way this value is determined. The performance code utilizes the measured data, while the average value was determined by the final mAh reading. The results suggest that the aircraft should be able to complete its entire flight without using all of the available battery energy as determined in the tests described in Section 5.4.2. The competition flight simulation test also gave an idea of how much thrust could be

expected for the second and third takeoffs during a flight. To ensure that the required takeoff distance could be achieved for the final takeoff in each flight, it was necessary to know how much thrust would be available on a partially depleted battery pack. A decrease in static thrust of up to 40% at the time of the third takeoff was observed. This was accounted for in the final performance estimates for the aircraft.

5.2.2 Propeller Selection

Since a commercially produced propeller with no more than slight modification must be used, finding a manufacturer with the highest performance propellers at a reasonable cost with good availability is important. APC™ propellers were chosen due to their reputation for good quality, wide selection and ease of acquisition. The APC™ 20x16 propeller was selected for its ability to provide the estimated required thrust when used with an Astroflight™ Cobalt 60 motor with its stock 2.72:1 ratio gearbox. The wind tunnel data supports this choice and also indicated that the setup would provide enough runtime to complete the mission with energy to spare.

5.3 Structural Analysis

5.3.1 g-Load Capability

The relationship between velocity and g-loading is presented in a V-n diagram shown in Figure 5.3.1 using the method from McCormick.[7] The positive and negative limit load factors resulted from an analysis of main spar strength and were found to be 3.1 g's and -2 g's, respectively. Structural tests showed the strength of a complete wing (spar, ribs, D-tube, and trailing edge spar) to be 33% greater that the main spar alone. This resulted in a strength correction factor of 1.33. A factor of safety of 1.5 was then removed from the corrected failure strength. For the cruise velocity of 105 ft/s provided by performance group, maximum dive velocity is calculated to be 157.5 ft/s from Raymer[1].

5.3.2 Wing Attachment

For ease of transportation, the wing was designed to be removable. Two methods were considered for joining the wing to the fuselage. The first was a fuselage mounted, epoxy reinforced, cardboard tube that accepted a wing mounted aluminum tube. This design allowed both wing halves to be detached from the aircraft. Due to the size of the aluminum tube needed to withstand the shear force and bending moment at the wing root, this was a very heavy design. The other method considered involved a one-piece wing with the main and rear spar carried through the fuselage. The plywood reinforced spars slid into reinforced "saddle" bulkheads in the fuselage and bolted into place. This method is used due to its considerable weight savings compared to the aluminum tube approach.

The strength of the carry-through spar design was highly dependent on proper transmission of load between the spar and "saddle" bulkhead. To that extent, a high level of precision wais required in the fit of the two components so there was a continuous contact area between the spar and bulkhead. For the negative g-loading case, as in a sudden drop in altitude, the bolts must each be able to resist a shearing force of 27 pounds as they were the primary load-path components between the fuselage and the wing for negative g-loading.

5.3.3 Landing Gear Design and Placement

Placement of the landing gear for the aircraft is shown in Figure 5.3.2. The tail-dragger configuration offers a weight savings of 0.6 pounds over a tricycle configuration and produces less drag. This led to its selection for use on the aircraft. Layout of the landing gear was designed using ground stability, tipback, and over-turn requirements given in Roskam.[8] The main gear height of 11.5 inches provided a four-inch clearance between the propeller and the ground when the aircraft centerline was parallel to the ground. The stance of the main landing gear was 16 inches and corresponded to a stable overturn angle of 58 degrees. Tendency for the aircraft to overturn during a taxiing turn was decreased as a result. However, ground loop behavior was unstable as it is for all tail wheel configurations.

Due to the tall landing gear height required for propeller clearance, no commercially available landing gear could be found. As a result, custom landing gear had to be fabricated. Materials considered included $^1/_8$" by 2" aluminum bar, $^3/_{16}$" and $^1/_4$" piano wire, and composites. The aluminum was very easy to bend into the desired shape, but could not support the expected weight of the aircraft. Piano wire proved to be too difficult to bend into the desired shape and thus was eliminated from consideration. This left composite material construction for the landing gear. Using carbon fiber and fiberglass cloth, a symmetric lay-up was created. Carbon fiber was used for the outer layers were tensile and compressive stresses were expected to be the greatest, while fiberglass served as a lightweight stiffener in the central layers. The landing gear attached to a plywood plate that was braced by the upper battery compartment longerons and the second bulkhead. Loads from the landing gear are transmitted primarily into the second bulkhead and distributed through the fuselage keel and secondarily into the longerons were it was transmitted into the first bulkhead and firewall.

5.3.4 Final Weights and Balance

Final aircraft weight was found by summing the major components of the aircraft. Tables 5.3.1 to 5.3.3 list these components and their corresponding weights and total weights and balance is given in Table 5.3.4. Weight of items such as the radio gear and the propulsion system were found by simply weighing the components. The weight of the structural members were calculated from component volumes, obtained from the building plans and material densities found in MatWeb.[9] The estimated empty weight of the aircraft was 13.34 pounds.

Location of the center of gravity was determined by multiplying each component weight by the distance of the component from the main spar. Payload is evenly distributed fore and aft of the center of gravity so the location of the center of gravity will vary little between loaded and unloaded flights. Location of the center of gravity is 1.08 inches ahead of the main spar with the battery pack centered at the main spar. The battery pack can be moved forward or aft for precise control of the location of the center of gravity.

5.3.5 Final Structure

The structure of the aircraft utilized I-beams as the main load bearing components. The fuselage keel and the wing main spar are both I-beams featuring carbon fiber top and bottom flanges and a

vertical-grain, balsa shear web. This is a very efficient design, providing high strength at low weight. There is some concern about the cutouts in the bottom of the keel that accept the saddle bulkheads for the wing attachment. Although these notches compromise strength in the keel, ample reinforcement have been made to the I-beam in the form of plywood doublers.

5.4 Stability and Control Analysis

For the detail design stage, the focus of the stability analysis was to determine the size of the tail. In order to minimize Rated Aircraft Cost, a short tail moment arm was desired. However, a stability-and-control analysis was performed to determine the minimum tail size and tail moment arm length while maintaining reasonable stability.

The preliminary analysis found that the static margin, of an aircraft with a 2.25 ft^2 horizontal tail and a tail moment arm of three feet, was approximately 11%. However, this calculation ignored the propulsive effects. The static margin was recalculated including the propulsive effects. This analysis revealed that this configuration would have an 8% to 9% static margin. This was considered unacceptable. Therefore, calculations were done to determine how much the tail arm needed to be extended to produce a desirable static margin. A desirable static margin was defined to be approximately 12%. It was determined that if the tail length was increased by six inches, the static margin would be acceptable. It was found that moving the cg of the aircraft forward by one-half inch could also increase the static margin by 4%. During flight-testing, the batteries can be moved forward to increase stability if required.

After the tail configuration was finalized, calculation of trim curves was performed. The initial trim curve calculation ignores power effects. A full span elevator of 33% of the tail chord was assumed. The analysis was conducted using the method in Raymer[1]. The power-off trim curves are presented in Figure 5.4.1. It was discovered that only three degrees of deflection would be required to achieve C_{Lmax}. This was a concern because the aircraft would be very sensitive to elevator deflection, so further analysis was conducted to see if this size elevator would be acceptable. Using Raymer's[1] method, the trim curves were recalculated to include the propulsive effects. These power-on trim curves are presented in Figure 5.4.2. Propeller downwash effect was desirable because it reduced the sensitivity of the elevator. The final horizontal tail configuration data are presented in Table 5.4.1.

The lateral stability analysis of the aircraft also employed the method found in Raymer[1]. First, the power-off value for $C_{n\beta}$ was calculated. Raymer suggests that for a low subsonic Mach number the value of power on $C_{n\beta}$ be 0.05 per radian. The power-on $C_{n\beta}$ calculated to be consistent with the value recommended by Raymer.[1] Due to time constraints lateral trim curves were not generated. Historical data from previous UIUC competition aircraft was used to size the rudder of the aircraft. The final vertical tail configuration data can be found in Table 5.5.1.

5.5 Performance Analyses

Three studies were carried out as part of the performance analyses. The first study was of the fuselage shape to determine best way to layout the payload, batteries, electronics, etc. Second, a study

was carried out to select the best wing airfoil. Finally, the performance of the competition aircraft was studied.

Data for each of the fuselage configurations designed by aerodynamics were entered into the performance code using their estimated individual weights and drag coefficients. Table 5.5.1 shows the results of this study. Although Fuselage 1 was more slender than Fuselages 2 or 3, the corresponding drag polars, resulted in Fuselages 2 and 3 performing similarly and better than Fuselage 1. Ultimately, Fuselage 3 was chosen because it allowed for better battery cooling, easier construction, and decreased length and cost.

The performance code was run with each of the airfoils selected in the aerodynamic studies (i.e., Clark Y, RG14, S8036 and SG6041) for varying geometries using the data provided by aerodynamics. Since the RG14 airfoil did not satisfy the previously discussed takeoff requirements (Section 4.3) for any geometry used in this study, it was removed. The results of this study are presented in Figure 5.5.1. The dashed vertical lines in the figure indicate the maximum area and aspect ratio combinations for a given percentage thickness that structures found possible to fabricate. The solid vertical lines indicate the aspect ratio at each planform area used in the study. Each airfoil presented here has a different thickness, indicated in the legend. The SG6041 and the Clark Y exhibited very similar behavior with respect to Mission Time*RAC. The minimum Mission Time*RAC occurred at a very similar area and aspect ratio combination for all three airfoils. The minimum value of Mission Time*RAC for the S8036 was the highest, but only by 1%.

Since the S8036 incurred only a 1% penalty in Mission Time*RAC for its large thickness, it was decided that using it to increase the aircraft's overall survivability and robustness would be a beneficial trade off. To confirm this, the airfoils were re-run in the performance code to take advantage of the added strength. To do this, the turns in the mission were carried out at the maximum allowable g loading for each airfoil. This value was calculated from structural tests. Figure 5.5.2 shows that the Mission Time*RAC decreased because the S8036 resulted in a much stronger wing than the other airfoils. The format of Figure 5.5.2 is the same as Figure 5.5.1 and shows that S8036 will indeed perform better than the other airfoils by utilizing its added strength. The SG6041 was left out of Figure 5.3.2 because it was even thinner than the Clark Y, which resulted in higher Mission Time*RAC values. In addition, geometries that could not satisfy the takeoff requirement were left out of this figure. Therefore, the choice to use the S8036 with an area of nine ft^2 and an aspect ratio of nine, was made taking into account the fact that using the S8036 resulted in a much stronger aircraft with, at worst, a Mission Time*RAC penalty of 1%.

The results of the wind tunnel tests of the propulsion system presented the opportunity of comparing the propulsion model generated by MotoCalc™ to one developed using the wind tunnel data. The results are presented in Table 5.3.2. These results represent the final performance estimates for the competition aircraft incorporating the final estimated aircraft weight and aerodynamics. The two models agreed extremely well with respect to takeoff distances, generating confidence that the aircraft will have

no trouble making the 200 ft. limit. However, both mission time and energy usage were quite different. The MotoCalc™ model predicted higher energy usage and mission time. Since the aircraft was designed using the conservative MotoCalc™ models, the performance predicted by them is considered a "worst case" scenario. Designing in this way insures that the aircraft will have sufficient energy to complete the mission while performing at its maximum level.

5.6 Detail Design Summary

The final aircraft configuration is shown in Figure 5.6.1. The final airplane has a wing area of nine ft^2 with an aspect ratio of nine. The overall aircraft length is 6.58 ft. The loaded weight of *Illiniwek 1* is 22.64 lbs. The batteries are situated below the payload within the fuselage. The aircraft uses an Astroflight™ Cobalt 60 motor with a 20x16 propeller. The aircraft is expected to complete the mission in approximately 218 seconds and use approximately 1400 mAh of battery energy. The Rated Aircraft Cost is expected to be 12.97 thousand dollars. This is shown in Table 5.6.

Table 5.1.1: Results of Fuselage Cross-sectional Configuration Study

Cross-section	A_{max} (ft^2)	Length (ft)	S_{wet} (ft^2)	ΔC_{Dofuse}	Aircraft C_{Do}
1	0.310	6.33	12.45	0.0075	0.0281
2	0.345	6.00	12.76	0.0078	0.0284
3	0.338	6.00	12.10	0.0074	0.0280

Table 5.1.2: Final Parasite Drag Buildup of Major Components

Component	ΔC_{Do}	% of total C_{Do}
Wing and Horizontal Tail	0.0177	65.8%
Vertical Tail	0.0021	7.8%
Fuselage	0.0042	15.6%
Landing Gear	0.0029	10.8%
Total	0.0269	

Table 5.2.1: Battery Specification Comparison

	Average Capacity (mAh)	Weight/Cell (oz)	Capacity/Weight (mAh/oz)	Dimensions (in)
SR Batteries 2000 Max	2000	1.86	1075.3	.9 x 1.69
SR Batteries 2400 Max	2400	1.9	1263.2	.9 x 1.69
SR Batteries 3000 Max	3000	2.47	1214.6	1.02 x 1.97
Panasonic P-170SCRP	1700	1.73	982.7	.91 x 1.69
Panasonic P-200SCP	2000	1.83	1092.9	.91 x 1.69
Sanyo N-1900SCR	1900	1.96	969.4	.91 x 1.69
Sanyo RC-2000	2000	2.00	1000.0	.91 x 1.69
Sanyo RC-2400	2400	2.09	1148.3	.91 x 1.69
Sanyo N-3000CR	3000	2.96	1013.5	1.02 x 1.94

Table 5.3.1: Fuselage Weights and Balance Sheet

Item	Unit WT. (lbs)	Qty.	Total Wt. (lbs)	Distance from Main Spar (in)	WT * DIST (in-lbs)
Fuselage					
Firewall	0.12	1	0.12	23.40	2.78
Bulkhead #1	0.02	1	0.02	11.90	0.22
Bulkhead #2	0.06	1	0.06	0.38	0.02
Bulkhead #3	0.04	1	0.04	-0.38	-0.01
Bulkhead #4	0.06	1	0.06	-4.00	-0.23
Bulkhead #5	0.06	1	0.06	-4.50	-0.25
Bulkhead #6	0.02	1	0.02	-12.00	-0.23
Bulkhead #7	0.09	1	0.09	-24.00	-2.10
Bulkhead #8	0.02	1	0.02	-31.25	-0.59
Bulkhead #9	0.01	1	0.01	-38.50	-0.48
Bulkhead #10	0.01	1	0.01	-43.70	-0.27
Bulkhead #11	0.00	1	0.00	-45.40	-0.14
Cargo longerons	0.04	4	0.16	0.00	0.00
Fuselage longerons	0.06	2	0.12	-10.70	-1.24
Keel shearweb	0.16	1	0.16	-9.75	-1.58
Carbon fiber flanges	0.18	2	0.36	-10.70	-3.89
Wing joint keel stiffener	0.09	2	0.19	-1.85	-0.35
Landing gear	0.50	1	0.50	5.10	2.55
Motor and gearbox	1.69	1	1.69	26.40	44.55
Propeller	0.25	1	0.25	29.40	7.35
Speed control	0.12	1	0.12	23.40	2.78
Battery	0.13	38	5.00	0.00	0.00
Receiver	0.14	1	0.14	-25.00	-3.50
Servo battery	0.23	1	0.23	-25.00	-5.78
Rudder servo	0.16	1	0.16	-31.00	-4.96
Elevator servo	0.16	1	0.16	-31.00	-4.96
Miscellaneous*	0.50	1	0.50	0.00	0.00
Fuselage Total Weight (lbs)			10.23		

* Miscellaneous refers to the total weight of minor components such as glue, fasteners, wiring, etc...

Table 5.3.2: Wing Weights and Balance Sheet

Item	Unit WT. (lbs)	Qty.	Total Wt. (lbs)	Distance from Main Spar (in)	WT * DIST (in-lbs)
Wing					
Full wing rib	0.01	20.00	0.13	-2.00	-0.25
Aileron rib	0.01	10.00	0.05	-0.55	-0.03
Front shear web	0.03	2.00	0.06	3.30	0.21
Main spar	0.78	1.00	0.78	0.00	0.00
D-tube	0.06	2.00	0.11	3.40	0.38
Rear shear web	0.06	1.00	0.06	-5.00	-0.32
Aileron	0.14	2.00	0.29	-6.25	-1.80
Rib caps	0.00	60.00	0.25	-1.50	-0.38
Aileron servo	0.16	2.00	0.32	-0.25	-0.08
Miscellaneous*	0.50	1.00	0.50	-2.00	-1.00
Wing Total Weigt (lbs)			2.55		

* Miscellaneous refers to the total weight of minor components such as glue, fasteners, wiring, etc...

Table 5.3.3: Tail Weights and Balance Sheet

Item	Unit WT. (lbs)	Qty.	Total Wt. (lbs)	Distance from Main Spar (in)	WT * DIST (in-lbs)
Tail					
Horizontal ribs	0.00	8.00	0.02	-42.00	-0.79
Horizontal leading edge	0.05	1.00	0.05	-38.00	-1.78
Horizontal trailing edge	0.05	1.00	0.05	-46.00	-2.16
Elevator	0.24	1.00	0.24	-47.50	-11.35
Vertical ribs	0.00	7.00	0.01	-41.00	-0.51
Vertiacl leading edge	0.03	1.00	0.03	-38.00	-1.32
Vertical trailing edge	0.03	1.00	0.03	-44.00	-1.53
Rudder	0.03	1.00	0.03	-45.00	-1.41
Tail gear	0.10	1.00	0.10	-45.00	-4.50
Tail Total Weight (lbs)			0.56		

* Miscellaneous refers to the total weight of minor components such as glue, fasteners, wiring, etc...

Table 5.3.4: Summary Weights and Balance Sheet

Total	
Weight Without Payload (lbs)	13.34
Payload Weight (lbs)	9.30
Weight With Payload (lbs)	22.64
Fullly Loaded Center of Gravity (Inches Ahead of Main Spar)	0.05
Empty Center of Gravity (Inches Ahead of Main Spar)	0.08
Reference Point (desired CG (% MAC)	33.30

Table 5.4.1: Empennage Attributes

Attributes	Horizontal	Vertical
Area	2.25 ft^2	1.5 ft^2
Aspect Ratio	2.25	2.6
Taper Ratio	1	1
Airfoil	NACA 0009	
Control Surfaces		
Elevator chord	33% of horizontal tail chord	
Elevator area	.74 ft^2	
Rudder chord	33% of vertical tail chord	
Rudder area	.66 ft^2	
Locations	Measured from propeller face	
Wing MAC	2.38 ft	
Vertical MAC	5.82 ft	
Horizontal MAC	5.82 ft	
Center of Gravity	2.47 ft	
Neutral Point	2.6 ft	
Static Margin no propulsive effects	12%	
Static Margin with propulsive effects	10%	

Table 5.5.1: Fuselage Cross-Section Study

	Time (s)	RAC (hours)	Time*RAC (s*hours)
Fuselage 1	216.28	12.53	2710
Fuselage 2	212.47	12.32	2618
Fuselage 3	212.12	12.32	2613

Table 5.5.2: Final Performance Estimates

	MotoCalc™	Wind Tunnel
Takeoff 1 (ft)	47.7	47.8
Takeoff 2 (ft)	156.0	154.4
Takeoff 3 (ft)	62.8	63.1
Mission Time (s)	214.1	203.5
Energy Used (mAh)	1491.6	1268.2
Mission Time*RAC	2776.9	2638.8

Table 5.6: Rated Aircraft Cost

			Cost
MEW*100	MEW=Empty Weight	13.34	1334
REP*1500	REP=(1+.25*(# of engines-1))*total battery weight	(1+.25*(0))*5	7500
MFHR*20	MFHR=(8*span)+(8*chord)+(3*# of control surfaces)+(10*length)+(10*Vertical surfaces)+(10*Horizontal surfaces)+(5*controllers)+(5*# of engines)+(5*# of propellers)	(8*9)+(8*1)+(3*2)+(10*6.6)+(10*1)+(10*1)+(5*5)+(5*1)+(5*1)	4140
	Total Rated Aircraft Cost	**12.97**	

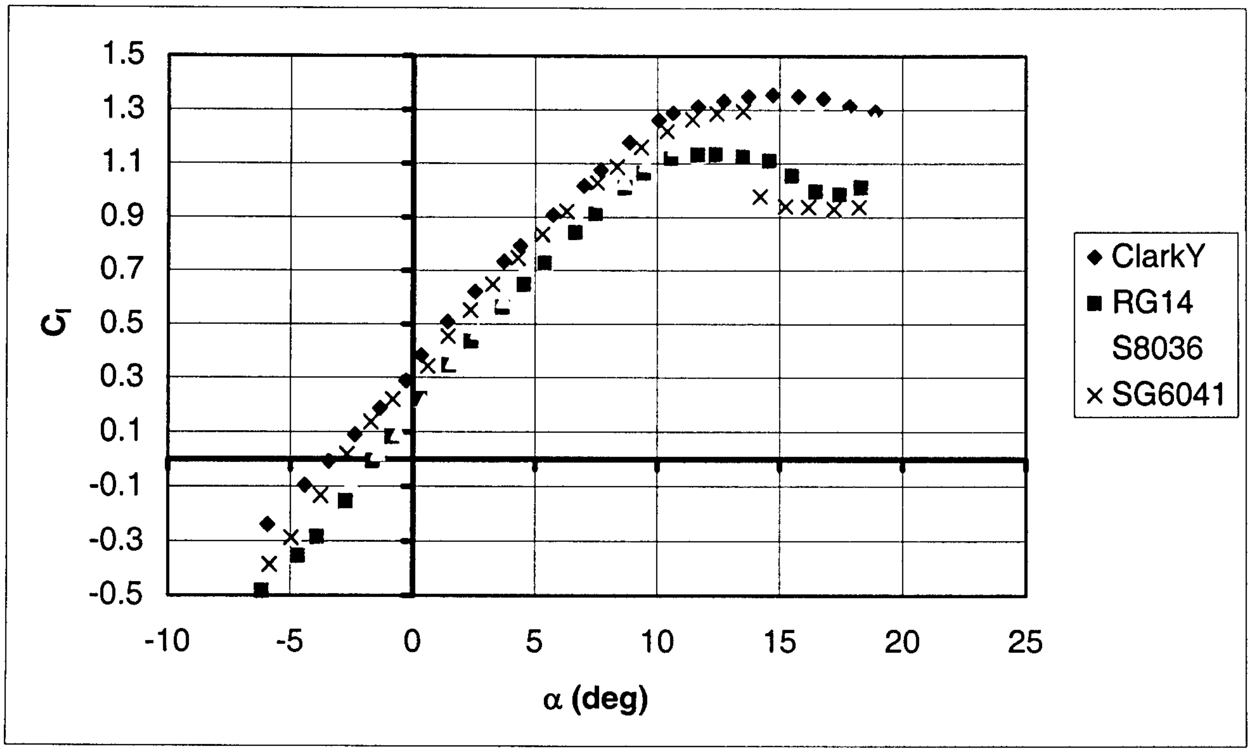


Figure 5.1.1: Lift Curves of Candidate Airfoil Sections at Re = 300000

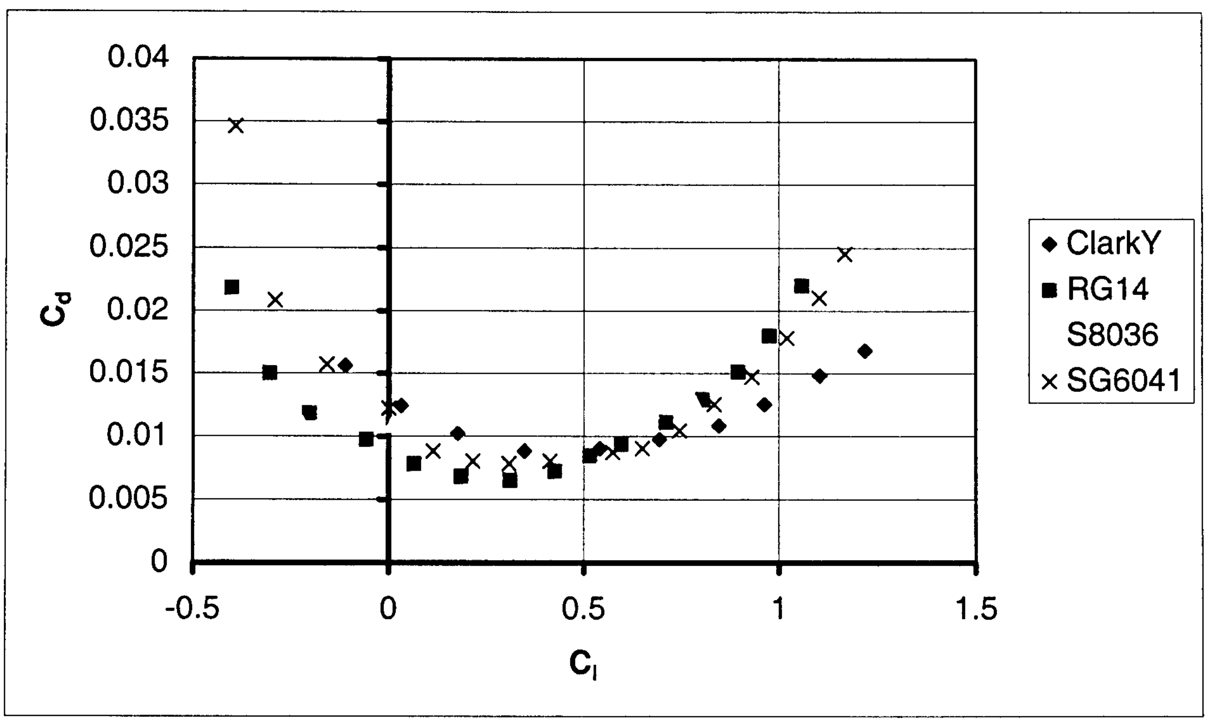


Figure 5.1.2: Drag Polars of Candidate Airfoil Sections at Re = 300000

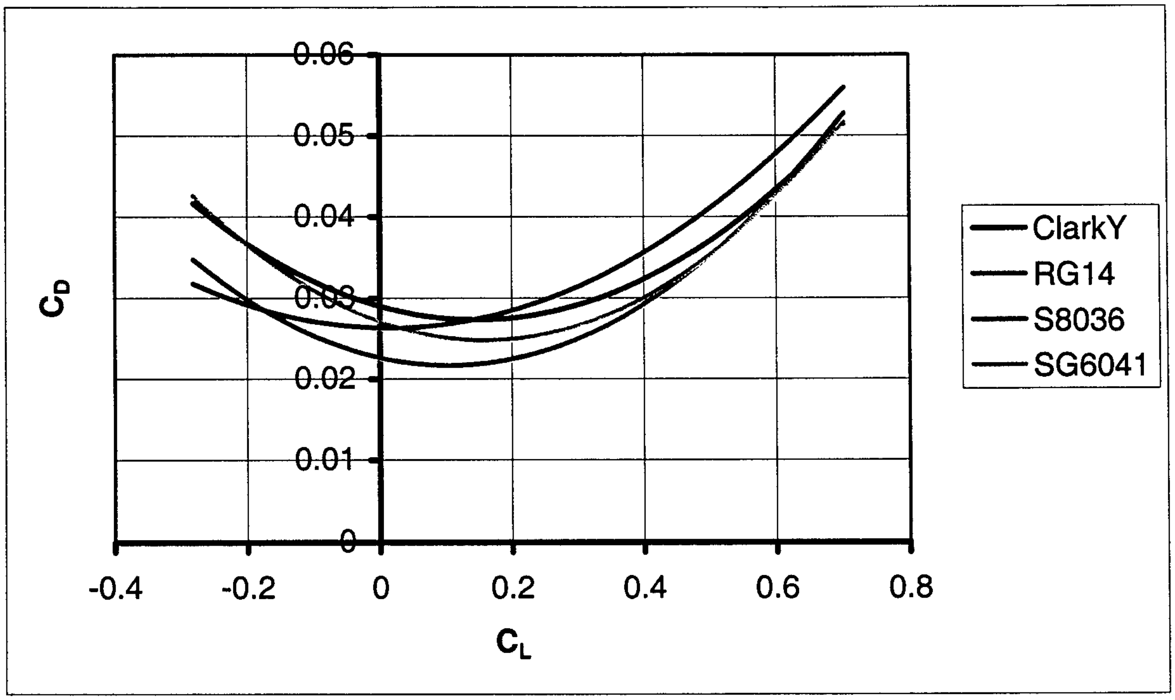


Figure 5.1.3: Untrimmed Aircraft Drag Polars for Varying Airfoil (with S = 9 ft^2 and b = 9 ft)

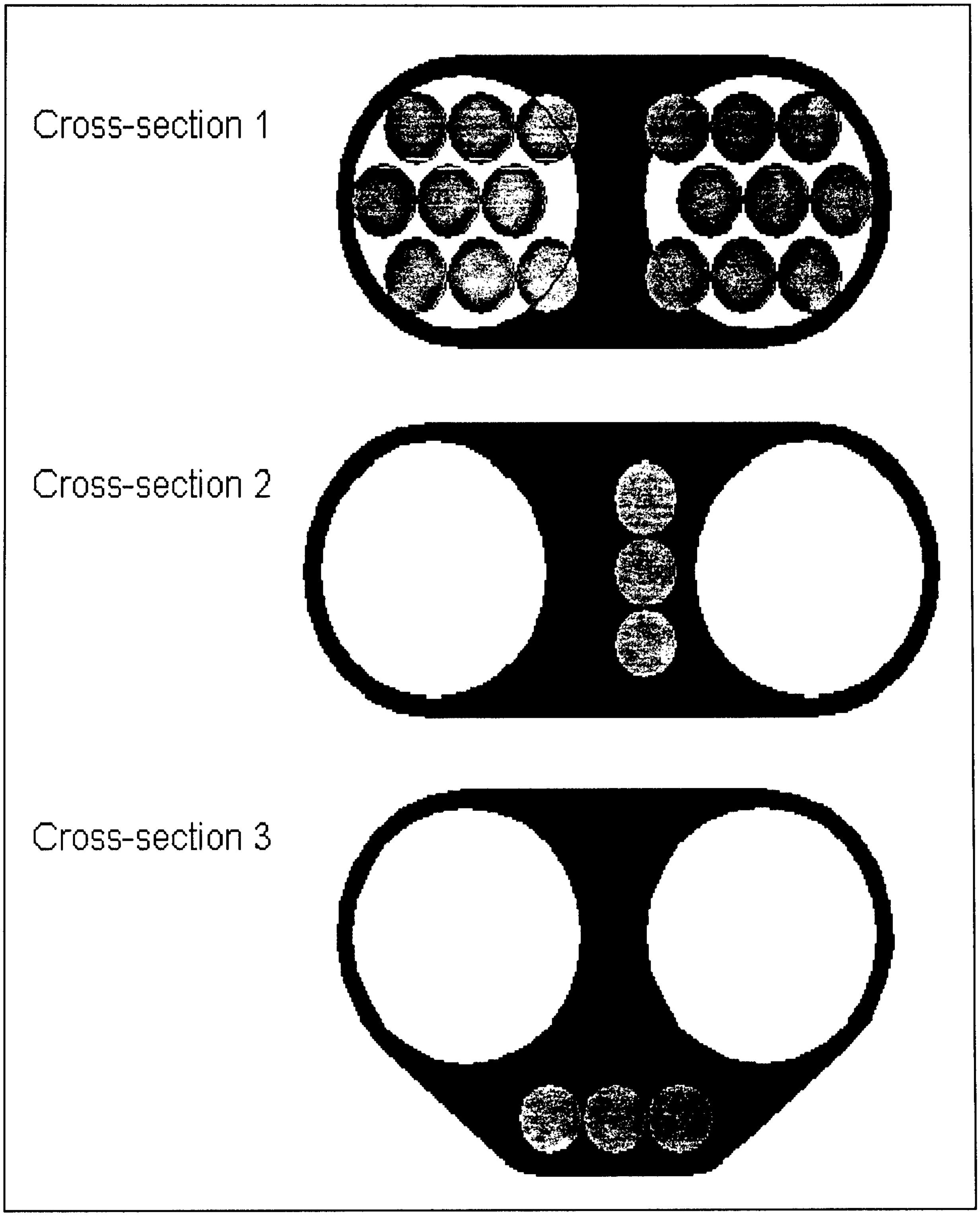


Figure 5.1.4: Proposed Configurations of Batteries in the Fuselage

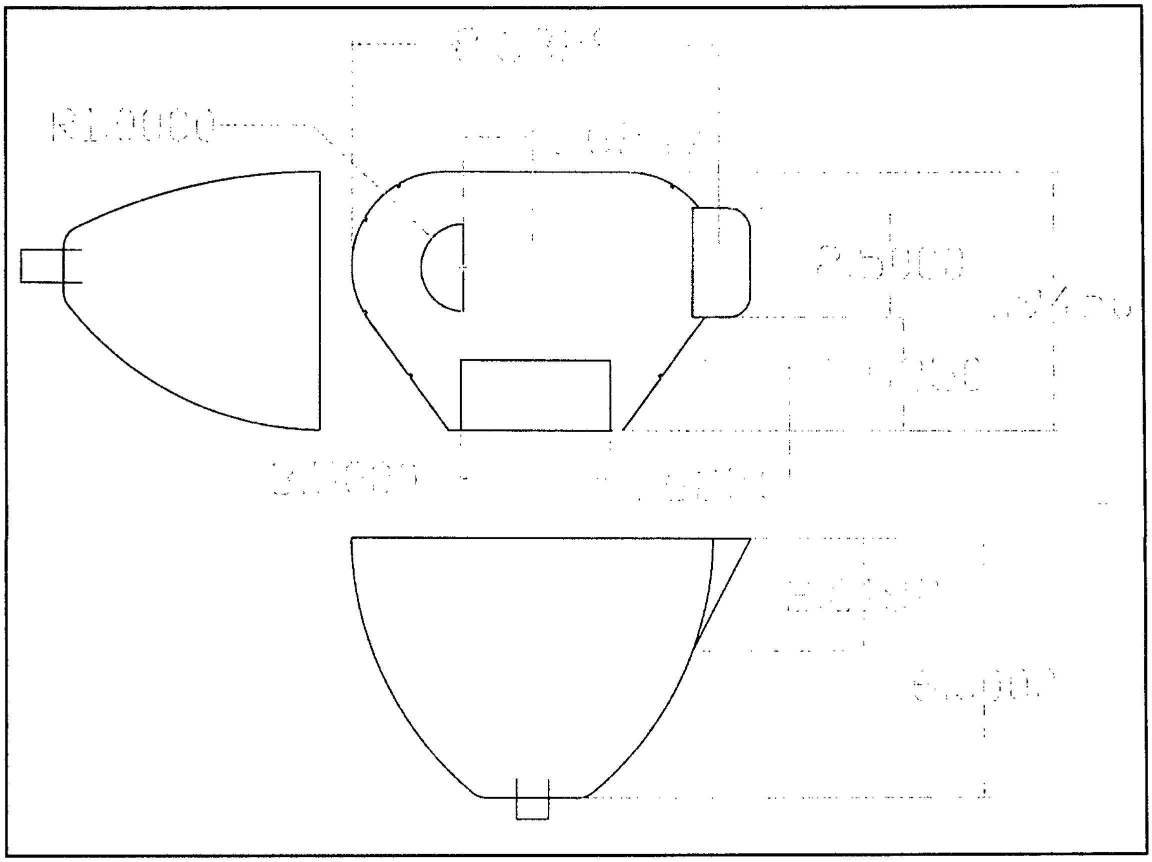


Figure 5.1.5: Cowl Design (dimensions in inches)

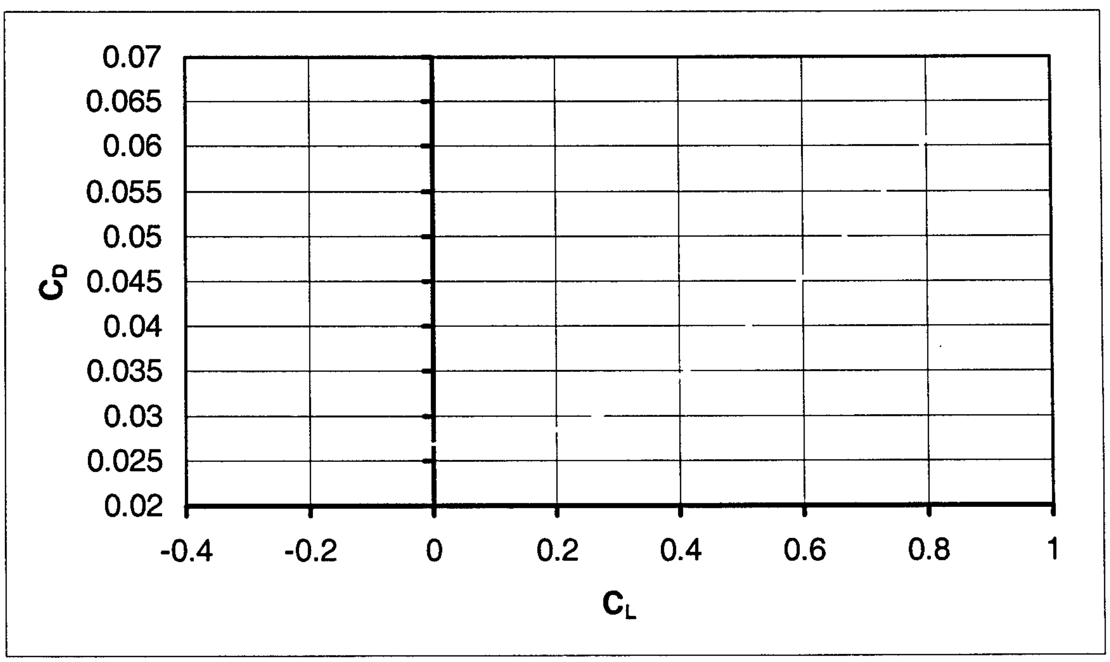


Figure 5.1.6: Untrimmed Aircraft Drag Polar for Final Aircraft Configuration

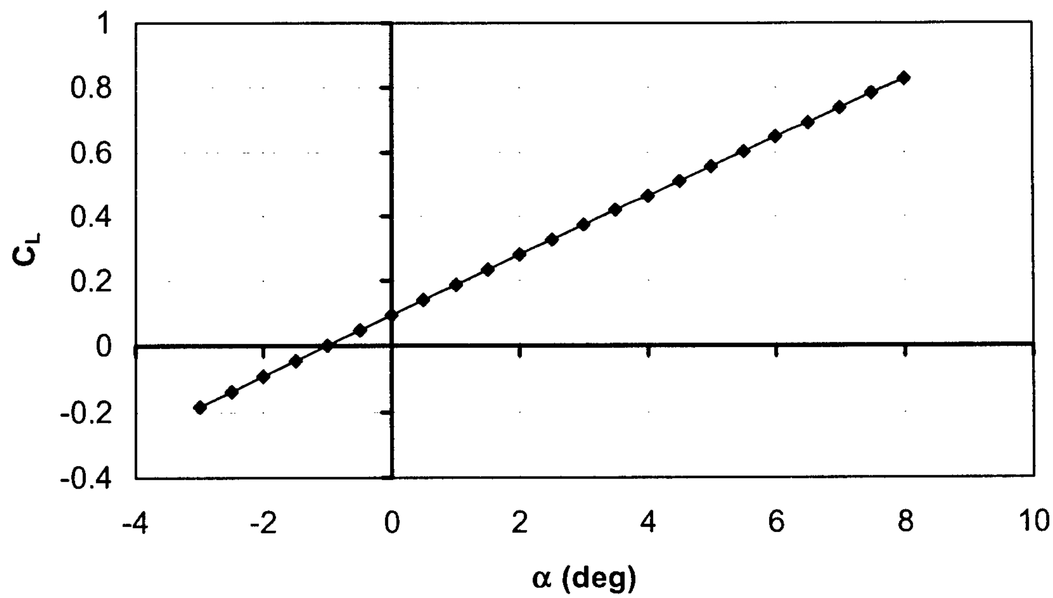


Figure 5.1.7: Lift-Curve of Final Aircraft Configuration (Linear Region estimation by LinAir)

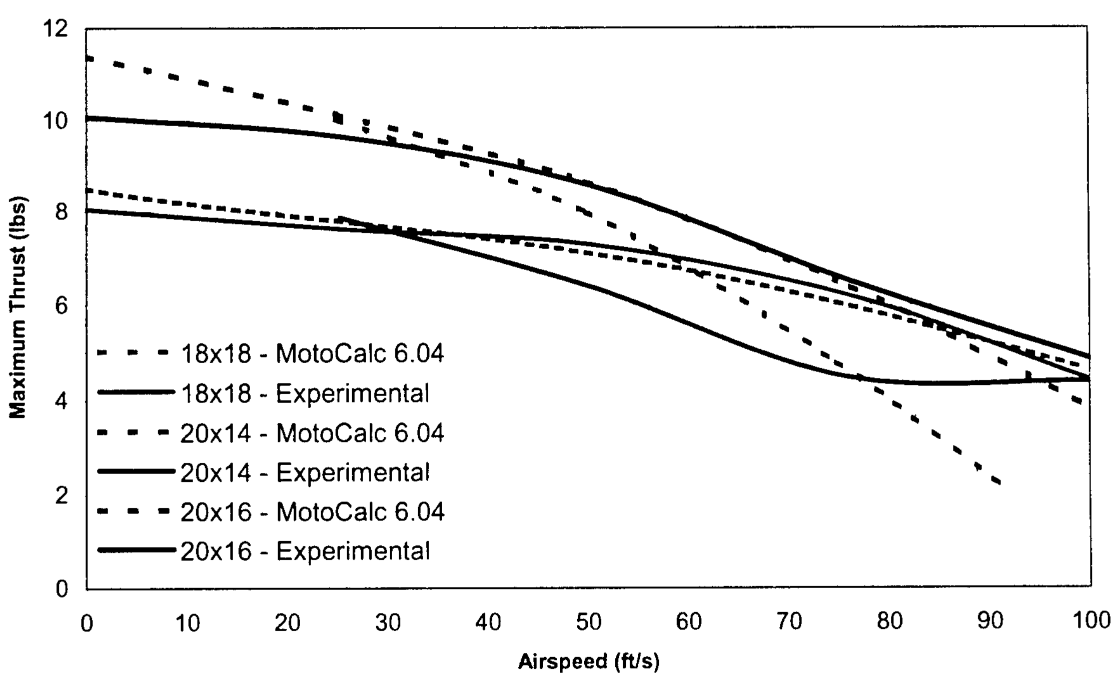


Figure 5.2.1: Maximum Thrust vs. Airspeed

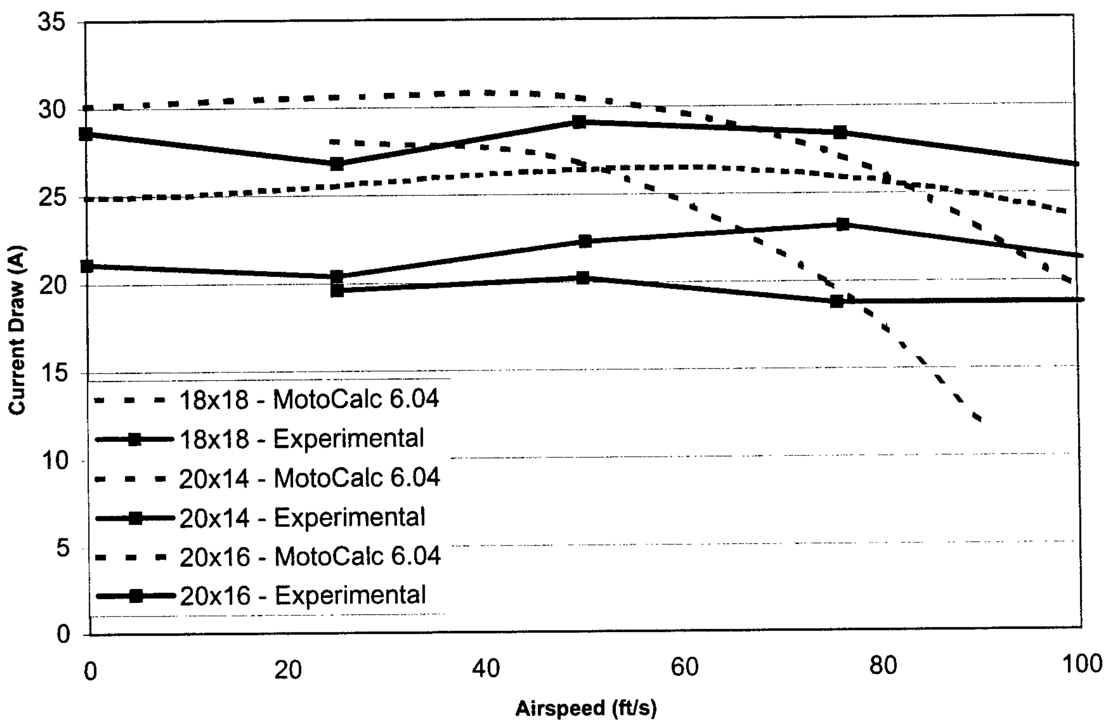


Figure 5.2.2: Current Draw vs. Airspeed at 100% throttle

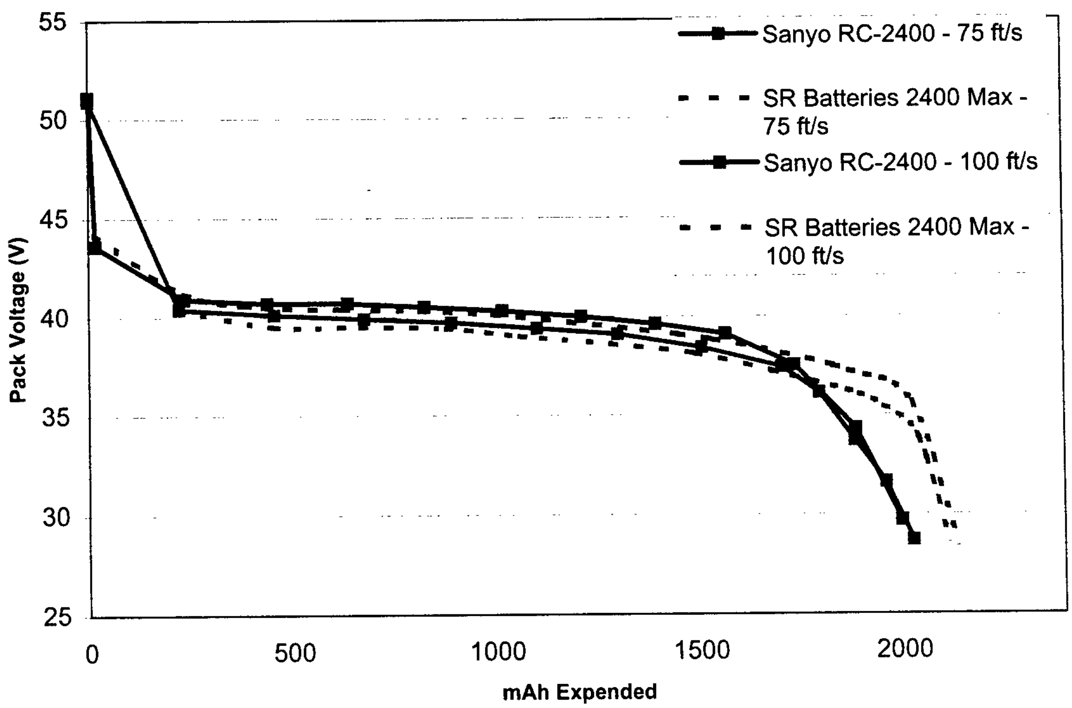


Figure 5.2.3: Battery Pack Voltage Characteristics at 100% throttle with APC™ 20x16 propeller

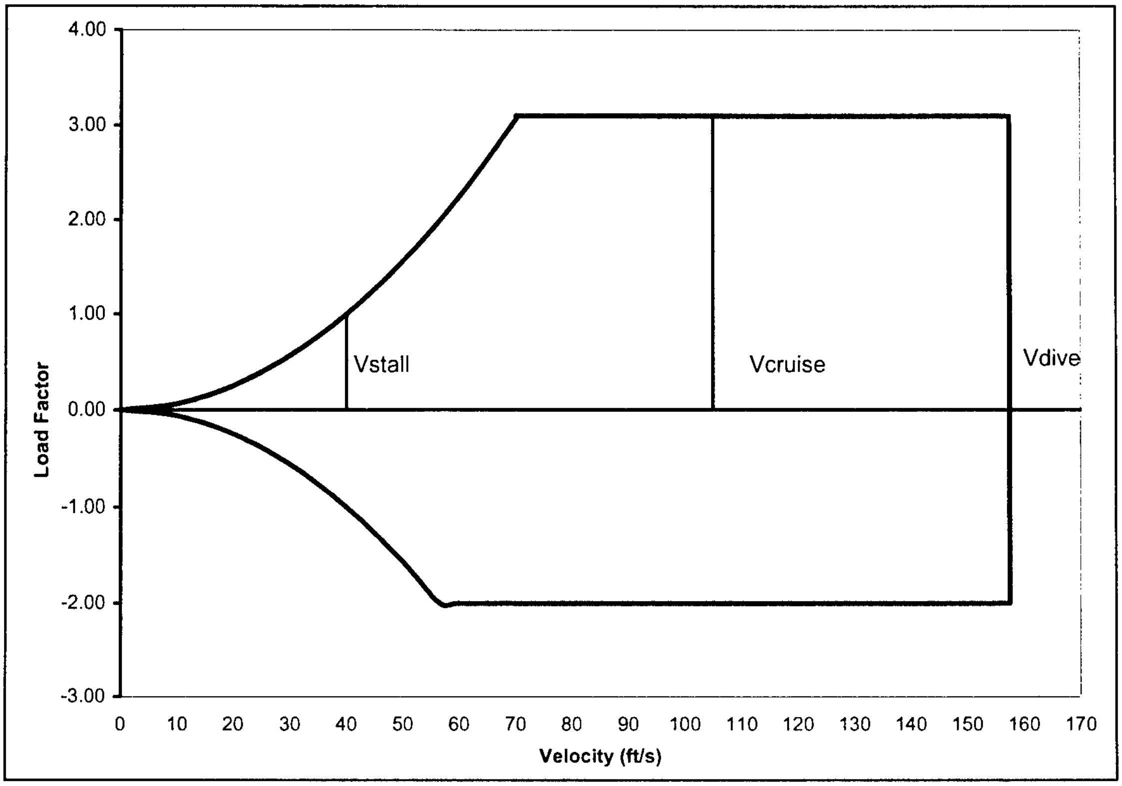


Figure 5.3.1: V-n Diagram

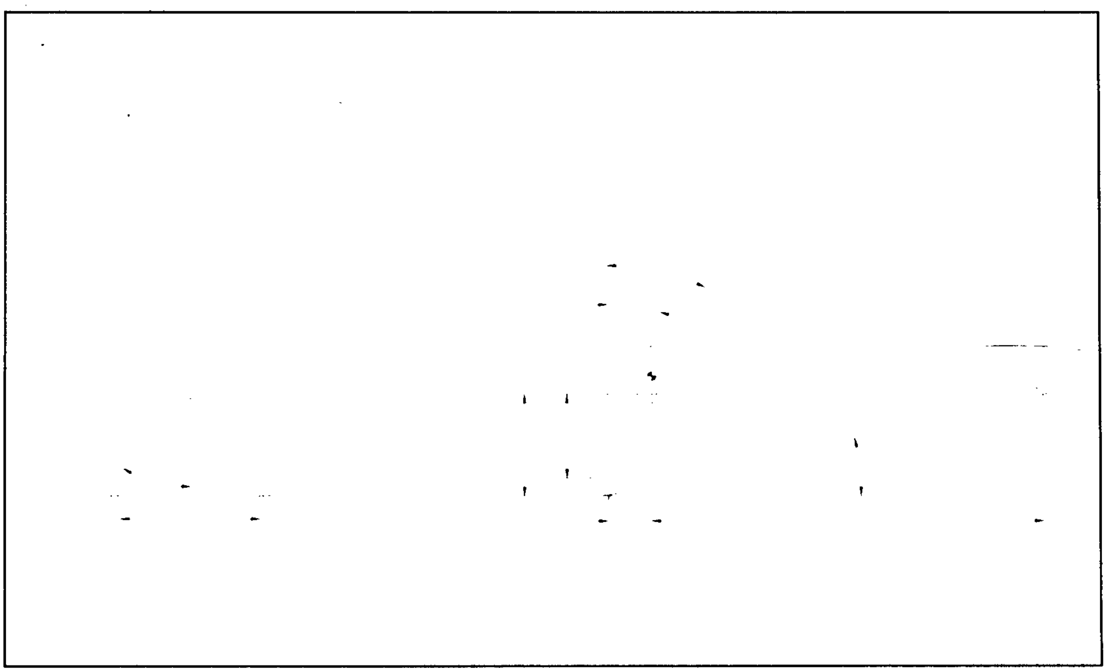


Figure 5.3.2: Landing Gear Placement

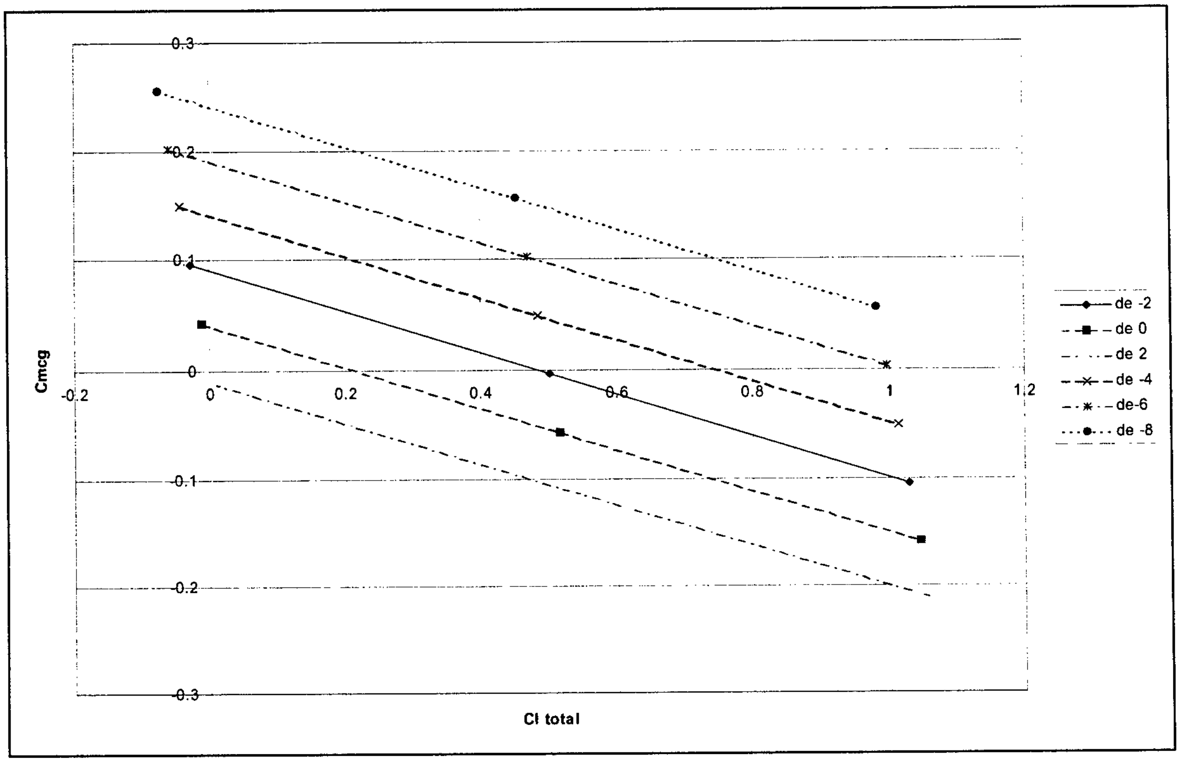


Figure 5.4.1: Power Off Trim Curves

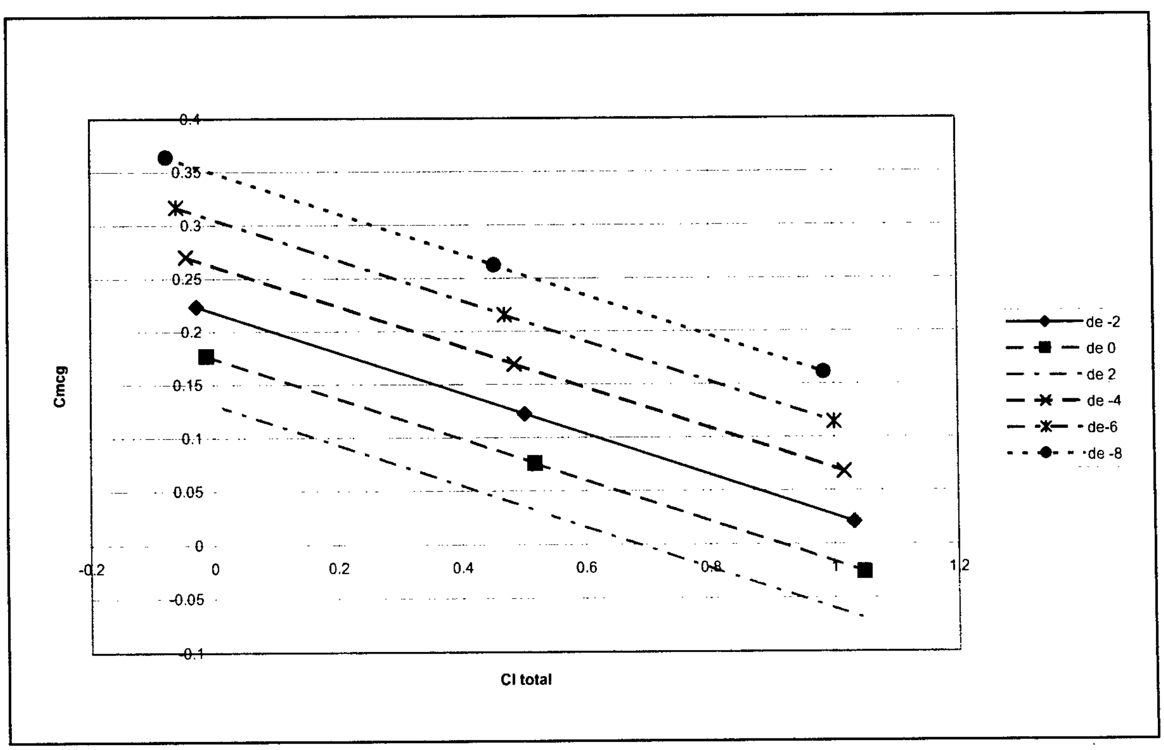


Figure 5.4.2: Power On Trim Curves

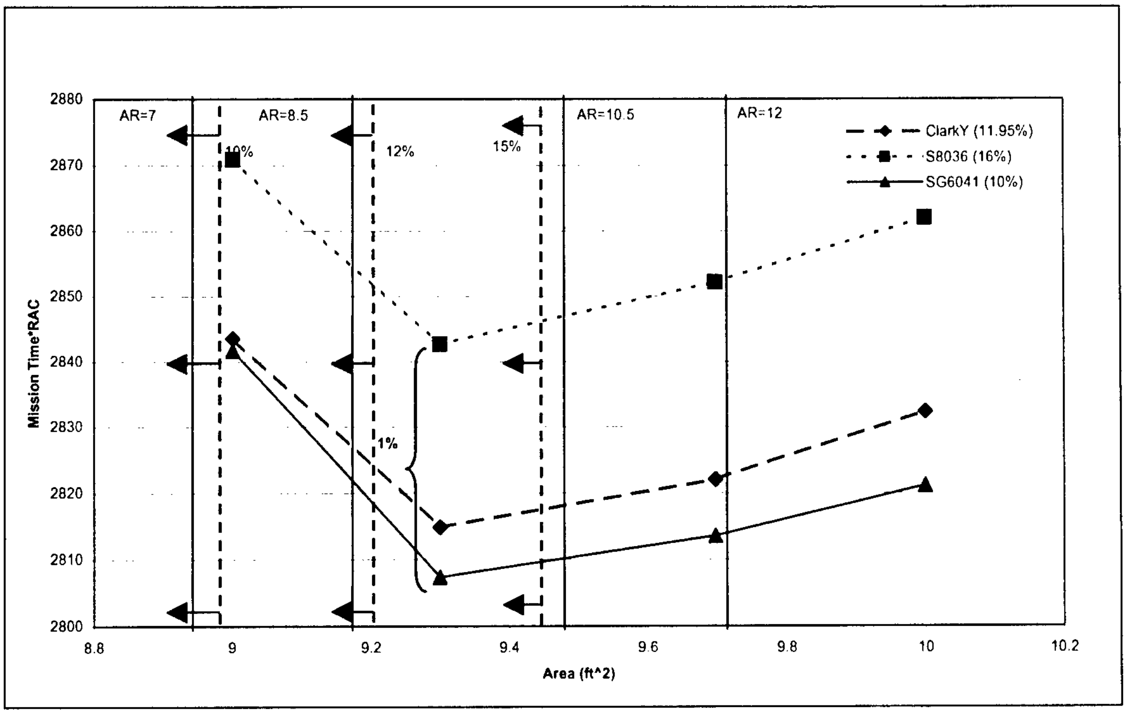


Figure 5.5.1: Airfoil Study

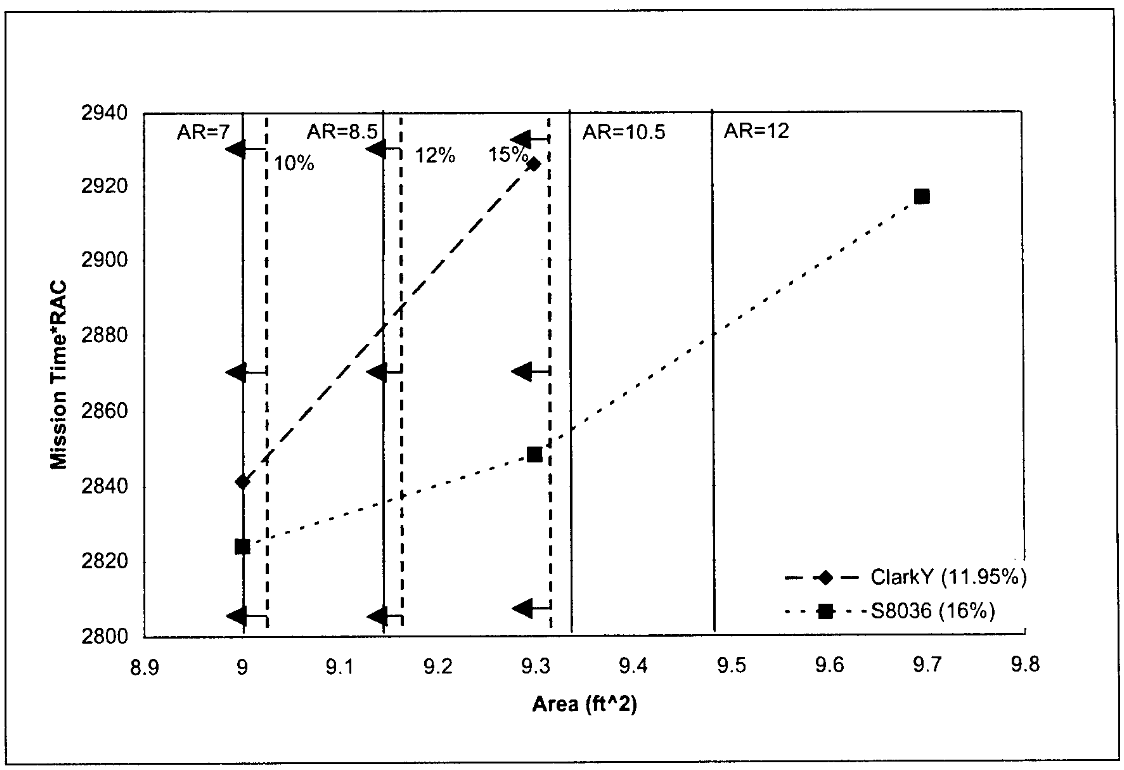


Figure 5.3.2: Airfoil Study with G-Loading

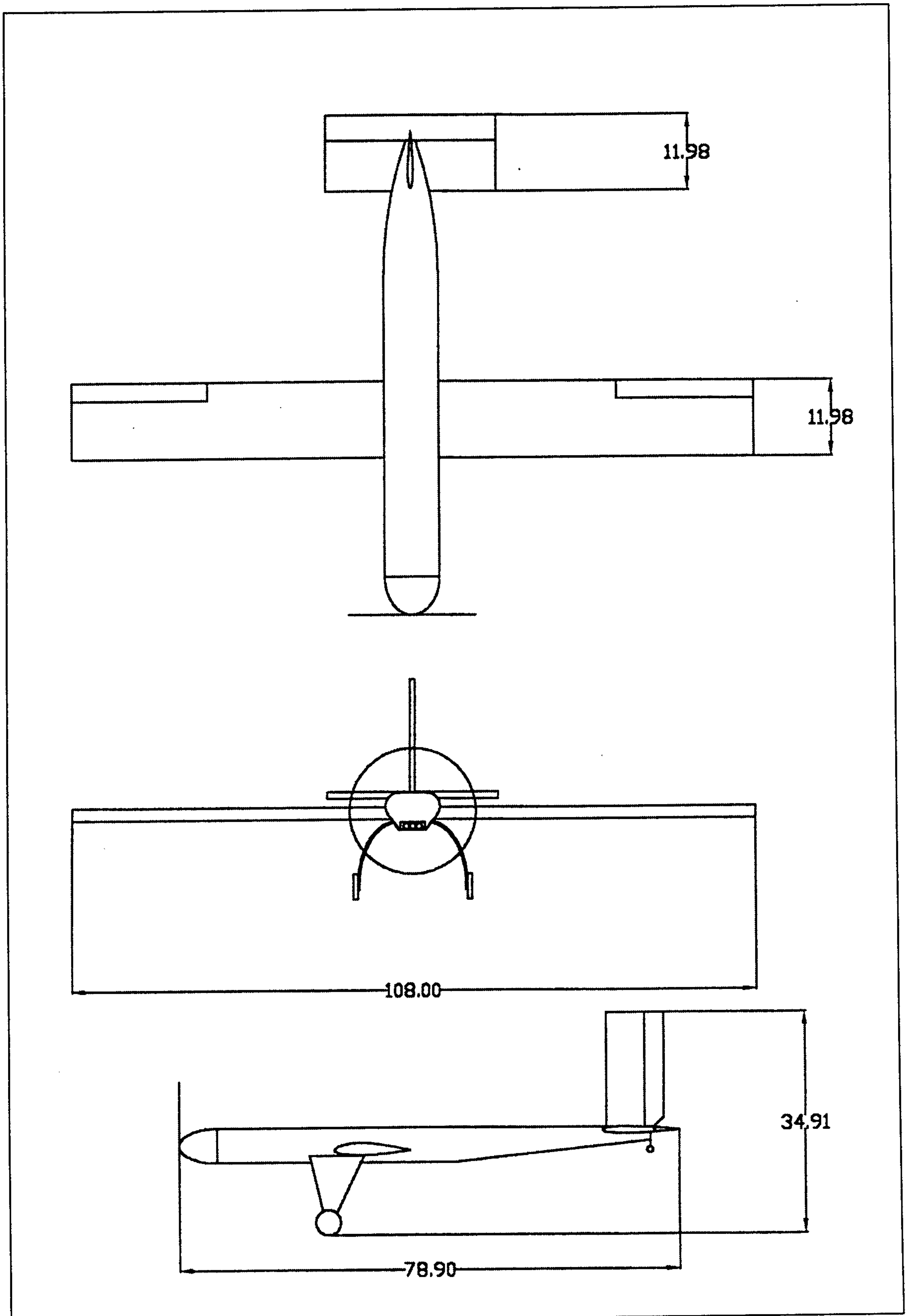


Figure 5.6.1: Aircraft Three View (dimensions in inches)

6.0 Manufacturing Plan and Processes

6.1 Manufacturing Techniques Available

Many potential construction techniques were researched and considered for the aircraft's primary components: the fuselage, wing, empennage, landing gear, and cowl. The techniques ranged from the traditional built-up, wood construction to the more contemporary fiberglass and carbon fiber lay-up.

6.1.1 Fuselage

The techniques considered for building the fuselage were the traditional balsa wood construction with Monokote™ covering, balsa construction covered with fiberglass, and a hollow composite shell. The balsa wood construction included a carbon fiber keel and/or bass wood longerons for rigidity.

6.1.2 Wing

Several construction techniques were considered including a traditional balsa built up wing, a foam core and fiberglass sheeted wing, and a hollow carbon fiber wing. The method of attaching the wing panels to the fuselage could be accomplished using an aluminum tube or using a carry through spar.

6.1.3 Empennage

Similar to the wing, the empennage could be constructed using balsa wood or composites. The horizontal tail could also be permanently attached or removable for transportation. Making the tail removable would be convenient, but also increase complexity and add weight.

6.1.4 Landing Gear

Potential materials for the landing gear included aluminum, fiberglass, or carbon fiber.

6.1.5 Cowl

An engine cowl was necessary to improve the aerodynamics of the aircraft. The cowl would also have to be removable to allow easy access to the motor. The cowl could either be constructed from wood or composites. Wood is easier to work with, but composite would allows more complex shapes.

6.2 Figures of Merit

In order to determine the construction method that would be used on the various components ad Figures of Merit ranking system was employed. The FOMs were:

- Strength per Weight: High strength to weight materials will result in a structurally sound, yet light aircraft.
- Ease of Construction: Each manufacturing technique requires a different skill level necessary to build the design.
- Speed of Construction: The plane must be built in a timely manner and there must be ample time to do flight testing.
- Durability: The aircraft must have the ability to complete many landings and takeoffs, and withstand handling.
- Material Availability: The ease of which the materials can be obtained.
- Material Cost: The monetary expense of the building materials.

Table 6.1 explains the ranking for the FOMs. Each Figure of Merit was assigned a value so that a quantitative comparison could be made. The final rankings are shown in Table 6.2. The data in Table 6.2 reflect the results of testing and analysis done by the structures group.

6.3 Manufacturing Processes

Based on the FOMs a traditional balsa wood construction was chosen for most of the aircraft structures. Once the overall techniques had been decided upon, the detailed construction began. The timeline for manufacturing is outlined in Table 6.3. The timeline was used to keep the manufacturing crew on time in order to assure adequate flight testing could be accomplished before the competition.

6.3.1 Fuselage

The fuselage design utilizes a centerline keel as its primary structure. The keel had a vertical grain balsa shear web and is capped with carbon fiber rods. Plywood shear web reinforcements are used over the wing saddle. In addition to the keel, the longitudinal loads are taken by four wood longerons. These not only support the fuselage, but also form the bottom and outer boundaries of the payload compartment. Bulkheads, which were constructed of either hard or light plywood, are used to maintain the shape of the fuselage. All of the bulkheads contain lightening holes. The front bulkhead is a balsa wood and fiberglass sandwich that is used as the firewall and the forms the forward boundary of the payload compartment. The battery compartment is located below the keel and bottom longerons. Plywood reinforcement is used on the bottom of the fuselage where the landing gear is attached.

6.3.2 Wings

The wing was one piece. The main spar and the rear spar carry through the fuselage. They were secured to bulkheads with nylon bolts. The main spar consisted of an I-beam with upper and lower caps made of $^1/_2$ x $^1/_8$ in. carbon fiber beams and a shear web made of $^1/_8$ in. balsa. The ribs were made of $^3/_{32}$ in. balsa with lightening holes to save weight. $^1/_{64}$ in. plywood was formed around the front of the balsa ribs to create a D-tube leading edge structure. The D-tube provided torsional stiffness to the wing. The ailerons were constructed in a similar manner, utilizing balsa ribs and a balsa leading edge. The wing was covered in Monokote™.

6.3.3 Empennage

It was decided to make the horizontal tail non-removable to save weight. The horizontal and vertical surfaces were constructed of ½ x ½ in. balsa leading and trailing edges. No spar is used in the tail surfaces. Balsa ribs gave the tail surfaces its shape. ¼ x ¼ in. balsa cross bracing is used for torsional stiffness. The elevator and rudder construction was similar to the aileron construction.

6.3.4 Landing Gear

Because all three landing gear construction methods received similar rankings in the FOMs, there was no one clearly optimal method. It was decided to construct the landing gear using composites and also to build aluminum gear to test. In order to obtain the desired strength and flexibility, a combination of Kevlar, carbon fiber, and fiberglass was used. The mold for the landing gear was constructed from foam.

The completed landing gear will contain over 20 layers of composites. The aluminum landing gear was bent from ¼ in. aluminum stock.

6.3.5 Cowl

The cowl has compound curves for good aerodynamic characteristics. Composite construction was necessary to create these curves. A foam plug of the cowl was shaped. The fiberglass and epoxy was formed over the plug. The foam was then removed with acetone, and the fiberglass was finish sanded to shape.

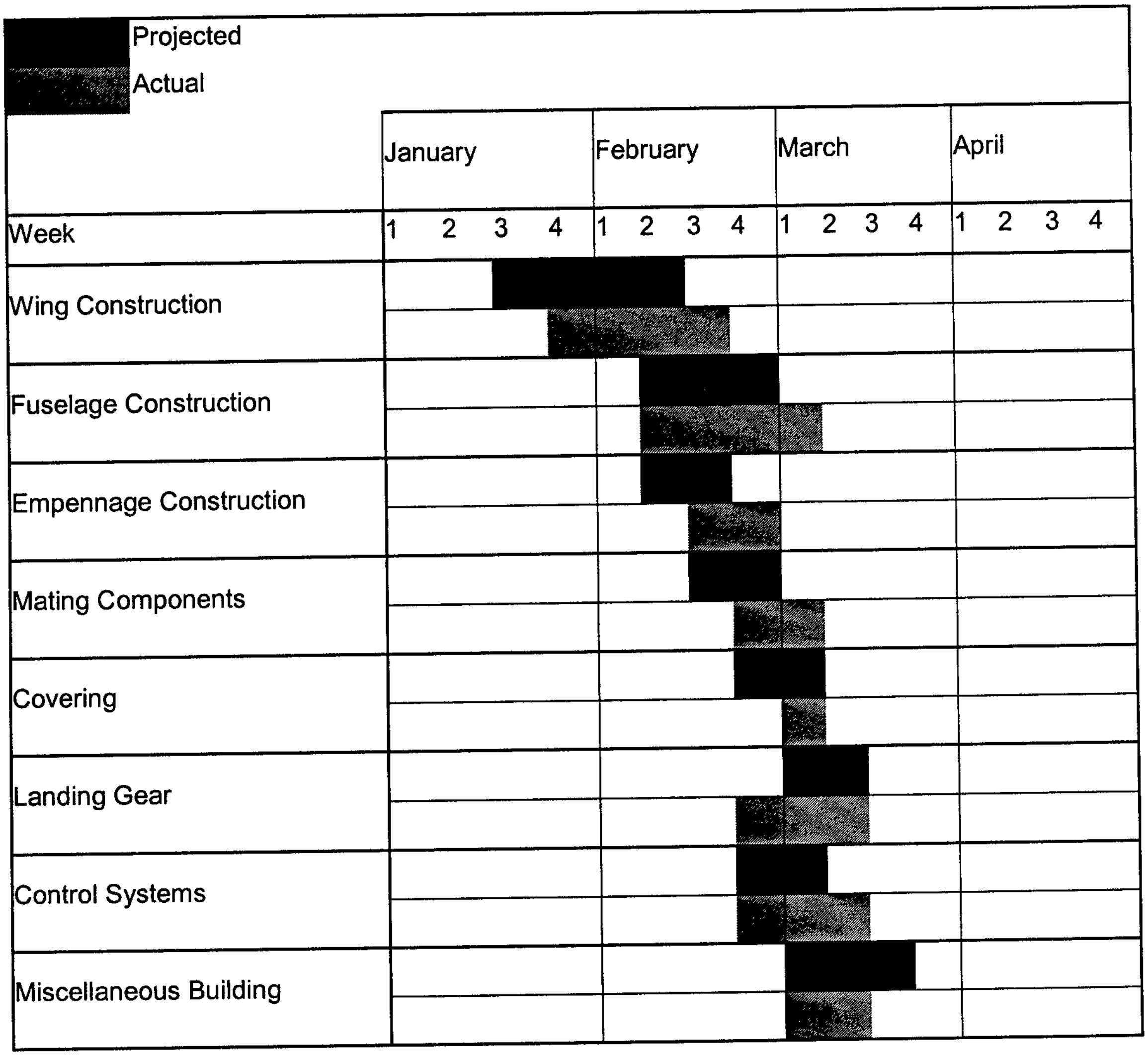
Table 6.1: Construction Figures of Merit

Figure of Merit	Ranking		
	5	3	1
Strength per Weight	High	Moderate	Low
Ease of Construction	Easy	Moderate	Difficult
Speed of Construction	Fast	Moderate	Slow
Durability	Durable	Moderate	Fragile
Material Availability	Readily Available	Moderately Available	Obscure
Material Cost	Inexpensive	Affordable	Expensive

Table 6.2: Aircraft Component Figure-of-Merit Results

		Strength per Weight	Ease of Construction	Speed of Construction	Durability	Material Availability	Material Cost	Total
Fuselage	Balsa/Plywood	4	5	5	3	5	4	26
	Balsa/Fiberglass Shell	4	3	2	3	3	2	16
	Carbon Fiber Shell	5	2	1	4	3	1	16
Wings	Foam Core and Fiberglass Shell	5	3	2	4	3	2	19
	Carbon Fiber Shell (no core)	5	1	1	5	3	1	16
	Carbon Fiber Spar with Balsa Ribs	5	5	4	5	4	3	26
Empennage	Foam Core and Fiberglass Shell	5	2	2	4	3	2	18
	Carbon Fiber Shell (no core)	5	1	1	5	3	1	16
	Balsa built-up	4	5	5	4	5	5	28
Landing Gear	Aluminum	3	3	3	5	3	3	20
	Fiberglass	4	4	4	4	3	2	21
	Carbon Fiber	5	4	4	4	3	1	21

Table 6.3: Manufacturing Timeline



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Università degli studi di Roma
"La Sapienza"

Facoltà di Ingegneria
Corso di laurea in

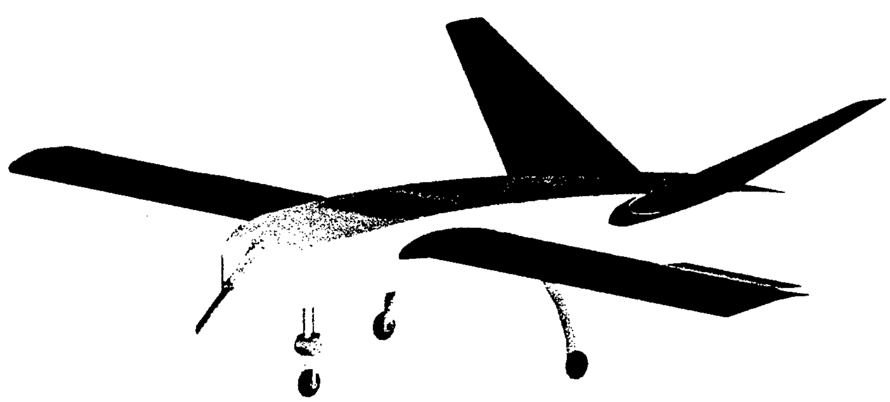
Ingegneria Aerospaziale

AIAA/CESSNA/ONR

Student design/build/fly competition

Design Report

Proposal Phase



...Only who dares flies...

(L. Sepulveda)

Page

1.EXECUTIVE SUMMARY

1.Introduction

In July 2001 a group of students from "LA SAPIENZA " University of Rome met to create a new team for the 2001/2002 AIAA/Cessna/ONR Design Build Fly Competition.

The goal of this new team was to overcome the marks reached last year using lessons learned and making ourselves avail of new precious elements to improve final result.

A large number of persons got interested on our work this year and this gave us the energy and the strength to give the best.

Thanking "La Sapienza" University, "Galileo Galilei" Technical High School and "Sistema Compositi" Srl, we could count on the best knowledge, labs, materials and tools. Very interesting was the collaboration with students of "G. Galilei", who so much helped us in construction, and with staff of "Sistema Compositi" where we had the opportunity to watch and learn how to use last generation machinery in field of composite materials.

To decide the characteristics of our aircraft we focused our attention on three basic elements: Flight Score, Design Report and Rated Aircraft Cost. To obtain the maximum Total score we had to design and build an aircraft combining velocity, reliability, ready to load and unload, sizing of dimensions.

Guided by report needing and by last year experience we divided the design work into three phases:

Conceptual, Preliminary and Detail Design. We started with several ideas and studying them deeper and deeper we arrived to the final configuration.

During the initial phases of the design process we have been continuously asking for suggestions and opinions to people from the RC modelers world. Most of all, we considered of primary importance to check the decision we were making with our "official" pilots, Mr. Luca Friggeri and Mr. Angelo Silvagni, who strictly interacted with us during the conceptual design phase. This is a good rule of behavior we have learned from previous D/B/F experiences. In fact, we wanted to be sure the configuration we were designing had the handling qualities required not to be too difficult in piloting and governing from the point of view of people who would have pilot our UAV.

1.1 Conceptual Design

In this phase, moving from competition rules we divided team in four parts: Aerodynamic, Structural, Propulsive and Mechanic of flight. Each group analysed a part of the design and elaborated different configurations to optimise the aspect of design they were inspecting. Team rated each airplane configuration based on criteria to as the figures of merit.

Then, using the different configurations elaborated by each group we divided the airplane in five parts:

Fuselage, wing, empennage, Landing Gear. Each part had a determined score in evaluating figures of merit. At the end of this phase we chose the most promising configuration to study it deeper in next phases.

1.1.1 Design Alternatives.

The team considered in this phase the following areas: Wing Planform , wing configuration, fuselage configuration, empennage configuration, Landing gear configuration, power plant.

Possible Fuselage investigated were: with two, three or four lines of balls. The wing planforms that were considered consisted of elliptical, rectangular tapered and rectangular not tapered, while the wing profile

investigated were SD7032 and FX63137. Finally we compared the advantages and disadvantages of T-tail and V-tail.

Aircraft properties analysed for all these cases were the weight of the aircraft and the Rated Aircraft Cost .

1.1.2 Conceptual design results

At the end of this Conceptual analysis we had a best configuration for the parameters considered, but this configuration certainly didn't optimise each aspect of the aircraft. This is the reason why in this second phase we analysed other parameters.

1.2 Preliminary Design

In the conceptual design phase, we focused our attention on some parameters and we disregarded some others. In this phase we first of all tested the prototype made on the conclusions reached by the first part of the analysis and then we studied some others figures of merit to analyse some parts of the aircraft.

These Figures of merit were analysed in some new tables trough which we could study the final configuration that suit our exigencies.

1.2.1 Design Alternatives

In this phase we analysed structures and materials of the following parts of the aircraft:

Wing

Fuselage

Empennage

Landing Gear

For each one we studied the best configuration taking in consideration several possibilities.

1.2.2 Preliminary Design Results

At the end of this part of the design we had a configuration and the results of some experiments made on prototype. We only had to make a deeper analysis of components.

1.3 Detail Design

In this phase each group made a job separate from others groups. In this part of the design, in fact, each part of the aircraft had to be analysed by different points of view. In particular we made an Aerodynamic study, a Structural analysis, and the whole system of flight were exanimate by the Mechanic of Flight group.

1.3.1 Detail design Results

At the end of this phase we had a configuration. We were ready to start the final building phase but design team was still at work to operate if the flight test would give negative results.

1.Conclusion

The challenge of this year was to improve last year results. We don't know if the final score will be better or worst but one thing is certain: we gave the best to be a credit to both our Country and our University and the final result make surely us very proud.

2.MANAGEMENT SUMMARY

2.Introduction

By the experience of last year, we learned that only an efficient organization would led us to a good work. This is the reason why our first meeting was focused on the team management.

The team consisted in twenty two people all aerospace engineering students and a pilot. Our aim was to use potentiality of each member of team highlighting every one's skills and making the best of every one's experience. In other words the philosophy of team was to make the best of human resources we had at our disposal.

Although the amount of work and the increase of know-how of our University teams has been a constant trend during the last three experience in the D/B/F competition, nevertheless good flight qualities of the UAV we had already designed and built, our best result in the contest has been far from classifications' top. This consideration drove us to a critical analysis about the organization of the work while we was planning to enterprise a new competition. We noticed that aspects of precedent experiences which could be much improved were essentially of two type: manufacturing and logistical ones. That is, the building solutions, the tools we could use in building, the organization of the team. Actually, as we are saying early, these are aspects which are linked the one to the others.

About the manufacturing problems, last year experience showed composite materials were the best, since the first three UAV in the final classification were built by using this sort of materials. We have to say our wooden products of previous year answered accurately the design directives, emphasizing qualities of the selected configuration, and had a optimum behavior in flight. But they lost. We think most of all because they had two main problems: weight and slowness in building. Which are linked. In fact: using composite materials allow to realize products lighter for at least two reasons. The first is in their higher ratios stiffness/weight and strength/weight. But it is not all: thanks to the fastness in building, with these materials is possible to realize early same prototypes to be tested. In this way is possible to notice that faults which calculations can not relieve, and so to correct them to optimize the UAV behavior. This is not possible with wood: the closed timings of the competitions do not allow to build more than one product, which should show enough safety and reliability so that it has to be necessarily over sized, and so heavier than necessary. And this is the second reason. More over, avoiding to spend too much time in manufacturing would have let us time for a serious session of flight tests, so to optimize configuration and mission management. To resolve mentioned problems, we looked for tools to be able to work with composite materials in a quite cheep way! For major clarity: we did not decide during the first phase of the design each single element of our UAV would have been realized in composite materials: this sort of choice would have been made in next preliminary design; anyway, we individuate at this point two main Figures of Merit we should ever consider in the next (the more lightweight and the fastness in building): and managed a way to reach that targets.

But looking after to previous contests, we noticed also some inefficiencies strictly linked to the team organization. We felt the necessity that responsibility and authority of final decision were delegated to a team chief, clearly recognized by all the team students. Then, we individuate some management principles to be considered to organize team architecture:

- the team chief has to be only one;

- tasks have to be shared to specialized subteams, each one with its own chief;
- the authority should be delegated to the sub-team chief who organizes its group's work and make specific, "local" decisions;
- global decision are made by the team chief with the help of technical "staff tables", specific about the subject is being discussed;
- competition phases are 5: design / build / fly / support / report; the team architecture should be similar to this division. So it should consist of 5 main sub-teams which depend from the directive organ and communicate within them when sitting at the technical "staff tables".

So we designed the architecture of the team, as we are going to explain.

2.1 Organization

Our first step was to divide team in six sub-groups. It was basic to organize each group at the best because the work done by each one was fundamental for the final result.

The sub-groups were:

- Design
- Building
- Flight
- Report
- Funds
- Logistics

2.1.1 Design

This group studied the conceptual, preliminary and detail phase of the Design. To participate at this phase was compulsory for each member of the group because this is first of all a work of engineering design and every one should have a design experience.

The Design group was made up of four sub-groups:

- Aerodynamic
- Mechanic of Flight
- Power
- Structure

Anyway the cooperation between sub-groups was fundamental. For the same part of the aircraft, in fact, we did different studies watching problems from different points of view and interacting to optimise the final decisions.

The leaders of these sub-groups were expert students. In particular they were persons who worked on design in last year competition and had the necessary knowledge and experience to led a group. Of course for each group the leader was a person specialised in that particular subject.

2.1.2 Building

The construction of the aircraft was not compulsory, but it was important that the maximum number of people got involved in this phase because this year we decided to use innovative materials and technologies and we understood that our ambitious program would need a hard work. Besides every one of us understood that this experience would be very formative and for this reason almost each member of the group gave his contribution to the building.

We built the aircraft in two different places. For the composite parts we could count on the collaboration of "Sistema Compositi Srl" Situated in Colleferro, a city far about sixty miles from Rome. This firm is in the forefront in the construction of composite pieces and they gave us the facilities and materials we needed for our work. We had to organize the travels and the shifts. Groups of three or four persons went to "Sistema Compositi" every day for a couple of months and worked on the aircraft parts. We coordinated our efforts whit students of "Galileo Galilei" Technical School were we constructed the other parts of the aircraft. Collaboration whit "G. Galilei" started three years ago and the help they give us was fundamental.
The aircraft was terminated at the beginning of March, but the work of this group was absolutely not finished because they provided even for tests on prototype and on the aircraft itself and for the last touching up before leaving for U.S.A.

2.1.3 Flight

From last year experience we learned that during the competition it was very important that the group was coordinated and that each member of the group had a specific rule and new exactly what to do. It was the main task of the master of this group to train the team for the competition and assign to each member a explicit mission.
We immediately decided the leader of this group, but the rest of the group was decided at the end of the design phase when we decided who would go to USA for the competition.
Another task of this group was to organize the flight test we did in Italy in Fiano Romano.

2.1.4 Report

The report is one of the most important part of the competition, it has a big influence on the final score and must be as much detailed as possible.
Once again learning by last year experience we began writing report as soon as the design phase started.
From the first meeting we nominated a master for the report who recorded every discussion made or decision took during the meeting. In this way at the end of design phase we had a detailed description of the path followed to get to the final configuration.
The master was assisted by some other reporters who while the design phase developed had the task to write a report of their experiences.

2.1.5 Funds

University "La Sapienza" of Rome allocated a fund for this project but we had to manage these funds and be very careful that the costs didn't overcome the fund we disposed of. Besides someone had to coordinate relationships whit the administration department of engineering faculty.
Another task of this group was to find eventual sponsors for our project

2.1.6 Logistics

logistic group is a news. Last year we had many difficulties in managing our facilities and human resources. The problem was basically that we didn't create a figure who coordinated activities of team. This year we had a person who coordinated relationships, sponsor and marketing and a person who organized the travel;
To manage the laboratories (turns and work)we nominated a Laboratory Master. Finally a person busied whit victually.

2.2 Institutional figures

Two main institutional figures follow us in developing the design. They were an "Advisor" and a "Team Head".

2.2.1 Advisor

The advisor is a lecturer whose duty is to follow our job, super visioning the whole work and highlighting the fallings and the winnings of the team. This task was carried out by Guido De Matteis, full professor of Mechanic of Flight" by our faculty.

2.2.2 Team Head

The Team Head is a teacher of the Technical school. His function is to coordinate us from the organization point of view. He is a kind of super visor who checks everyone's work. Besides he organised the cooperation with students of the High school, and permitted us to use the laboratories of the school where we worked. This person was Giorgio Sforza Aeronautical Engineer, he collaborates with the university as expert of Aeronautical construction.

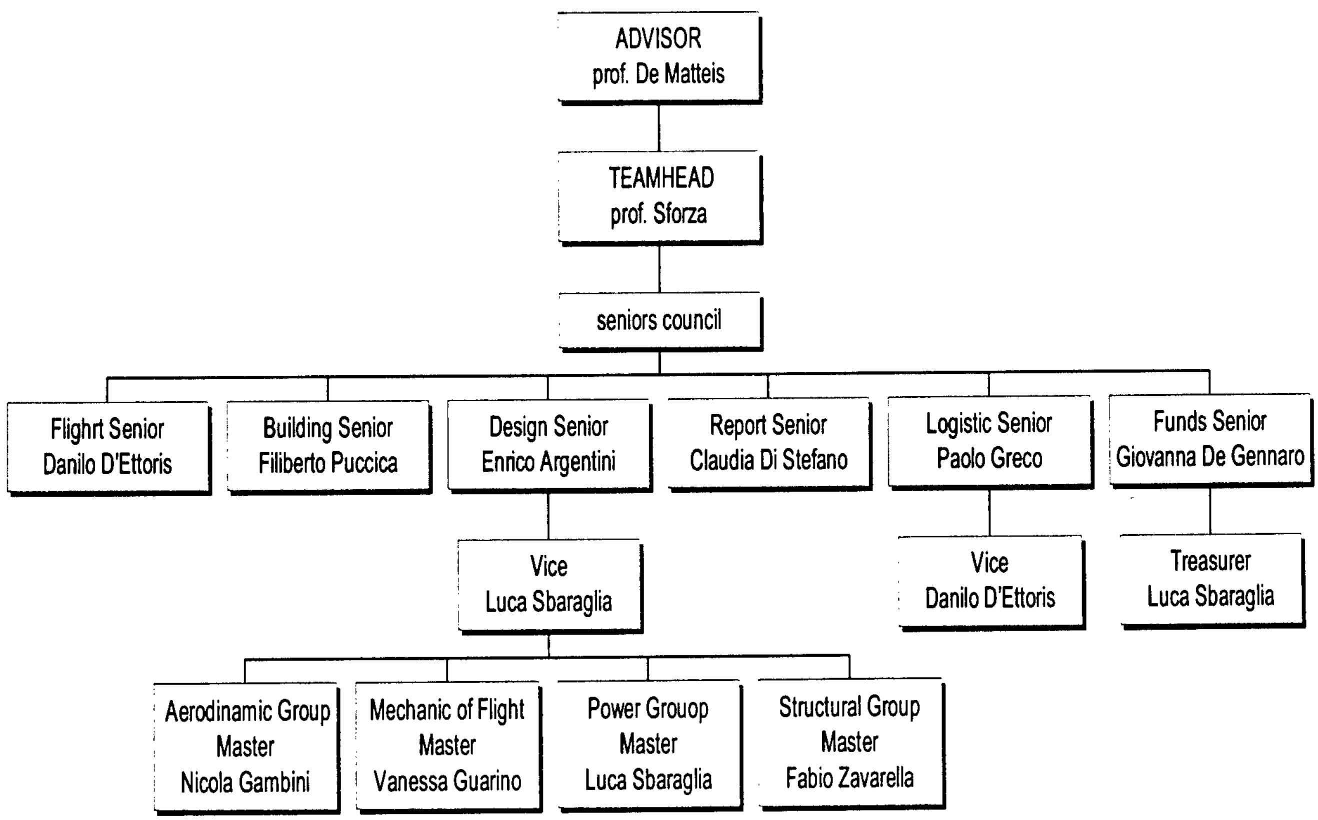


Figure 2.1 Organization of Team

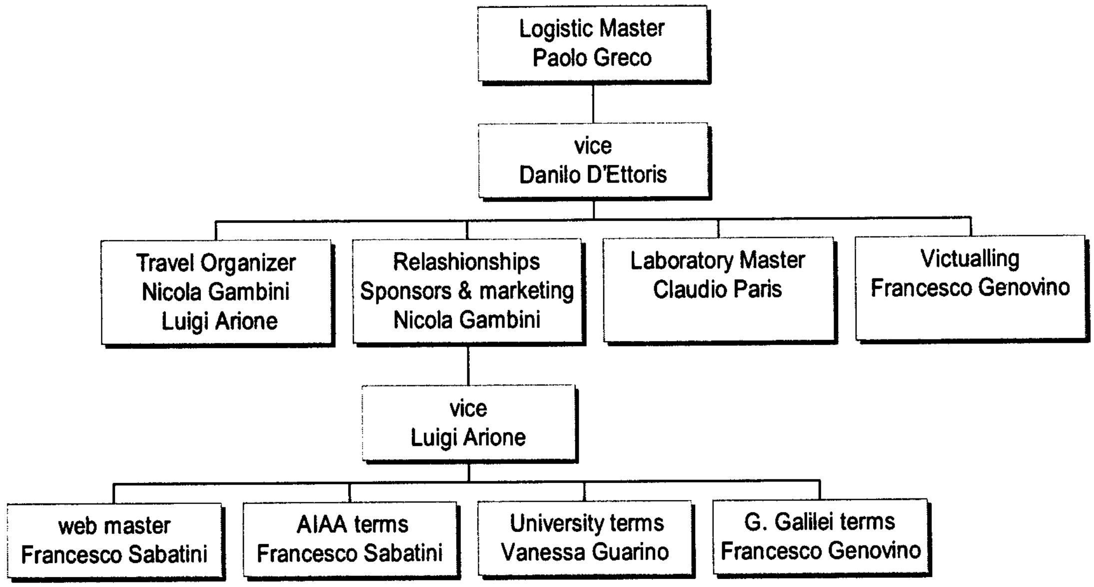


Figure 2.2 Logistic

	RULES KNOWLEDGE	PROPOSAL IDEAS	FOM	AERODINAMICS	PROPULSIN	STRUCTURES	FLIGHT MECHANICS	MATERIALS	CONCEPTUAL	PRELIMINARY	DETAIL	REPORT	NUMERICAL SIMULATION	FEM	CAD	TOT
Alberto Polimeni	4	3	4	0	0	5	0	3	1	3	3	0	0	0	0	26
Angelo Nanni	2	1	2	5	0	0	0	0	2	1	0	0	1	0	0	14
Claudia Di Stefano	5	2	4	0	0	5	0	3	5	5	5	5	1	4	0	44
Claudio Paris	2	1	1	0	0	5	0	3	1	1	1	0	0	0	0	15
Danilo D'Ettoris	1	0	0	0	5	0	0	0	0	1	2	0	0	0	0	9
Enrico Argentini	5	5	3	5	0	3	3	3	4	4	5	3	3	0	5	51
Fabio Zavarella	4	4	3	0	0	5	0	5	4	5	5	5	5	5	5	55
Filiberto Puccica	3	4	4	0	0	4	0	3	5	3	4	5	0	0	5	40
Filippo Gambini	2	0	0	3	5	0	1	0	2	1	2	0	0	0	0	16
Francesco Genovino	1	0	0	0	0	0	0	2	0	0	0	0	0	0	0	3
Francesco Sabatini	3	2	1	5	0	0	0	0	2	1	2	0	3	0	0	19
Giovanna De Gennaro	2	1	0	0	0	5	0	4	2	1	4	2	4	5	0	30
Giulio Feliziani	1	0	0	0	5	0	0	0	0	0	2	0	0	0	0	8
Lawrence Therese	3	3	3	0	0	5	0	3	2	3	4	2	1	3	0	32
Luca D'Arienzo	2	1	0	0	0	0	5	0	1	1	0	0	0	0	0	10
Luca Sbaraglia	5	5	0	0	5	3	1	3	3	2	4	5	4	0	1	41
Luca Trefiletti	2	1	0	0	0	0	5	0	2	1	2	3	0	0	0	16
Luigi Arione	4	1	0	0	0	5	0	3	2	1	3	0	2	3	0	24
Nicola Gambini	5	5	5	5	0	0	4	0	3	5	5	5	5	0	0	47
Paolo Greco	5	5	2	0	0	5	2	4	4	4	5	5	5	5	0	51
Pier Domenico Tromboni	1	1	2	0	0	2	0	0	0	0	0	0	1	5	1	13
Vanessa Guarino	2	1	3	1	0	0	5	0	3	4	2	4	3	0	0	28

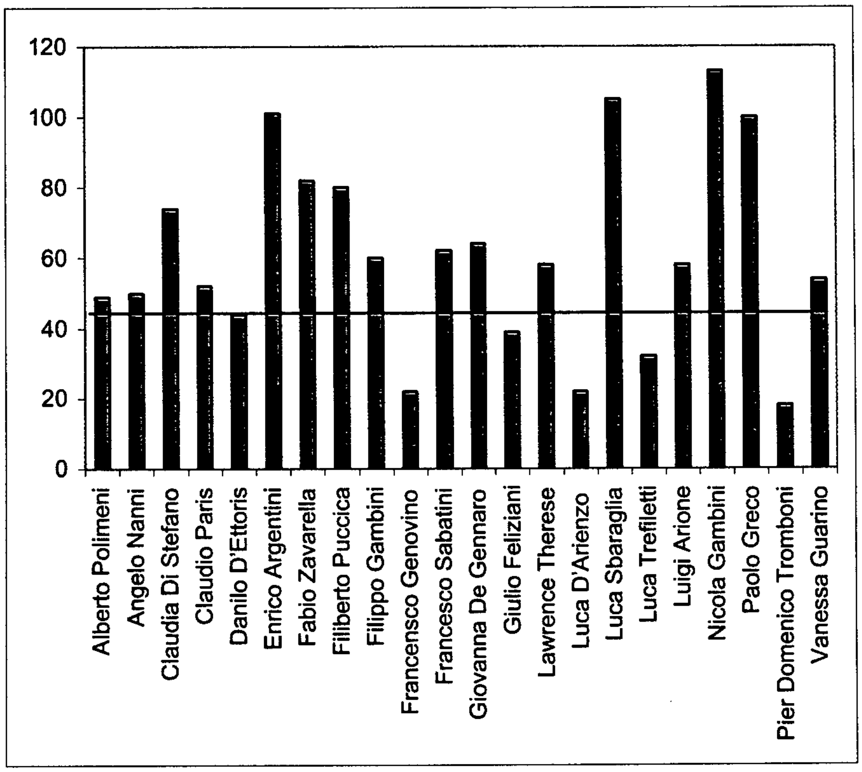
2.1 Program of Control

Thanking good results of last year, this year the team was very numerous. For this reason it was not possible for each member of the group to go to USA because of the limitation of funds. The only solution was to make a sort of chart between the people who participated to the design and building phases of the aircraft. From the results of this chart we obtained the people who could go to USA.

To construct the chart we analysed for each member of the team the contribution he gave to the project.

	LOGISTIC	PUBLIC RELATION	UNIVERSITY TERMS	SEGRETARY	G. GALILEI	SISTEMA COMPOSITI	FUSOLAGE	WINGS	EMPENNAGES	LANDING GEAR	MOTOR	COMPONENTS	TESTING	WEB MASTER	ENGLISH KNOWLEDGE	TRAVEL ORGANIZER	VICTUALLING	WIND TUNNELL	TOT
Alberto Polimeni	1	0	0	0	5	3	3	3	4	0	0	0	0	0	3	0	1	0	23
Angelo Nanni	1	0	1	0	5	5	3	2	3	3	1	2	0	0	2	0	3	5	36
Claudia Di Stefano	4	3	0	1	4	2	1	1	3	1	0	0	1	1	5	0	3	0	30
Claudio Paris	5	0	0	0	5	4	3	3	4	2	2	2	0	0	2	0	5	0	37
Danilo D'Ettoris	4	3	3	2	1	5	3	2	3	2	0	1	0	0	2	0	4	0	35
Enrico Argentini	5	3	3	1	5	5	4	4	3	3	0	4	2	1	3	0	4	0	50
Fabio Zavarella	1	0	0	0	5	3	2	2	2	3	0	1	2	3	3	0	0	0	27
Filiberto Puccica	3	5	1	0	5	5	2	2	2	1	0	2	5	0	5	1	0	1	40
Filippo Gambini	5	2	0	0	5	0	1	4	2	0	5	4	1	0	5	0	5	5	44
Francesco Genovino	5	1	3	1	0	0	0	0	0	0	0	1	0	0	3	0	5	0	19
Francesco Sabatini	5	5	1	0	2	4	3	1	1	0	0	1	0	5	5	1	4	5	43
Giovanna De Gennaro	3	5	5	5	2	1	1	0	0	2	0	0	1	0	2	2	5	0	34
Giulio Feliziani	5	2	0	0	4	1	1	0	3	0	4	3	0	0	3	0	5	0	31
Lawrence Therese	0	0	0	0	5	3	2	4	4	2	0	1	3	0	2	0	0	0	26
Luca D'Arienzo	0	0	0	0	4	1	1	1	2	0	0	0	0	0	3	0	0	0	12
Luca Sbaraglia	5	5	5	5	5	5	3	4	2	3	5	4	5	0	2	1	5	0	64
Luca Trefiletti	0	0	0	0	5	1	1	1	3	1	0	0	0	0	4	0	0	0	16
Luigi Arione	4	1	1	0	5	5	3	2	1	5	1	0	1	0	2	0	3	0	34
Nicola Gambini	5	5	2	3	5	5	1	5	5	5	0	4	1	2	3	5	5	5	66
Paolo Greco	5	5	1	0	4	5	4	4	2	5	0	1	3	0	4	2	3	1	49
Pier Domenico Tromboni	1	1	2	0	0	0	0	0	0	0	0	0	0	0	1	0	0	0	5
Vanessa Guarino	1	5	2	2	4	4	2	1	0	0	0	0	1	0	4	0	0	0	26

Alberto Polimeni	49
Angelo Nanni	50
Claudia Di Stefano	74
Claudio Paris	52
Danilo D'Ettoris	44
Enrico Argentini	101
Fabio Zavarella	82
Filiberto Puccica	80
Filippo Gambini	60
Francensco Genovino	22
Francesco Sabatini	62
Giovanna De Gennaro	64
Giulio Feliziani	39
Lawrence Therese	58
Luca D'Arienzo	22
Luca Sbaraglia	105
Luca Trefiletti	32
Luigi Arione	58
Nicola Gambini	113
Paolo Greco	100
Pier Domenico Tromboni	18
Vanessa Guarino	54



3.CONCEPTUAL DESIGN

3.Introduction

The goal of the *conceptual design* phase is to select the airplane concept which best suit the mission required.

Some items are required to be settled up during the early stages of the design: 1) to investigate a number of different airplane concepts; 2) to describe their physical behavior by some analytical models; 3) to identify the major parameters which affect the plane's performances; 4) to screen those features which cannot be translated into formulas by the use of some *figures of merit (FOMs)*. These items have been managed by all the technical groups. After identifying the parameters, each was studied further by modeling complete airplane designs with different combinations of these parameters. Each design received a rating based on weighted figures of merit. The results of the ratings determined the designs that were either eliminated or further analysed in the preliminary design phase. In particular, all the team members were involved in the proposal of different airplane concepts.

The development of analytical models' started during the conceptual stage, but their results were useful during the following stage, *preliminary design*, since the sizing process was focused on only one among the competing concepts. Some basic analytical studies have been done at this stage, involving a take-off run analysis, a first selection of wing plan and suitable airfoils.

On the other hand, in the process of configuration selection we trust also on the know-how acquired during previous years observations and experiences, particularly about the selection of power plant configuration.

3.1. Design Parameters and Aircraft Configurations

During the conceptual design phase has been identified design parameters and aircraft concept to be investigated. These have been rated using figures of merit to determine the design features that would be analysed in greater detail during the following phases. At this point the aircraft configurations that were judged too difficult to build or with a too high a rated aircraft cost, has been eliminated.

3.1.1. Wing Planform

Though it has not a direct effect on rated aircraft cost, the choice of wing planform can have an important effect on the performance of an airplane. Planform shapes evaluated included: tapered (λ=1/3), rectangular, and elliptical.

Previous experience by team has shown that low Reynolds numbers encountered at the wing tips of small aircraft having tapered wings result in stall risk in slow flying phases (landing!). Anyway, the tapered planform shape has been considered further in conceptual design, taking into account this its disadvantage.

The rectangular planform was evaluated because of its easy construction, and because of, letting the same wing area, provide a lower RAC.

According to aerodynamic theory, the elliptical is the best planform: the lift slope and the efficiency for the elliptic wing are greater than for other wings, for any aspect ratio. Manufacturing an elliptical wing was far more difficult than rectangular or tapered, and this has been taken into account in the next phases. The elliptical wing also tends to have very low Reynolds numbers at its tip, which may cause the tip to stall.

3.1.2. Wing Configuration

Wing configuration greatly changes the performance of an airplane: depending on the wing configuration can be obtained a greater lift slopes, increased efficiencies, and reduced drag.

Anyway, since from the conceptual design phase, has not been considered configuration such as biplane, canard, or wings with winglets. This because the experience from the other years showed us that the simple mono-wing is the best compromise between performances, manufacturing and rated cost.

Winglets were not considered because of the small performance advantage compared to the large rated aircraft cost penalty.

The greatest advantage of using a bi-wing is the increase in lift that is created by two wings. Because of this extra lift, a heavier payload could be carried, resulting in a higher total score. However, due to the interaction between the two wings, quite difficult to value, both with analytical and numerical tool, the lift per unit wing area for a bi-wing is not as high as it is for a mono-wing having the same aspect ratio.

In addition, rated aircraft cost increases due to the penalty placed on a bi-wing configuration. More, the maximum payload is assigned by the rules (24 softballs ~ 10 lb), and the total lift requested to carry it (no more than 30 lbf) can be produced by a mono-wing not too large (less than 10 ft2).

In the end: an efficient mono-wing with a large wingspan avoids higher rated aircraft costs and is more easily manufactured.

3.1.3. Fuselage Configuration

The fuselage is the "core" of the airplane. It must transmit the aerodynamic actions produced by the wings and tail, and the thrust of the motor, to the payload. It must also withstand the forces produced in the landing. The fuselage must also be aerodynamically efficient, light, and easy to build.

The fuselage must be large enough to carry 24 softballs, and must allow for and easy loading and unloading of payload.

Considering the rules about the disposition of the payload, the fuselage should be quite flat and wide. So the shape considered for the fuselage is a horizontal airfoil. The airfoil is the most aerodynamically efficient shape available. Not only it has very low drag, but also it could provide some lift.

For the fuselage section has been chosen the WASP, an airfoil with these characteristics: low moment, lower surface almost flat for an easy softball storing, and a low drag coefficient for angles of attack near zero

The disposition of the softballs has an effect on the fuselage's dimensions. Has been analysed three different dispositions: 2 rows of 12 softballs, 3 rows of 8, or 4 rows of 6. These three kinds of fuselage will have different width, length, and front and wetted area. This would have an effect both on the rated cost and on the aerodynamic behaviour of the aircraft.

The length of the fuselage is an important factor in the rated aircraft cost, a shorter fuselages gives to lower rated costs.

A fuselage with a small front area (i.e. a narrow and long one) will have a lower shape drag but a higher friction drag (due to the greater wetted area). An aircraft with a wide fuselage will have (letting unchanged the exposed wing area) a wing with a higher aspect ratio, reducing the induced drag.

In this contest the final score is a function of the rated cost and the mission time. Considering that one of the most important terms in the expression of the rated aircraft cost is the cell weight, the design will aim to a aircraft with a low consumption, with a low RAC.

3.1.4. Empennage Configuration

The tail surfaces are used to stabilize and control the airplane in pitch and yaw. In the conceptual design phase, a tail shape has been selected and evaluated, considering performance and stability. Have been considered two tail structure: V-tail and conventional tail.

The conventional tail consists of horizontal and vertical surfaces located aft of the centre of gravity, with the vertical one located above the horizontal one. The conventional tail is easily manufactured and well studied.

The V-tail consists of two surfaces arranged in a V-shape to provide stability in pitch and yaw. A V-tail requires a more complex work to analyse, and operate than a conventional tail. In fact in a V-tail both the surfaces give their contribute to the stability and control of both pitch and yaw. On the other hand, considering the rules about the horizontal surfaces, which state that an horizontal surface will be considered a wing, and then will increase greatly the rated cost, if its span is more than 25% of the larger one, a V-tail can are better than the conventional. In a V-tail there are two surfaces which can be each 25% of the main wing. This way, letting the same stability and controllability, is possible to have an airplane far shorter.

Any way, both the conventional and the V-tail, will be analysed.

3.1.7. Landing Gear Configuration

The main characteristics we wanted from the landing gear were enough strength to resist impact during landings and enough flexibility to absorb energy without transferring all the load to the fuselage's structure. More over, as any structural element, it had to be lightweight and to provide good handling characteristics while in the ground phases of the target mission. We knew those were the targets we had to reach during the sizing process of this structural subsystem. But there were a couple of points we should consider earlier.

In fact, two main question regarding its configuration were born since this conceptual stage of the design process. The first, about the choice between a nose-wheel configuration or a tail gear one. The second, regarding the selection of a fixed or retractable carriage. Due to its relative unimportance to the outcome of the conceptual design phase (when the major problem the team had to solve was to identify which sort of plane shape would have optimised the competition requirements), the decision to use a fixed or retractable solution was deferred to the next design phase, selecting both the configurations for detailed analyses. But the fist choice was strictly linked to the selection of the concept and the analysis of its performances, because both of possible solutions have a big influence on the vehicle behaviour in two critical phases of the flight mission, as take-off and landing. So, we had to investigate that problem since the beginning. From previous experiences in studying and setting RC UAV, we had enough know-how to make our first decision.

<u>FOMs considered and solution</u>

The tail gear solution, i.e. two main wheels placed before the center of gravity which support almost all the weight, and a small wheel placed under the tail, increases take-off and landings runs and affect them with directional instability (the "p-effect"), although it makes easier that maneuvers from a piloting point of view.

The front tricycle is composed of two main wheels placed behind the centre of gravity, and a nose wheel carrying a little percentage of total weight. It's advantage is to reduce take-off and landing runs, but it needs more cleverness in piloting. However, we knew this was not a problem for our "official" pilots, who strictly interact with us during the conceptual design stage.

So, in this choice we had to consider two main FOMs: the handling characteristics of the UAV (the easiness in piloting it during the initial and final phases of the mission), FOM #1, and the possibility to reduce take-off and landings runs, FOM #2. Quickly: the best FOM #1 would have meant the worse FOM #2., and vice versa. But the pondered value of the second FOM was felt to be higher, because of the mentioned reasons. So, we opted for a front tricycle carriage for further studies in the next phases of the design.

3.1.6. Power plant

The various components which compose the power plant – propeller, motor, controller, batteries – have been selected so that their combination would allow our UAV to best fit the following requirements:

- to take-off before 200 ft;
- complete the mission before 10 minutes;

and with this target

- to carry on board the minimum weight of Ni-Cd cells sufficient to allow us to complete the mission in the minimum time.

In fact this year the propulsive team, on the base of the contest rules and requirements, felt since the beginning that the all five pounds of Ni-Cd cells would have penalized too much the overall score (by increasing the RAC) even if they would probably have provide the best performances. This opinion has been verified in later analysis and tests.

A "philosophy" already strengthened since a long time in our propulsive team, "a big motor means a big efficiency", leaded us to the choice of one only motor. Then, the studies have been specified on two components: propellers and batteries. We did not produce analysis on the motor: the bigger availability in Europe of Graupner items compared to the Astroflight ones leaded us straight to the same choice of last year: a Graupner 3450/7 with a reducer (2.15:1) we made with the precious help of Mr. A. Silvagni, a big friend of our team.

To choice the best coupling motor-propeller, we thought to perform wind tunnel tests on MENZ and APC items, while we planned at this stage to make measuring sessions, varying voltage and current, so to have certain data on the electric features of commercial cells.

These data have been used in farther simulations with the software MATLAB to evaluate the minimum number of cells to perform the mission target.

3.1.7. Aircraft Configuration

Combining above design parameters we obtained 36 different configurations. They are shown In TABLE 3.1.The aim of this phase, was find those configurations who optimized figures of merit, so we gave a final judgment to every configuration and we chose the four with the highest score.

3.2 Figures of Merit

A figure of merit is a way to quantify and compare the benefits and drawbacks of a given design concept. The figures of merit were selected for conceptual design in an attempt to produce the highest possible flight score at the competition. Only two parameters directly affect the flight score. These are the weight of the aircraft and the rated aircraft cost. We don't consider the number of softball carried because we noticed, during the previews experiences, that only aircrafts witch carried maximum payload were challenging. Handling qualities were judged to be very important and to have a significant effect on the flight score, even because this year is very important to not lose time during the mission. However, handling qualities were not considered during the conceptual design phase because the analytical tools used were not sufficient to acquire a true representation of the handling qualities. This factor was instead deferred to the detail design phase. Therefore, the following three figures of merit were chosen based on the scoring criteria for the competition: weight and rated aircraft cost.

3.2.2. Weight

The first contributor to the total score is the weight of the aircraft. It depends by the dimension of the aircraft and by materials used to built it. To have a low weight is very important because if the aircraft is very weight we need of a big lift, end this will carry a big drag, and to equilibrate this we need of a big number of battery, etc.

To have an estimate of the weight we considered wings and fuselage surface (second is obtained by balls configuration and constant control power of elevator) and other fixed weight (engine, servos, etc.).

3.2.3. Rated Aircraft Cost

Rated aircraft cost was included in the figures of merit both as a contest requirement and as a method of quantifying the cost of the concepts being studied. Because the total score is inversely proportional to the rated aircraft cost, the higher the cost, the lower the overall score.

3.3 Configuration Selection

Analyzing the results of figures of merit is evident that one configuration is better than others. Of course we chose this configuration. In particular we decided to build an aircraft whit the following characteristics:

Plan form: Rectangular not tapered

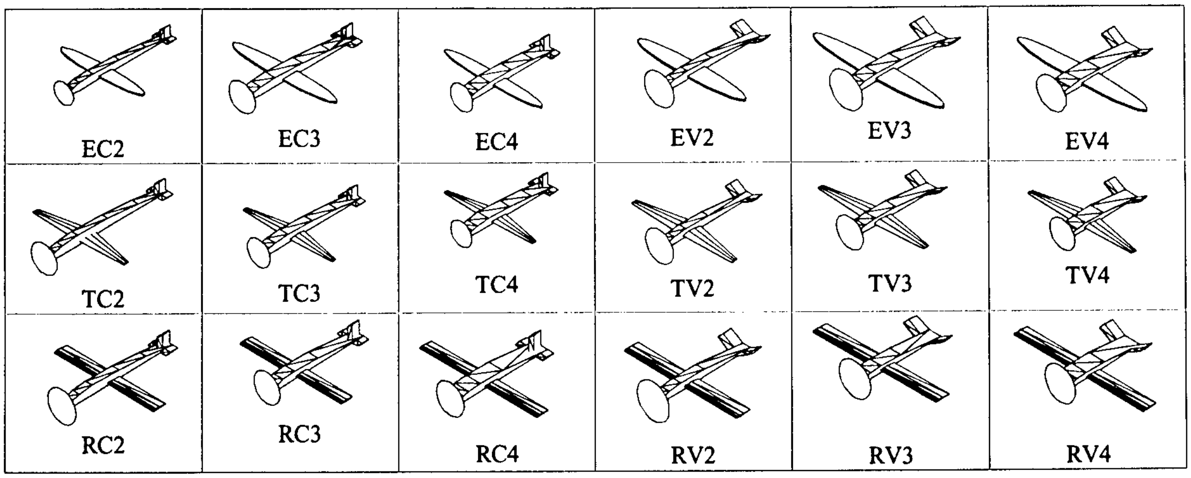
Profile : FX 63-137

Tail : V-tail

Rows of balls: 4

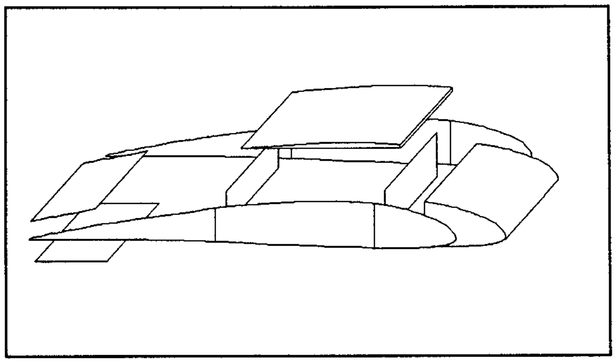
Planform E=Eiptical R=rectangular T=Tapered	Profile FX=FX63-137 SD=SD 70 32	Tail C=conventional V= V-tail	n° of rows of balls	Wingspan meters	Maximum chord meters	Wing Surface squared meters	Wing weight kilograms	Aircraft length meters	Total Weight kilograms	R.A.C.
E	FX	C	2	2,6	0,45	0,8485	1,79149	3,6102	6,09649	12,8127
E	FX	C	3	2,7	0,45	0,8485	1,79149	3,32278	6,04949	12,6662
E	FX	C	4	2,8	0,45	0,8485	1,79149	3,14161	6,03249	12,5961
E	FX	V	2	2,6	0,45	0,8485	1,79149	2,61266	6,01549	12,0403
E	FX	V	3	2,7	0,45	0,8485	1,79149	2,36219	5,97249	11,919
E	FX	V	4	2,8	0,45	0,8485	1,79149	2,21533	5,95949	11,8722
E	SD	C	2	2,8	0,49	1,00092	1,82994	4,06509	6,18294	13,2589
E	SD	C	3	2,9	0,49	1,00092	1,82994	3,76802	6,13394	13,1056
E	SD	C	4	3	0,49	1,00092	1,82994	3,57742	6,11594	13,0291
E	SD	V	2	2,8	0,49	1,00092	1,82994	2,8753	6,08394	12,3563
E	SD	V	3	2,9	0,49	1,00092	1,82994	2,61925	6,03994	12,2311
E	SD	V	4	3	0,49	1,00092	1,82994	2,46694	6,02594	12,1806
R	FX	C	2	2,6	0,35	0,84	1,673	3,06732	5,924	12,3594
R	FX	C	3	2,7	0,35	0,84	1,673	2,80001	5,878	12,2264
R	FX	C	4	2,8	0,35	0,84	1,673	2,63751	5,863	12,1689
R	FX	V	2	2,6	0,35	0,84	1,673	2,29923	5,866	11,7426
R	FX	V	3	2,7	0,35	0,84	1,673	2,06037	5,823	11,6289
R	**FX**	**V**	**4**	**2,8**	**0,35**	**0,84**	**1,673**	**1,92428**	**5,811**	**11,5895**
R	SD	C	2	2,84	0,38	1,0032	1,724	3,40729	6,011	12,7454
R	SD	C	3	2,94	0,38	1,0032	1,724	3,13392	5,965	12,6083
R	SD	C	4	3,04	0,38	1,0032	1,724	2,96537	5,949	12,5467
R	SD	V	2	2,84	0,38	1,0032	1,724	2,49551	5,94	12,0314
R	SD	V	3	2,94	0,38	1,0032	1,724	2,25315	5,898	11,9156
R	SD	V	4	3,04	0,38	1,0032	1,724	2,11357	5,885	11,8737
T	FX	C	2	2,7	0,5	0,83333	1,48794	3,73017	5,80594	12,9094
T	FX	C	3	2,8	0,5	0,83333	1,48794	3,44159	5,75894	12,7622
T	FX	C	4	2,9	0,5	0,83333	1,48794	3,25912	5,74094	12,691
T	FX	V	2	2,7	0,5	0,83333	1,48794	2,68193	5,72094	12,1028
T	FX	V	3	2,8	0,5	0,83333	1,48794	2,43079	5,67694	11,9808
T	FX	V	4	2,9	0,5	0,83333	1,48794	2,28317	5,66394	11,9336
T	SD	C	2	2,9	0,55	0,99	1,53876	4,26756	5,91276	13,4201
T	SD	C	3	3	0,55	0,99	1,53876	3,96698	5,86376	13,2645
T	SD	C	4	3,1	0,55	0,99	1,53876	3,77288	5,84576	13,1857
T	SD	V	2	2,9	0,55	0,99	1,53876	2,99219	5,80576	12,4596
T	SD	V	3	3	0,55	0,99	1,53876	2,73412	5,76176	12,3331
T	SD	V	4	3,1	0,55	0,99	1,53876	2,57979	5,74776	12,2812

TABLE 3.1 Figures of Merit of Conceptual Design Phase.



3.4 A new philosophy in manufacturing

As we said earlier, this year we could count on the collaboration of a fabric which produces composite materials manufactured items. So, quite soon in the conceptual design phase we had to estimate if it was really worth to adopt this sort of materials in our design, and how to exploit best this opportunity. A proper application of the technology of composite materials should not limit to change one by one the structural elements that used to be built with conventional materials with composite ones. It is necessary to come back to define a new design "strategy". So we individuated a new solution to emphasize the characteristics of fastness in building of such materials. We thought to a "box modular structure" with high ratio strength/building easiness to realize early some prototypes of the different structural cells to validate the design with a serious session of static and dynamic verifications, aerodynamic tests and flight tests. This manufacturing strategy has been selected also to make quite easy and not much expensive the application of a new method in building about which we did not have a deep know-how. In particular, we thought this choice was the best to make easy and fast the assembly of different components in one only structure, increasing the advantage we could have from the help of our main supplier, the Sistema Compositi. In fact we thought it could be possible to proceed "step by step", i.e. to manufacture the single elements using the S.C. facilities, and then to assembly them at our laboratories. And thus it was also possible to make some change in a particular elements without involving the others. This was the philosophy we adopted to fit the new technology to our requirements. But only later in the design process we would have evaluate which



structural elements had really obtained a significant advantage in being realized in composite. We have to say actually this prospect represented a big new in the Flying Centurions curriculum, and that is also because it brightened up all the team's trust and enthusiasm in enterprising this new challenge.

In the precedent figure is illustrated the concept of "box modular structure" solution with regard the fuselage's cell.

3.5.1 Wing Structure

Once the geometry had been fixed, we assumed several possible solutions for the wing structure:

a. carbon-epoxy monocoque with fuselage bayonet coupling and composite ribs;
b. sandwich with carbon-epoxy skin and polystyrene foam core, with a filament wound carbon-epoxy tube along the whole wing span and root ant tip's ribs;
c. sandwich with glass-epoxy skin and polystyrene foam core, with a composite spar;
d. traditional ribs and spars wooden structure.

We submitted these four structural options to a first screening on the base of the following FOMs, considering integer values from 1 to 5 :

- estimated weight (W);

  we estimated the total weight for each of the cited wing configurations by using an work sheet considering the different densities and quantities for the materials involved. This was the main FOM, so we associated to it a pondered factor of 0.4, the bigger within all the FOM introduced. Higher the value heavier the configuration.
- Manufacturability (B);

  this parameter supplies an indication about the difficulties we could find during the manufacturing process; we had sufficient know-how to say a wooden structure is quite difficult to build, and each single element is not in reproducible in series, while the technology of composite materials, by using moulds, allow to build sets of each piece. Anyway, since we planned to begin the manufacturing process quite early with respect previous years, we gave this parameter low importance, so its pondered value is 0.1; higher the value harder the manufacturing process.
- Fuselage junction (J);

we felt this was a slightly more important target than the above one, since a simple philosophy of junction could allow us to make little changes in wing setting. The pondered value is 0.12. Higher the value less simple the junction.

- estimated manufacturing time (T);

  this is the second more important parameter we considered, as we planned to build all the considered configuration and to choose the best one. Higher the value longer the time required to manufacture the wing. Its pondered value is 0.3.

- costs ($).

  This is the less important parameter we considered. Indeed the "Sistema Compositi" supplied us free materials. Moreover cost of further material we bought (especially wood) is very cheap. Higher the value more expensive the configuration. Its pondered value is 0.08.

Table 1 - Wing FOMs

configuration	**W**	**B**	**J**	**T**	**$**	**sum**	**pondered sum**
A	2	4	1	3	5	16	5,4
B	2	2	1	3	4	14	3,73
C	1	2	3	2	3	14	3,26
D	3	5	3	5	1	21	7,22

On the base of this first analysis, we rejected the solution number 4; we carried on all the other three for more detailed studies.

3.5.2 Fuselage structure

Once the aerodynamic team assigned us the fuselage configuration, a wing section, we started analyzing different design solution:

a. Traditional ribs and spars wooden structures
b. carbon-epoxy composite monocoque
c. box-shaped structure.

By considering opportune FOMs we selected the winning solution:

- Estimated weight (W);

  it provides an estimation of the final weight of the fuselage assembled according to available configurations. Higher the value, heavier the fuselage. It's the most important parameter we consider, assigning to it a pondered value of 0,4

- Manufacturability (B);

this parameter supplies an indication about the difficulties we could find during the manufacturing process; we had sufficient know-how to say a wooden structure is quite difficult to build, and each single element is not reproducible in series, while the technology of composite materials, by using moulds, allow to build sets of each piece. Anyway, since we planned to begin the manufacturing process quite early with respect previous years, we gave this parameter low importance, so its pondered value is 0.2; higher the value harder the manufacturing process.

- Estimated manufacturing time (T);

  this is the second more important parameter we considered, as we planned to build all the considered configuration and to choose the best one. Higher the value longer the time required to manufacture the wing. Its pondered value is 0.3.

- Costs ($).

  This is the less important parameter we considered. Indeed the Sistema Compositi supplied us free materials. Moreover cost of further material we bought (especially wood) is very cheap. Higher the value more expensive the configuration. Its pondered value is 0.08.

Table2 - Fuselage FOMs						
configuration	**W**	**B**	**T**	**$**	**sum**	**pondered sum**
a	5	3	4	1	14	3.9
b	2	3	3	4	14	2.7
c	1	2	2	5	13	1.9

We discarded the first two configuration and went on to analyze the third that it is going to be built.

4.PRELIMINARY DESIGN

4.Introduction
In this phase of the design we needed to make sure that the outcomes we had reached were right. For this reason we started making some test on our prototype.
After that we considered some figures of merit specific for each part of the aircraft keeping save previous results.

4.1 Prototype Testing
nicola

4.2. Power Consumption
Let us assume power Consumption basically dependent of wet area rather than on aircraft weight. To confirm our assumption we made the Figure 4.1 where we deduced that structural weight is strictly connected to the payload one.

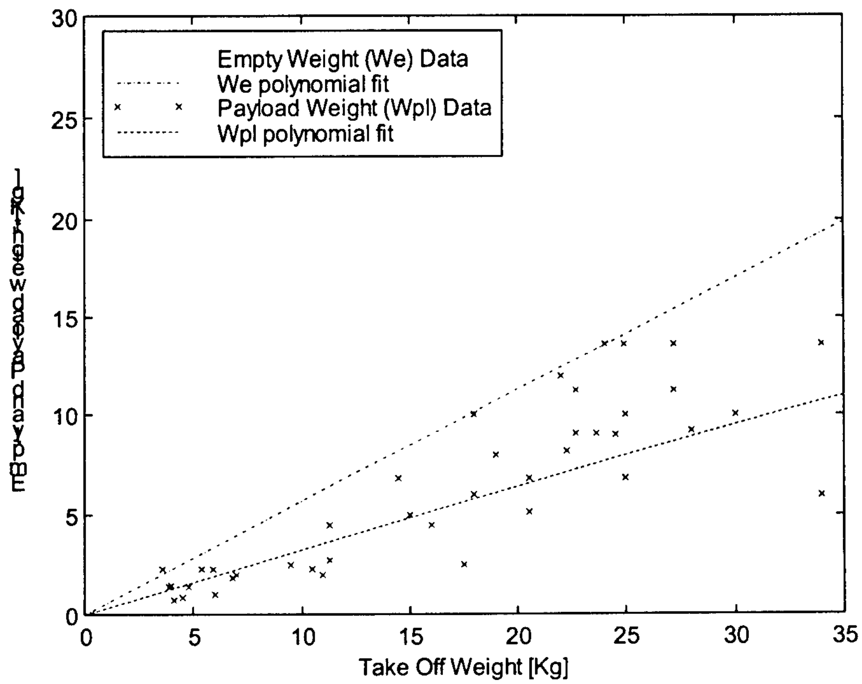


Figure 4.1 Power Consumption.

If we wanted to transport five Kilograms of payload the weight of structures couldn't be less than six or seven Kilograms; In this way total take-off weight was eleven or twelve Kilograms. Besides we can't neglect that important parts of the aircraft(as engine and batteries) had fixed weight. On the other hand we could find solutions to optimize wet area.

4.3 Design Parameters Investigated
In the Preliminary Design Phase we refined the parameters from Conceptual Design. In particular we studied structure and materials of the following part of the aircraft:

Wing

Fuselage

Empennage

Landing Gear

In this part was fundamental the contribution of structural group.

4.3.1 Wing Structure and Materials

In the second phase of designing process, the structural team planned to size and to produce a single prototype for each challenging solutions survived to the conceptual step.

Solution 1: carbon-epoxy monocoque with fuselage bayonet coupling and composite ribs

It consist of a manual lay up of carbon fiber epoxy prepreg (1 layer of 500 g/m^2 fabric, 3 layers of 200 g/m^2 UD ribbons placed on the maximum thickness line as spar taps) vacuum cured for the wing skin, four sandwich ribs with honeycomb (nomex) core and carbon fibers reinforced plastic (cfrp) skin equidistant along the half wing span, and a filament wound cfrp tube for the junction with the fuselage.

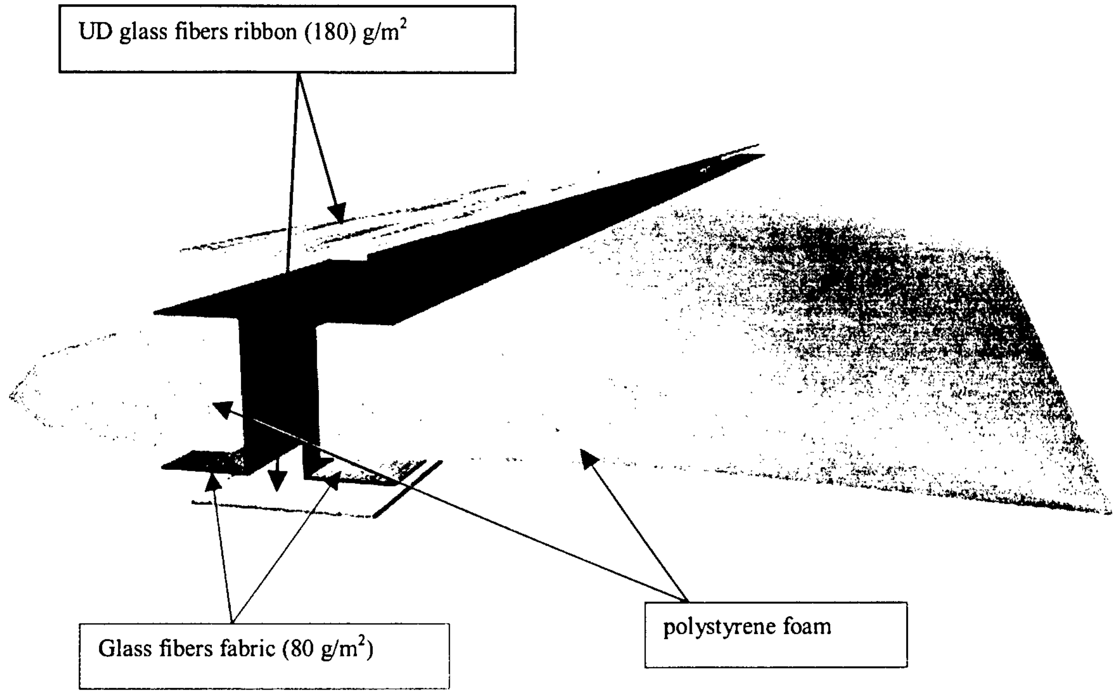
Although it was quite easy and fast manufacturing the various elements of this structure, it was very difficult their assembly because of unexpected problems during the curing cycles: in fact the skin deformed because of residual local stress fields due to the different polymerization of materials employed. It suggested us to neglect this solution and investigate the other two.

Solution 2: sandwich with carbon-epoxy skin and polystyrene foam core, with a filament wound carbon-epoxy tube along the whole wing span and root ant tip's ribs;

This option wanted a single cfrp filament wound tube along the whole wing span. But to realize it, we needed an opportune steel spindle very long (1.5m actually), with the correct diameter. Since we could not find a spindle with the wished requirements, we had to reject also this solution.

Solution 3: sandwich with glass-epoxy skin and polystyrene foam core, with a composite spar

We realized this option was very simple to build. Actually, we manufactured a prototype half wing in about half a day! The building process is then described: after having produced an opportune core with the shape of the half wing by slicing a block of polystyrene foam with a hot wire machinery, we cut it in two parts along the line of maximum thickness. Then we made a manual lay up of a glass fiber fabric (80g/m^2)with an epoxy resin on the cut surfaces. By the junction of the two parts, we obtained a "double t" spar. Then we increased the spar taps' thickness to the opportune design value by laying up other strips of UD glass fiber ribbon(180g/m^2), as in the following figure. Finally we made the skin with a single layer of glass fabric (80g/m^2).

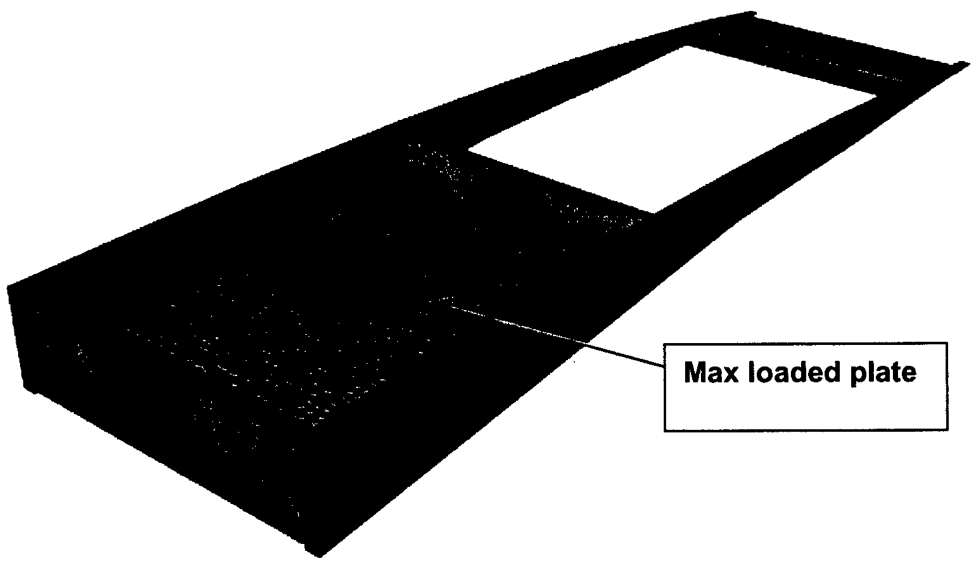


4.3.2 Fuselage Size and Materials

Before beginning sizing of the structural elements, we made a little change on the shape the aerodynamic group had designed. This airfoil has been modified, decreasing the thickness and flattening the lower surface, to increase the "volumetric efficiency" (volume of the payload over volume of the fuselage).

Fuselage's box shaped structure consists of a composite laminate made of two carbon epoxy skin (500 g/m^2 prepreg) and a fenolic honeycomb core. All the pieces used to assembly fuselage were obtained by a single layer to reduce costs and manufacturing time. They consists of two external ribs with a chord of 1,65 m, two rectangular plate 12 x 42 cm and a bottom plate 42 x 65 cm We arranged an analytical study to estimate the laminate thickness: we analyzed components of the fuselage loaded with major forces and the result was that the most stressed component is the bottom plate of the cargo bay (see following figure) as it will bring the payload mission. We estimate a uniform load of 90 N (it's equivalent to a load factor of about two).

Fuselage box shaped structure



4.3.3 Empennage Size and materials

For the structural design of the tail two manufacturing solutions have been investigated. A classical one using structural ribs made of wood, shape ribs made of polystyrene foam and a wooden spar. An advanced one consisting of a cfrp monocoque able to resists the loading condition and a short cfrp tube used to join the tail to the fuselage. Both solutions had a comparable estimated weight so we manufactured both of them to choose the one best fitted our specification. We discovered the surprising result that the classical wooden solution was twice lighter than the composite one due to an undervalued quantification of glue needed to join the crfp monocoque parts.

4.3.4. Landing Gear Configuration.

In the previous phase of the design process, we had already opted for a front tricycle configuration of the landing gear subsystem. Now, we had to choose how to design it. First of all, we investigated the possibility to adopt a fixed or retractable gear. Then, we studied the best way to manage its manufacture.

<u>FOM considered and solution</u>

A retractable gear second mentioned configuration has an important advantage: if it is properly constructed, during the cruise phase of the mission it does not affect the vehicle with an additional aerodynamic drag. But it has also several problems. In fact, it needs opportune structures and mechanisms which increase the structural weight without any doubt. More over, in the specific contest of this competition, it has another

disadvantage, since it needs its specific servo, meaning another addendum in the calculation of total weight and Rated Aircraft Cost. Other aspects we had to consider about were availability of opportune materials to build it, mounting complexity, and reliability.

But the choice within the two cited configurations was not the only aspect we planned to evaluate at this point. In fact, we began to wonder about the manufacturing process of the carriage. We identified two main possibility: to build a structure using composite materials (reinforced plastics with carbon, glass or kevlar fibers), or to choose a traditional aluminum system, which had been used successfully in previous contests, and gave us a big confidence from the reliability point of view. In fact, traditional solid spring landing gears are quite common to RC-Modelers. They have been time tested to be the most reliable landing gears available, and have good ratios strength/weight. But we did not want to reject a composite materials solution before having estimate its eventual advantage in reducing the total gross weight by a significant percentage. So, we evaluated the two couple of different options on the base of the following FOMs: near each FOMs relative pondered value (pv) is indicated:

- strength (S) 0.2
- flexibility (F)
- light weight (W)
- handling characteristics (H)
- cruise behavior (drag addendum) (D)
- RAC addendum (R))
- materials and mechanisms availability (A)
- easiness in assembly (E)
- reliability (Y)

The Configurations we compared are:

a. Fixed aluminum landing gear
b. Retractable aluminum landing gear
c. Fixed Kevlar landing gear
d. Retractable Kevlar landing gear

Configuration	**S**	**F**	**V**	**H**	**D**	**R**	**A**	**E**	**Y**	**Pondered Sum**
a	3	3	3	2	2	3	1	3	3	2.6
b	2	2	1	2	2	1	2	1	2	1.9
c	3	2	3	2	2	3	2	2	3	2.65
d	2	1	2	2	3	1	2	1	1	1.55

The fixed configurations best fit our requirements, and the composite landing gear has a light advantage towards the aluminum one. We decided to consider both fixed configuration for further analyses.

4.4 Analytical Tools

Nel paragrafo 4.4 si dovrà fare una carrellata degli strumenti analitici usati negli studi descritti nei paragrafi precedenti. Per strumenti analitici si intende programmi fatti al computer e eventuali formule usate.

4.6 Engineering Requirements

The engineering requirements were established during Preliminary Design phase because during Conceptual phase we had not take yet decisions about some parameters necessary to calculate them.

Gross Weight : *12 Kg*
Baseball Ball Capacity : *24*
Payload Laps : *2*
Wing Planform : *Rectangular Not Tapered*
Wing Span : *2.84 m*
Landing Gear : *Not Retractable*
Batteries Specification : *Sanyo RC 2400*

5.DETAIL DESIGN

5.DETAIL DESIGN

1.Introduction

5.1 Mechanic of Flight

5.1.2. Control System

The aileron control power can be approximated using a strip method. The portion of the wing having the aileron is broken into strips and the lift increment due to aileron deflection is estimated as a flap effect using the classic flap theory. It takes into account the aileron span, deflection δa, and the reduction in zero-lift angle (equal to the increase in lift coefficient due to aileron deflection divided by the lift-curve slope). A stationary roll rate of 100deg/s has been imposed at V=20 m/s and δa=20 deg. Equating the roll moment due to the damping roll effect, and the one due to the aileron, we have:

$$\frac{1}{4}\rho USC_{l_p}b^2\Delta p = \frac{1}{2}\rho U^2 SC_{l_{\delta A}}b\delta_A$$

where C_{l_p} is the roll damping coefficient and $C_{l_{\delta A}}$ is the aileron control power coefficient. The latter depends on $\tau_{\delta A}$, aileron efficiency obtained by a purely empirical expression which only parameter is c_a / c (c_a=aileron chord!). Hence, aileron chord may be sized by solving equation with respect to $\tau_{\delta A}$. Final result is c_a=7 cm.

The tail has been built in a V-tail configuration because it is the best solution in terms of hours in the Rated Aircraft Cost. At the same time it allowed to test a new solution never studied before in our last years projects. To obtain satisfactory stability and control, the V surfaces must have an equivalent size to the total area of the conventional empennage layout. For Galileo this equates to 0.2976 m².i.e.0.1488 m² per ruddervetor. At this point we could chose between conventional, variable incidence or all moving tail plane. In conventional layouts the setting angle is fixed giving rise to inefficiencies for cruise capability with regard to c.g. and speed fluctuations. Variable incidence has similar problem to above. So the choice has been the all moving tail plane which has resulted an effective design. Aerodynamically it proved to be sound because the whole surface acts as a control surface and the size has been made smaller. So our tail plain have the following measures: root chord: cr=0.36 m; tip chord ct=0.12 m; span b= 1.47m;sweep:Λ=10 deg .As regard the lateral-directional stability it has given an effective yaw control especially because of the wing without sweep and dihedral angles does not give enough contribution to yaw and roll stability.

5.2. Performance Analysis

In this section we present the results obtained on the final configuration using the power plant and aerodynamics models given into the previous section. Are included the calculation for the Rate of Climb, for static and dynamic stability and the consequent results regarding the controllability and handling qualities.

5.2.1. Take-off and Climb

The following calculations are made in two flight conditions:

Full payload: W=127.53 N;

Empty: W=83.4 N;

We estimated take-off distance for the competition in 48.4 m. The available static thrust has been scaled to the 75% over the maximum available, which should allow battery saving at the first take-off at least. With the gross weight of 127.53 N the thrust required (at steady level flight) is T_R=15 N, as you can see in figure 5.2. The Rate of Climb is obtained using the common approximate expression valid for an aircraft with small angle of attack and small climb angles:

$$RC = \frac{V(T_A - T_R)}{W}$$

The value obtained are: RC =1.76 m/s (for gross weight) and RC=3.6 m/s (for empty weight).At this point we have checked that CL values corresponding to γ_{max} were smaller than CL $_{max}$,in order to have a good security margin on the stall-condition!

5.2.2. Range, Endurance, and Payload

Considering the previous years' experiences, we noticed that the winning teams were the teams which flew carrying always the maximum payload. Therefore we concentrated our attention on an aircraft which was able to carry 24 softballs and could be able to fly with full payload in every payload sortie. The range and endurance are calculated using a flight simulation program SIM 2.3. It integrates the equation of motion and contains an autopilot subroutine in order to track a given flight path and another subroutine which give a prediction of the power consumption around the flight course. SIM 2.3. takes into account the three different mission tasks (as provided in the DBF rules) assuming 30-seconds payload loaded (or unloaded) times between sorties and giving the total flight period. The result is 432 sec, a time smaller than 10 min, an optimistic prediction which will be verified during the flight test before the contest!

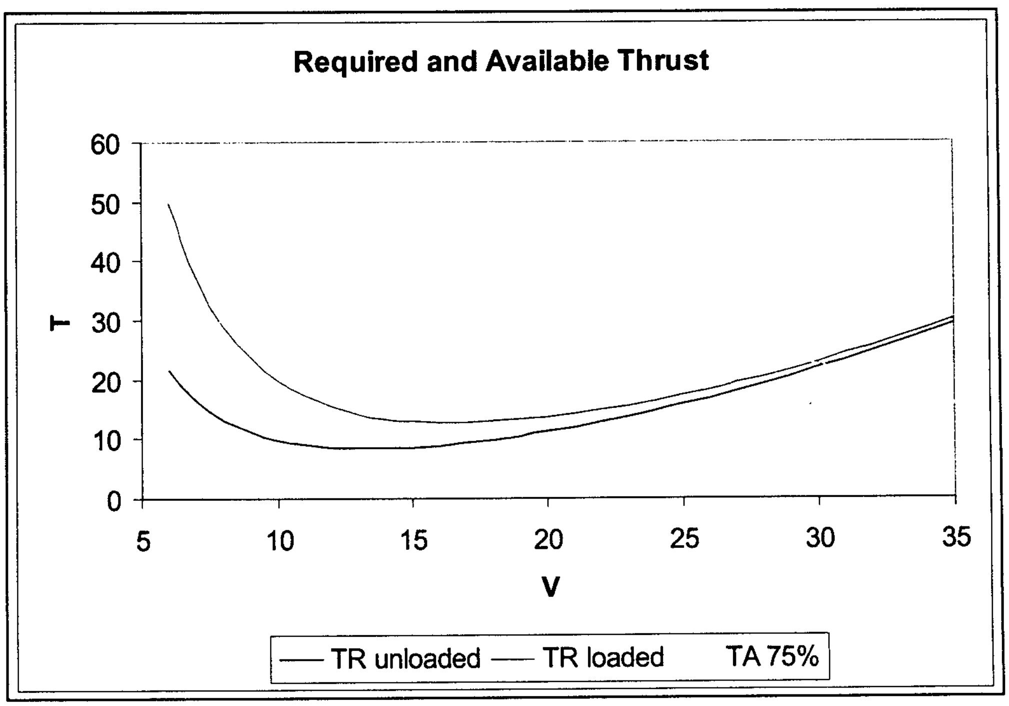


Figure 5.1. Thrust are measured in N, V in m/s.

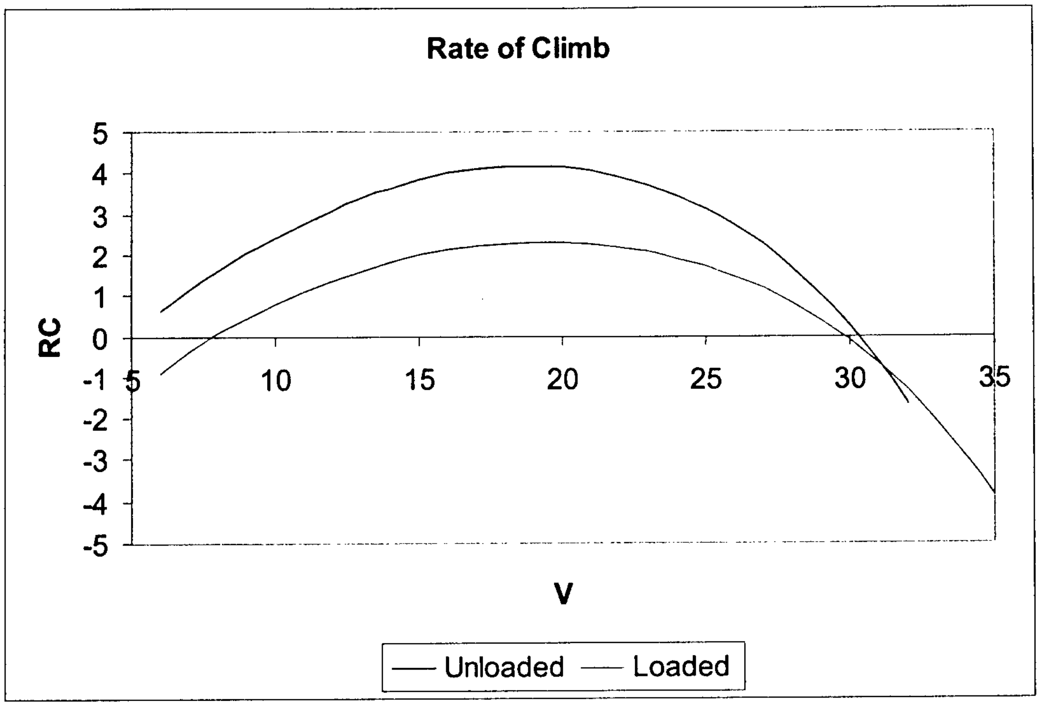


Figure 5.2.

5.2.3. Handling Qualities. A trim analysis has been made to determinate a static margin which could be adequate to every flight condition and at the same time could give enough handling qualities. So we choose the position of the tailplaine to give a static margin of 0.4.

Both the longitudinal and the lateral-directional motions have been studied separately, by a classic eingenvalues approach. In this case the equation of motion have been linearized around the cruise condition at level flight. The non- dimensional stability derivatives were calculated mainly by using the classic DATCOM methods.

Longitudinal dynamics. The following tables shows the non-dimensional and the dimensional longitudinal derivatives and the moment of inertia around Y body axes in full payload condition:

Table 5.3. Longitudinal non-dimensional derivatives.

$C_{L,\alpha}$	$C_{D,\alpha}$	$C_{M,\alpha}$	$C_{L,q}$	$C_{D,q}$	$C_{M,q}$	$C_{L,\dot{\alpha}}$	$C_{D,\dot{\alpha}}$	$C_{M,\dot{\alpha}}$
5,027429	0,3377749	-1,3688	0,154220793	negligible	-13,1939	7,55218721	negligible	-2,184261248

Table 5.4. Longitudinal dimensional derivatives and moment of inertia around Y body axes.

I_{yy} [kg m²]	X_α	M_α	$M_{\dot{\alpha}}$	M_q
1,831824	-3,47287	-53,77931602	-0,06878	-9,0716362

The longitudinal dynamics shows the following modes for a steady-level flight:

Mode	Eingenvalues	Period [sec]
Phugoid	-0.047079±0.6920723i	9.0578865
Short Period	-5.86245671±6.546492623	0.714993388

Lateral-directional dynamics. The following tables shows the non-dimensional and the dimensional lateral-directional derivatives and the moment of inertia around X and Z body axes, in full payload condition and for a steady- level flight:

Table 5.5.Lateral-directional non-dimensional derivatives.

$C_{y,\beta}$	$C_{l,\beta}$	$C_{n,\beta}$	$C_{y,p}$	$C_{l,p}$	$C_{n,p}$	$C_{y,r}$	$C_{l,r}$	$C_{n,r}$
-0,49265	-0,0533465	0,191938	0,045959759	-0,44777	0,0410893	0,13239734	0,140565643	-0,065900733

Table 5.6. Lateral-directional dimensional derivatives.

Y_β	L_β	N_β	L_p	N_p	L_r	N_r
-5,0652719	-30,7153	15,47166	-18,047	1,6560584	5,66534	-5,312109412

Table 5.7.Moment of inertia [kg m²]

I_{xx}	I_{zz}	I_{xz}
1,241566	3,074555	0,0362539

Y=lateral force; L= moment around x-axis; N=moment around z-axis; β=sideslip angle; p=rolling rate; r=yawing rate. Interesting is to observe that in the expression of C_y derivatives the bigger contribution is given by the tail because of its V-tail configuration in spite of a rectangular wing area. So for the tail we have obtained: $C_{y,\beta}$ =-0.48415, while $C_{y,p}$ and $C_{y,r}$ are practically equals to the values in the table.

The resulting lateral-directional modes for a steady-level flight are listed below:

Mode	Eingenvalues	Period [sec]
Roll	-18.0008	0
Spiral	-2.49266	0
Dutch Roll	-2.749285±2.7478920i	1.61642726

So we noticed all the modes were stabile. We were glad having found the right equilibrium within lateral and directional stability.

5.3 Propulsion

In order to determine the propeller's characteristic curves, were carried out some tests in the wind tunnel. Were taken in to account only the wooden made, double-blade, Merz propellers, because they had the highest score received from the components selection. The tests regarded 16 inch diameter and 10, 12,14, 16 inch pitch propellers. Comparing these tests' results with the last year results obtained using the

simulation program "ELICA", we recognized remarkable differences. Using a 2kW, voltage stabilized power supply (brand: EUTRON mod. BVR2000), we could supply our engine in a rigorous and controlled way. We know engine performances (efficiency, RPM, Pin and Pout) for different current values (from 0 to 50A) and fixed voltage values.See fig ****. We built a dynamometric balance specially or the tests. The balance is made of a vertical steel arm, sheltered from the tunnel stream by a fiberglass shell, on wich the engine is fixed. We gauged the strength provided by propeller, gauging deformations of the arm near the joint. Deformations ere gauged by an S/G bridge. Using a data acquiring system (Keithley 2700+7700 card), we acquired the following values:

RPM

Trust

Voltage in

Current in

Stream speed

From these data, we obtained the values of Ct and Cq coefficients, for different values of γ, as in the following figure.

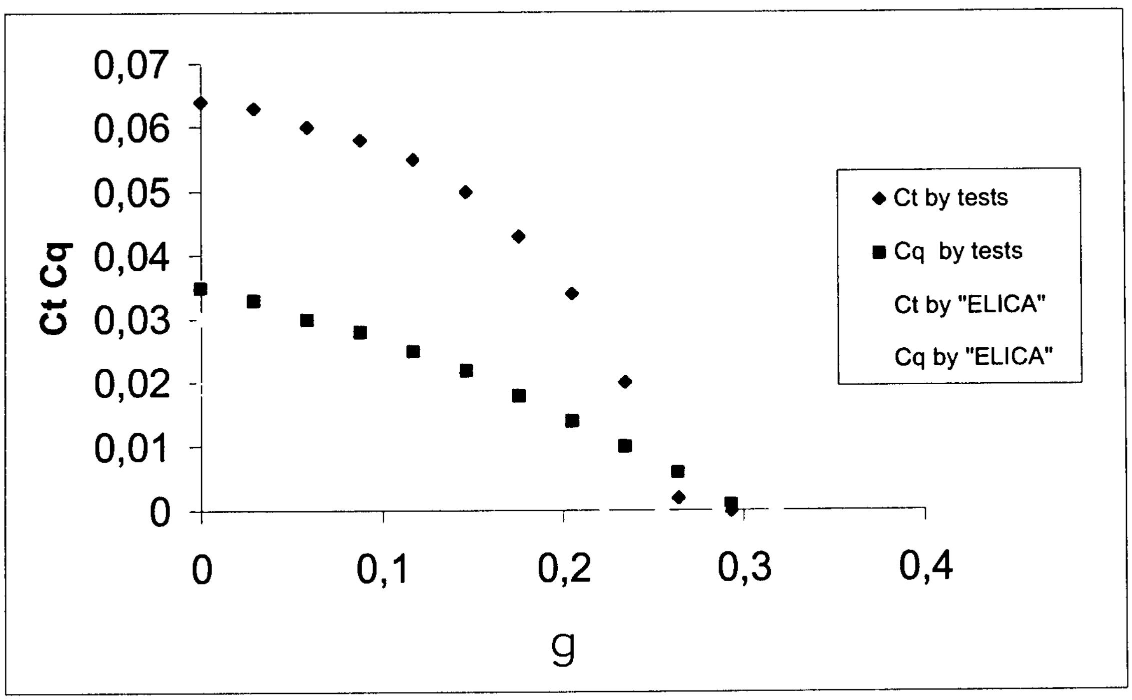


Figure 5.3

$$\gamma = \frac{V}{n * D}$$

V= stream speed
N = rpm
D = diameter of propeller

Having "reliable" propeller's curves, we simulated the performances of whole power plant using "Motor 2.3", the latest version of an our Matlab program.
The final performances are shown in fig ---.

Conclusions
number of battery: 35 SANYO RC2400
propeller: Menz 22x12
engine: Graupner 3450/7 geared 2.15

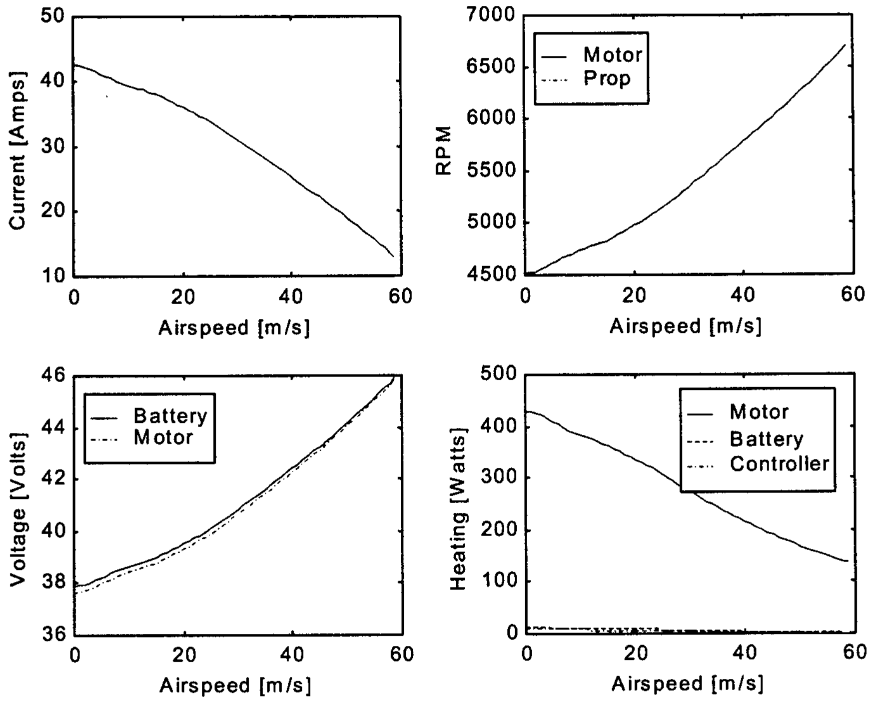


Fig 5.4

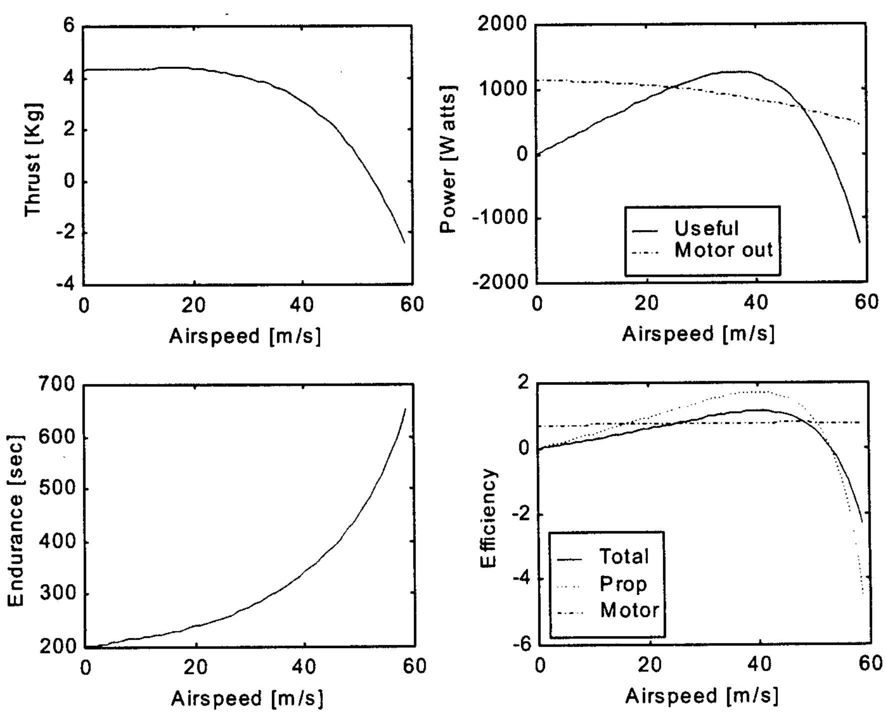


Fig. 5.5

5.4 Structural Analysis
5.4.1 Wing

To reduce the design time, we decided to size the spar caps to be able to resist all the flexural load acting on the whole half-wing. We assumed as maximum stress condition the structural verification the UAV will attend at the contest site, i.e. being lifted with one lift point at each wing tip to verify adequate wing strength. Our target was to size thickness and width of the taps in order to resist the maximum flexural moments, locate at the root section. The height of the spar being fixed by the maximum thickness of the airfoil chosen (47 mm). We assumed for the tip load value a force of 100 N (equivalent to the half full weight multiplied for a safety factor equals to 1,5). To optimize structural weight distribution, we produced a parametric study, by ranging the thickness t and the width a of the taps. In the following figure is plotted the surface corresponding to the maximum flexural stress at the root section as a function of both the mentioned parameters. In the same plot we identify a plane of maximum allowable stress equals to the flexural strength of the material used, 100 MPa.

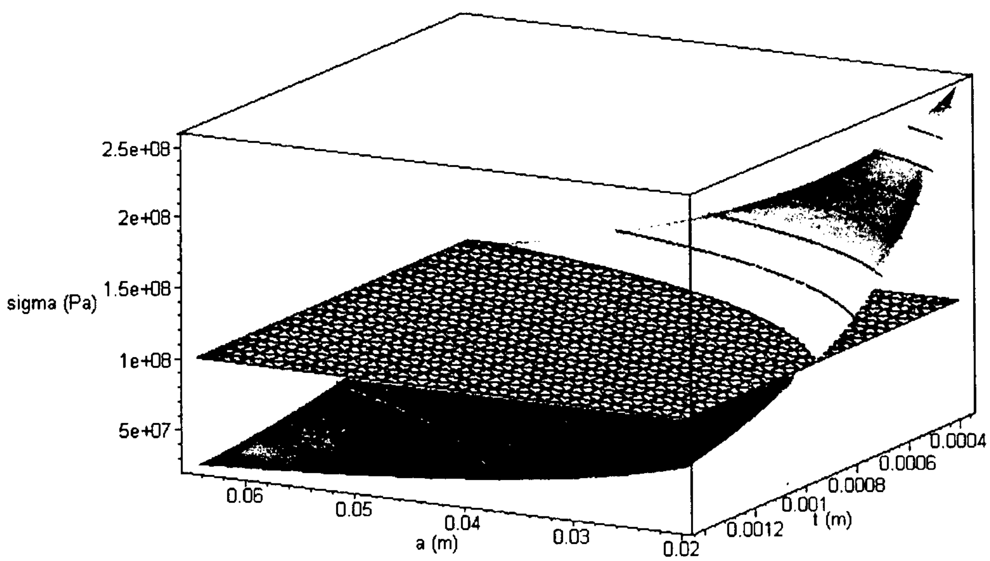


Figure 5.6

The intersection of the surface with the plane shows a line, "limit curve", figure 5.6: this is the site of combination (a,t) which verify the limit condition (above which we are in safety condition).

In this sort of study, the optimization of the weight corresponds to the minimization of the function W=a*t. So, in the plane (a,t) of figure 2, we plotted the iso-lines obtained by ranging the function W identified. The target

points of our research are the intersection of all these possible iso-lines with the "limit curve". Results are shown in next figure:

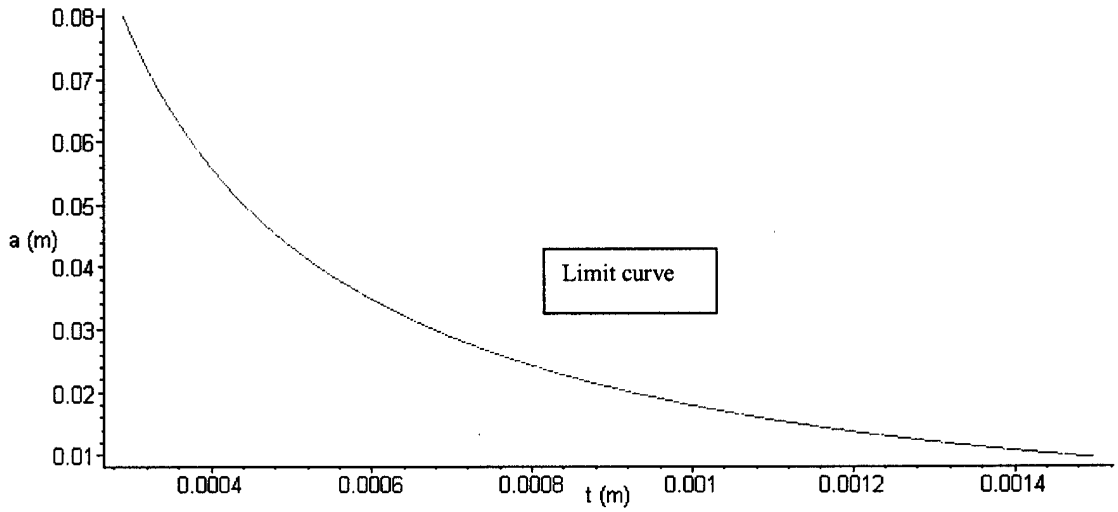


Figure 5.7

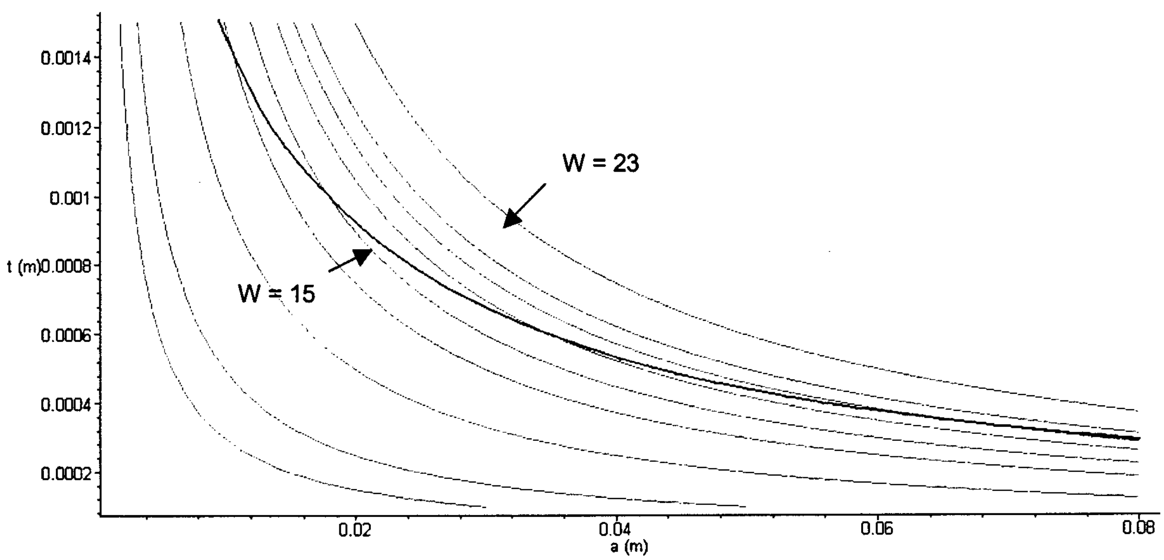


Figure 5.8

We deduced that the possible combination (a,t) that fits the condition vary only in the range $15mm^2 < W < 23\ mm^2$. The first value corresponds to a thickness of 1,5 mm and a tap's width of 10 mm. But we neglected this option because the thickness so high would require to decrease the thickness of the foam core to respect the line of the airfoil, and the dimension so little for the taps is not useful as a bonding surface for the laminate. So to gain sizes which would optimizes both the weight and the manufacturing easiness, we choose to move to the right along the "limit curve" and identify our final result: W=17 mm^2 , a=17 mm , t=1 mm, that means three layers.

Twisting study

Once fixed the geometry of the airfoil, to consider the torsion strength of the wing, we have to consider the thin walled sections shown in figure 5.9. Obviously, the actual resistant section consist of the forward D box, elements 1 and 2 in the figure. The study started by considering a possible critical condition form the point of view of a coupled twisting-flexural loading, as the one which could happen in a sudden pull up at a load factor of 2.3. So we assumed a twisting moment of the wing M_t = 20 Nm, considering a value for c_m equals to 0.17 and a safety factor of 1.3, and a shear force equals 100 N. We produced an analytical study: developing the equilibrium equations, we gained the one solution system which gave us the shear flows. Since the material we used to manufacture is a composite laminate, as shear strength we assumed the debonding shear, so we could size the thickness of our sections.

Figure 5.9

The results are: 0.2 mm for element 1 and 0.5 mm for element 2. That is: a single layer for the front box, and two layers for the tap's core.

Finite Element Method Analysis

After having produced the analytical study illustrated above, we made a numerical simulation to validate the results we gained. We modeled with a fem software, Femap V 7.0, the actual structural elements we had sized. We used 2D elements of laminate property and modeled the exercise condition of the wing by a fixed constraint at the root section and a vertical load of 100 N distributed on 5 nodes (i.e. 20 N each one), with regard the critical moment of the static inspection at the contest site.

By running the analysis with MSC Nastran 70.5 we valuated a maximum displacement of 5 cm at wing tip, that was well acceptable and agreed the analytical results. Then we calculated the distribution of Von Mises stresses and we studied the inter-laminar stresses within the plies of the gfrp spar by using the Hill failure criterion.

Figure 5.10 shows a contour plot of Hill failure indices on the spar, all below warning values.

Figure 5.11 illustrates the distribution of Von Mises stresses on the gfrp skin.

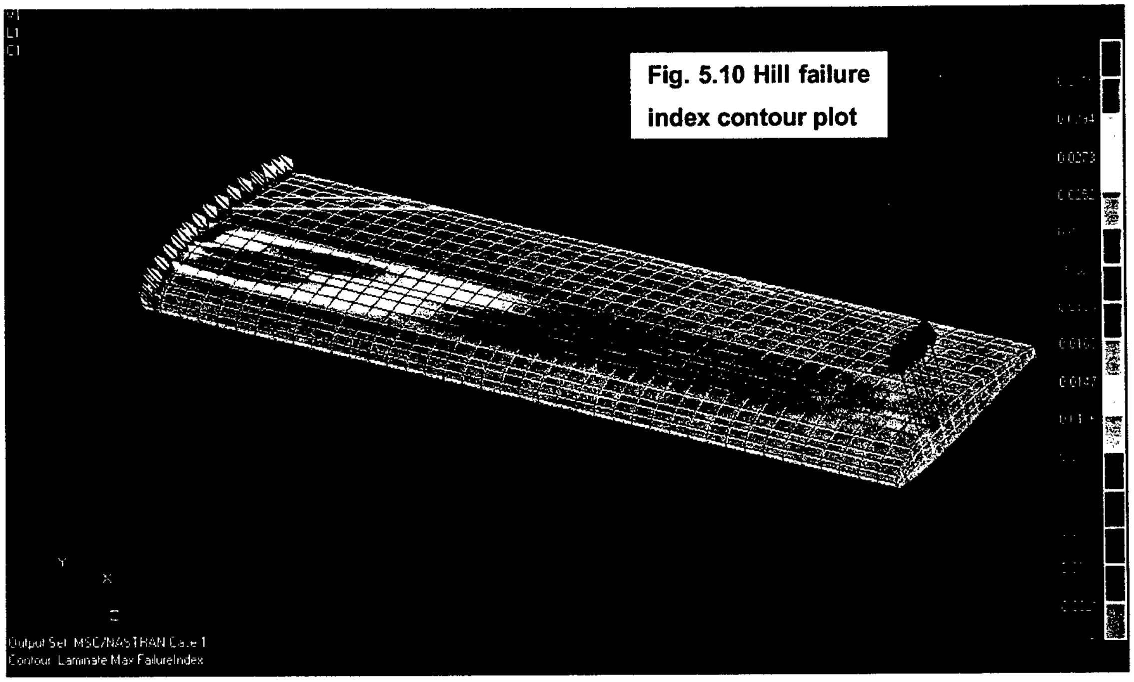


Fig. 5.10 Hill failure index contour plot

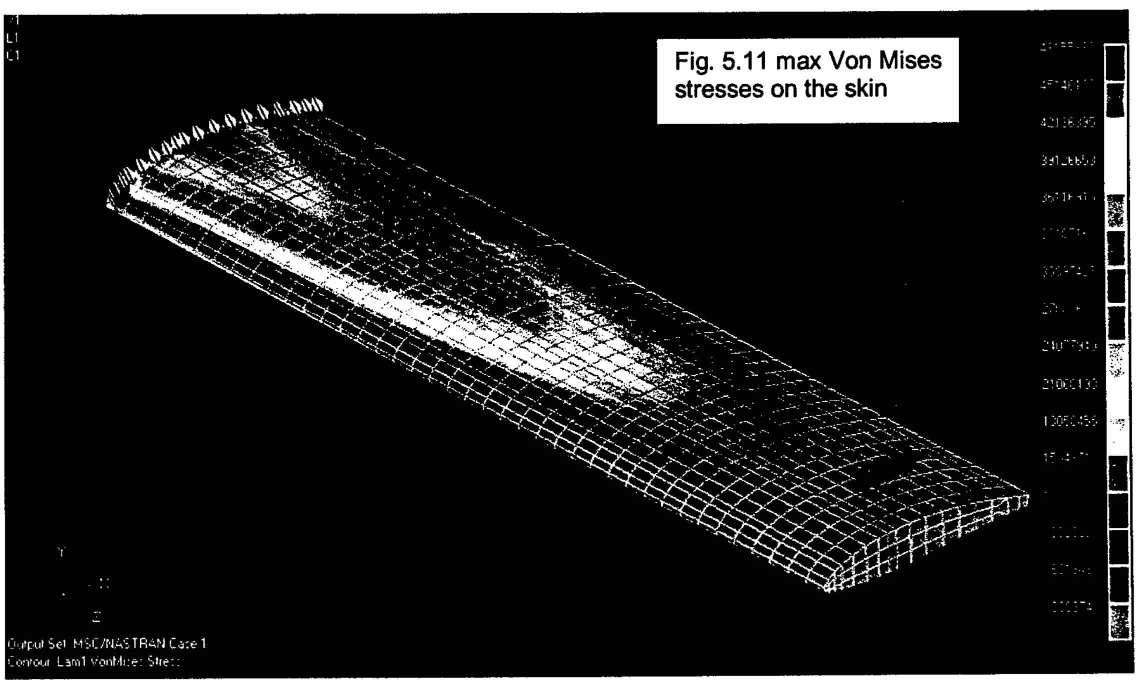


Fig. 5.11 max Von Mises stresses on the skin

Structural Tests

To value the results obtained in the analytical and numerical studies, we did a session of experimental tests to estimate the real stiffness and strength with respect the nominal data. We manufactured three half wings and collocated an electric strain gauge near the root section, at the point of maximum airfoil thickness (where the spar is located), to measure strains on the skin. Then we simulated the static inspection condition, i.e. we fixed the root section of the wing as a cantilever beam and applied to the tip an increasing load.

The measures chain consisted of:

1. a S/G bridge;
2. an amplifier;
3. a notebook computer for data acquiring, processing and visualization.

The S/G bridge works like a Wheatstone bridge. It uses the variation of the resistivity of conductor when it is deformed by a mechanic stress. In the following figure is showed the bridge electric scheme used in our test, a particular type of S/G half-bridge which has the following benefits:

- to avoid temperature effects;
- to avoid the effects of unwished tensile or compressive forces applied on the wing.

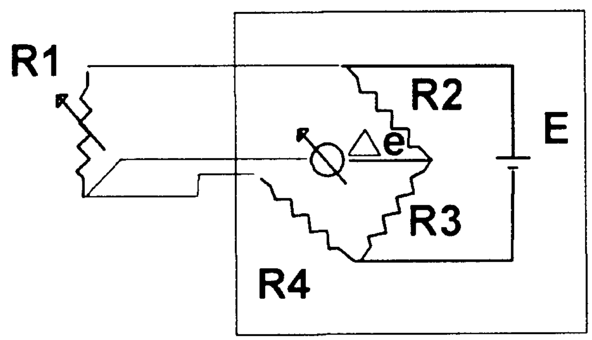


Fig. 5.12.

We used this particular half-bridge because of the asymmetry of wing section. The S/G and the resistor shown in figure 1 are R1 and R4 in the electric scheme. The S/G has been glued on the wing lower surface, 5 cm from root section upon the lower spar cap, by a phenolic resin.
The S/G strain is:

$$\varepsilon = \frac{4 \cdot \Delta e}{E \cdot F}$$

where:

Δe is the voltage measured at the bridge edges;

E is the supply voltage (2.5 V);

F is the gain factor.

We managed data acquisition and setting of the device by the HBM software supplied with the SCOUT 55 amplifier. In the following figure is shown an example of output.

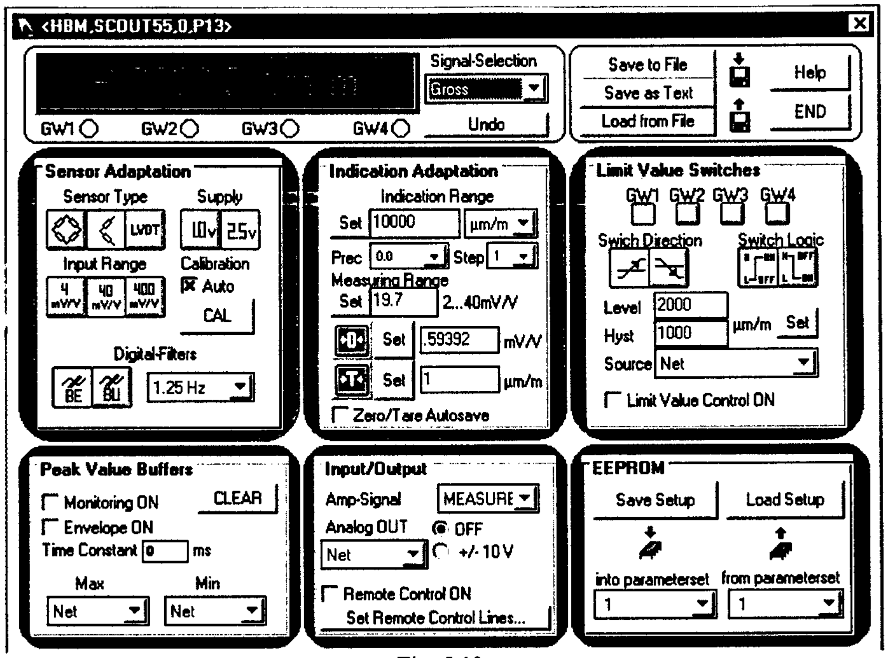


Fig. 5.13.

In the table are described the items used:

item	brand	model	characteristics
N°1 strain gauge	HBM	6/120 LY11	120 W g.fact. 2.03 ±1%
N°1 amplifier	HBM	SCOUT 55	Mono channel amplifier for measure from passive transducer
N° 1 notebook	ASUS	A 1000	Pentium III 700 MHz 256 MB ram.

To load the wing we used plastic water bottles of different sizes: 0.5, 1 and 1.5 litres. This way we could have loads varying from about 0.5 kg to 6 kg. Data analysis for the three specimen are summarized in following table:

	Wing 1	Wing 2	Wing 3
load [kg]	strain[µm/m]	strain[µm/m],	strain[µm/m]
0	0	0	0
0,53	84	91	85
1,05	140	153	142
1,56	257	281	257
2,1	350	381	375
2,61	427	511	472
3,12	518	643	561
3,65	650	the wing broke	683
4,69	817		the wing broke
6,25	1201		

We plotted data in three different diagrams, figure 1, 2 and 3, and we noticed a good agreement for the three trend lines. An unexpected result was the difference within the loads that broke the wings. Inspecting the fracture points, we could notice in the crack area the glass fibres were not properly impregnated of resin. Actually, three different groups of people had build the three specimen. So we learned a very important lesson: manufacturing composite materials requires a proper check of quality. To take into account this effect, we decided to consider in the next structural sizing a corrective coefficient to be applied on the value of the nominal strength of the composite material. In our case this coefficient is determined by the most performing wing because built it for last and with more accuracy. The coefficient is equals to 5. By analysing the same data, calculating the inertia moment of the resisting section we deduced the value of the normal stress in the skin. So we plotted in figure 4 the experimental diagram σ–ε for the first wing. We obtained a value of the elastic modulus equals 20 GPa, exactly the one we used in the sizing study, while the value of strength was so different. This result validate our first impression: the offset from nominal behaviour of the material is certainly due to local manufacturing faults.

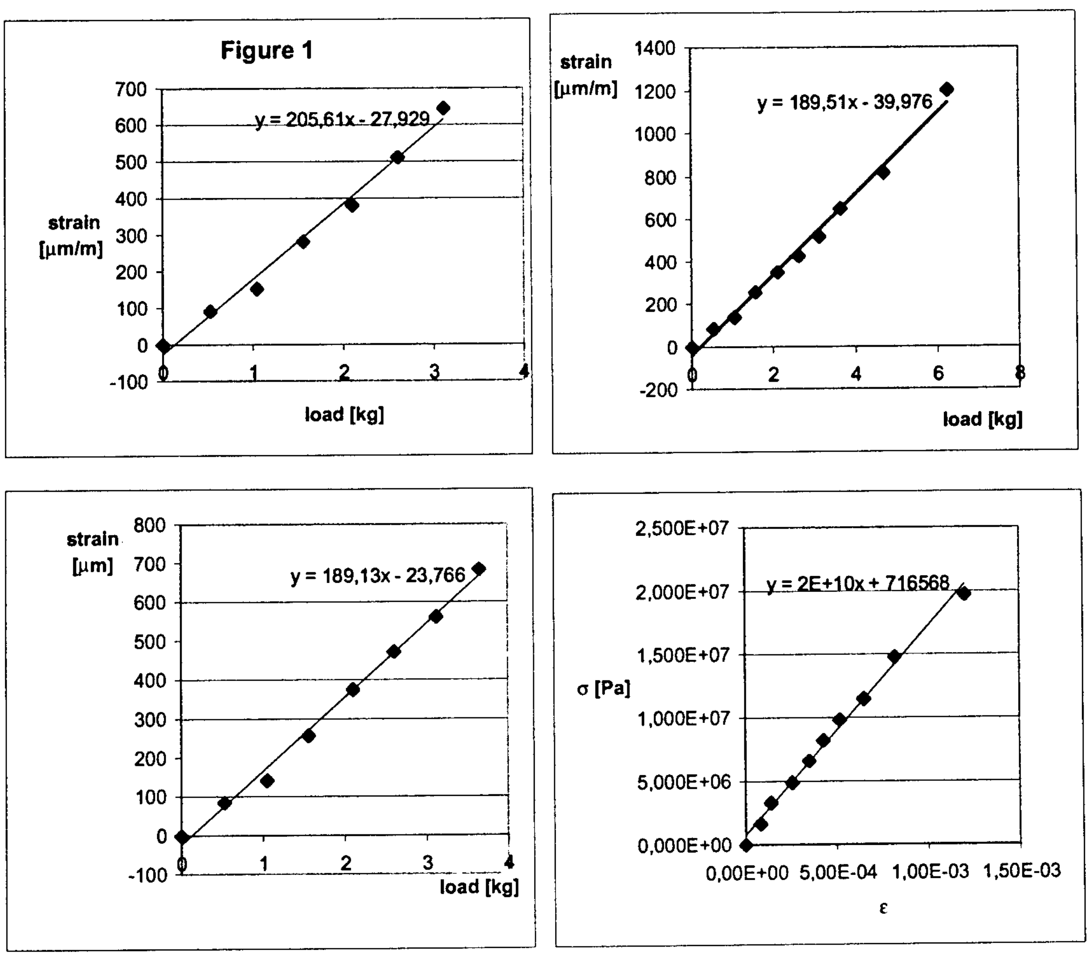


Fig. 5.14.

Using corrective factor cited before we sized a new wing based on same tools we used in the previous study. The Fig 5.15 shows final solutions we adopted.

It is evident on the figure the use of carbon fibers ribbon instead of glass ones. The reason of this choice is in the bigger value of carbon strength.

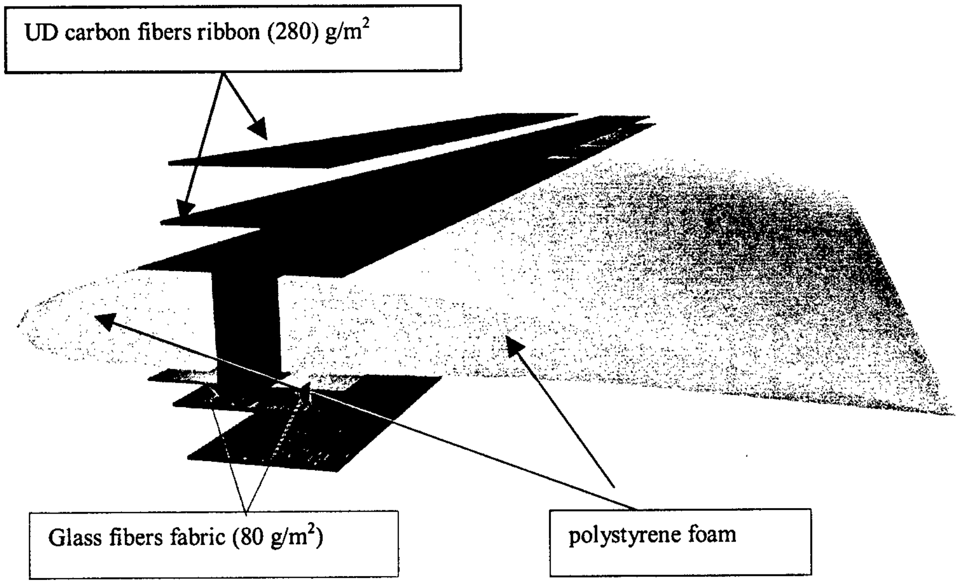


Fig 5.15.

Dynamic Testing

We have made a dynamic test on a half wing. We performed a test with an impact hammer with two piezoelectric accelerometer, after fixing the wing at root. The accelerometer on the hammer's head was a PCB 356B18 with a shock limit of 500 g, while the measurement accelerometer was a PCB333B30 with a 50g shock limit. The acquisition system was made of a 16 bit National board NI DAQ Card AI 16XE 50, with 16 channels and a scan rate of 20000 samples per second. We have meshed the wing surface with a 5 pts. Grid and after 125 measurement we have obtained the function transfer matrix, from which we have obtained the Resonant Frequency. Because of the high damping property of polystyrene foam we have been able to visualize only the first three. In order to elaborate data we have used "*elaborazioni.vi*", a program that we have made to run under LabView ambient.

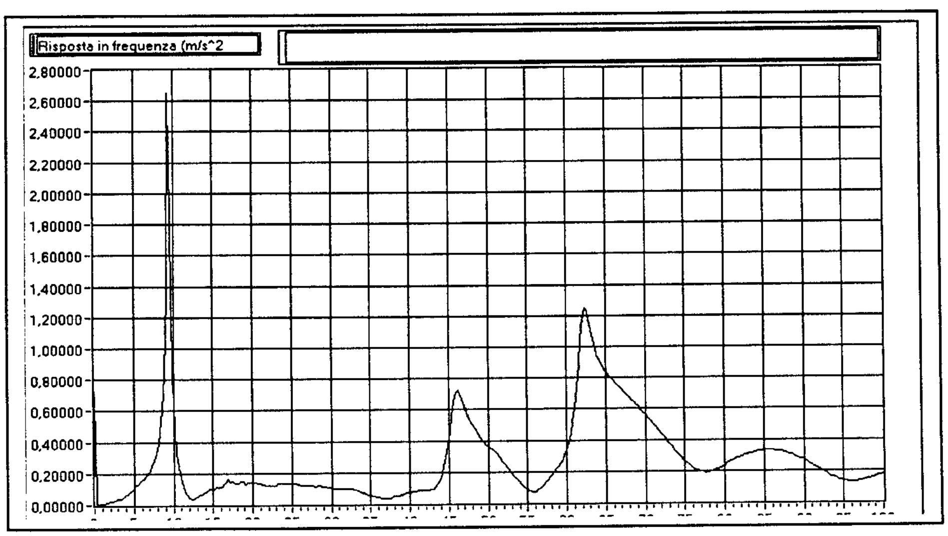


Fig. 5.16

As it can be seen in the Fig. 5.16, the frequency we found were:

ω_1	ω_2	ω_3
9.00125 Hz	46.99701 Hz	62.98261 Hz

5.4.2 Fuselage

We decided a sandwich structure would have been the best solution for the fuselege's components. The factory that has supplied us with the materials gave us only a type of honeycomb, 8 mm high. We considered two skins of cfrp coupled with this core and verified this solution guaranteed enough stiffness and strenght calculating the deformation and the stresses on the surfaces more critical. For the lower plate we used an analytical tools, the method of Galerkin, supposing the plate pinned at all its four edges. For the lateral ribs, loaded by the wing root sections and by the junction spar, we used a numerical simulation with a fem software, MSC Nastran. We estimated a structural optimization, regarding honeycomb thickness, would not have allow us to save a significant percentage of the total weight. So, as already explained, we produced all the pieces of the fuselage using the same sandwich. Table 1 and 2 show the mechanical charachteristics of the material used in the analyses.

ICI Fiberite 934 cfrp 200 g/mq, fabric 0/90 prepreg		
Fuselage sandwich skin		
ρ	666,7 Kg/m^3	volumetric mass density
t	0,3 mm	thickness
E_1	62 GPa	elastic modulus in direction 1 and 2
G_{12}	5,5 GPa	12 shear modulus
G_{13}	10^7 Pa ≈ 0	13 shear modulus
G_{23}	10^7 Pa ≈ 0	23 shear modulus
ν	0,325	Poisson ratio
$\sigma_{1comp}=\sigma_{2comp}$	552 MPa	compression strength in direction 1 and 2
$\sigma_{1traz}=\sigma_{2traz}$	586 MPa	tensile strength in direction 1 and 2
τ	30 MPa	shear strength

Hexcel HRH-10 – 3/16 – 3.0[1] (2) hexagonal Nomex, (3/16 inch), thickness 8 mm[2]		
fuselage sandwich core		
ρ	80,77 Kg/m^3	volumetric mass density
t	8 mm	thickness
d	3/16 in = 4,76 mm	size of the cell
E_1	100 Pa	elastic modulus in direction 1 (L direction)

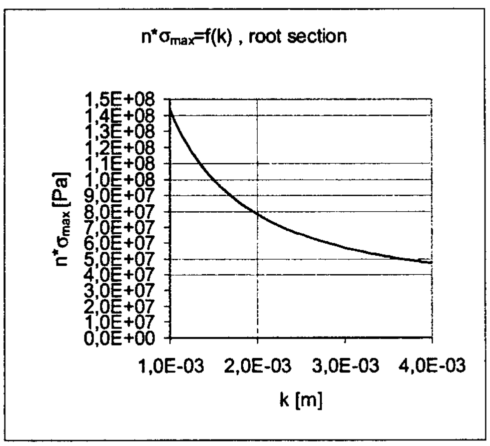
[1] It's the volumetric mass density of the hopneycomb 3/16 in thick, pound/ft^3.

[2] All the values (except the density ρ) are calculated for a thickness equals 0,5in=12,7mm, as detailed by Hexcel; since the actual honeycomb is 8mm thick, values are under estimated.

E_2	100 Pa	elastic modulus in direction 2 (W direction)
E_{3comp}	138 MPa	elastic modulus in direction 3 (compression only)
G_{12}	1000 Pa	12 (LW) shear modulus
G_{13}	40 MPa	13 (L3) shear modulus
G_{23}	24,15 MPa	23 (W3) shear modulus
v	0,25	Poisson ratio
$\sigma_{1comp}=\sigma_{1traz}$	1000 Pa	compression/tensile strength in direction 1 (L)
$\sigma_{2traz}=\sigma_{2comp}$	100 Pa	compression/tensile strength in direction 2 (W)
τ_{13}	0,89 MPa	13 (L3) shear strength
τ_{23}	0,48 MPa	23 (W3) shear strength
σ_{3comp}	1,45 MPa	compression strength in direction 3

5.4.3 Empennage

The spar made of lime (*tilia cordata*) was sized to resist normal stresses induced by a distributed load of 40 N (including a loading factor of 2) acting along the half tail span, considering an insurance coefficient n=1,5. We fixed the width of spar's caps to 8mm and let its thickness vary to choose the minimum value satisfying the strength requirement. In figure 5.17 is plotted the maximum stress (root section) multiplied by the insurance coefficient as a function of caps thickness. We could deduce that a value of 2 mm was enough. Then we evaluated the shear stress at the root section. Its value, multiplied by the insurance factor, is plotted in figure 2 as a function of spar's core thickness. We could assume a value of 2mm.



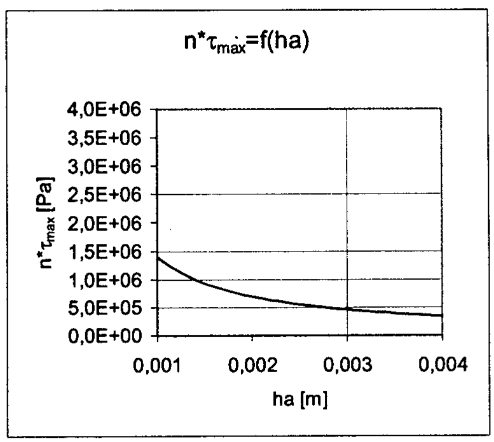


Fig. 5.17.

5.4.4 Landing Gear Shaping, Sizing and Positioning

From earlier studies we had chose a fixed (non retractable) front tricycle landing gear. That meant the main (after) carriage should be of typical solid spring construction, so to support the major percentage of the plane's weight during taxi phases of the mission and to absorb nearly all the vertical kinetic energy at landings. We planned to optimize weight and shape by produce analytical and numerical studies of a same form carriage realized in same different materials. We thought Aluminum and Kevlar were ideal because of their flexibility, while carbon fiber could guarantee a more lightweight structure. We wanted to assure enough strength, and to limit deflection to prevent the propeller or the long after fuselage to iMPact the ground at

landings. The need of sufficient strength moved us to consider a 3-g maximum gross weight load acting on all the two tips of the main (after) carriage. While the requirement of limiting deflection drove us to fix an allowable range for vertical displacement of carriage tips within 5 and 20 mm, when subject to the iMPact load. The optimization process would have consist in a parametric analysis: we should vary thickness and materials combination (the second only for the composite options, obviously) until the requirements identified were satisfied with the minimum weight configuration.

What about the shape? The structural team considered a few possibilities about the geometry of the main carriage, while for the front leg it seemed to be quite obvious to choose a linear beam shape, because the lower wetted surface means the lower aerodynamic drag, and considering not much important the structural requirements in shaping an element slightly loaded. We focused on two possible geometry of the after carriage: a classical and an inverse leaf spring. We thought the first could guarantee more structural stiffness, without any particular disadvantage in the manufacture. Then, we wondered if it should be designed as two separated legs connected at the lateral surfaces of the fuselage or as one only element joined at the bottom of the pay load plate. Once more structural reasons and the easier assembly drove us to the second option, even if the first could assure a lower weight.

Only after having made the decision about shape and materials, and having sized them in detail, we would have considered the opportunity to add a fairing to reduce drag during flight, as we did in last year competition. We delayed this choice until the flight tests.

We remembered at this point that landing gear affects the performance of an airplane in the air and on the ground. We had not to neglect the behavior of the wheels to design the best carriage subsystem! In fact, the long series of problems with last D/B/F UAV, Galileo II, began exactly with a fail of the wheels on the asphalt of Websterfiled's runway! About gear wheels, we decided to find them within commercial items. So, we looked for them in the world of RC modelers, and managed to procure a series of wheel of different toughness (and different weight, actually!), so to test them in flight sessions and choose the ones which best fitted the target requirements, as enough flexibility, slide ness and reliability. We decided main gear to support 85% of the total weight, so it should be placed at 5,7 cm after the center of gravity.

Now we made a simple analytic calculation of the structure under the mentioned loading condition using a typical aluminum isotropic material to evaluate the order of magnitude for the thickness; it resulted it should be around 4 mm. Then we produced several finite elements runs, combining thickness and materials, sizing some carriage respecting the requirements we had chosen. (We have to say the choice of synthetic fibers we considered was quite limited because of their availability at the "Sistema Compositi" facilities, the factory which supplied us with materials). And we had a result which surprised us. In fact it seemed a carbon carriage would have be too much rigid, and it would have damage the fuselage in a sudden landing. The aluminum was good, but a combination of carbon and Kevlar was the best: it could assure a structure enough flexible and lightweight. Only it presented displacement at the tips a little higher than the limit we fixed; we noticed it could be reduced by adding a simple nylon tie-rod. We concluded two solutions were acceptable: we chose as our default carriage the composite one, but we planned also to manage a second carriage in aluminum, remembering its optimum behavior as our historical option in previous D/B/F. If flight tests showed the first was not enough reliable, we would have adopted the second.

We are illustrating in the following same aspects of the fem analysis. The software we used in calculation is MSC Nastran v.70.5; the software for modeling was Femap v.7.0. Tables 1, 2 and 3 show the mechanical

characteristics of the composite materials used, two 0/90 prepreg fabric. We modeled the elements as 2D orthotropic laminate. We selected the Hill failure criterion for the laminate. Figure 5.18 shows a contour plot of the Hill failure index for the loading condition mentioned and one of the constraint set we considered, as the fitting at the fuselage edges, which is a conservative case.

Table 1: Hexcell afrp K285 F155 (42% fiber) 170 g/mq		
t	0,4 mm	thickness
ρ	425 Kg/m^3	volumetric mass density
E_1	28,3 GPa	elastic modulus in direction 1
E_2	28,3 GPa	elastic modulus in direction 2
G_{12}	4 GPa	shear 12 elastic modulus
G_{13}	10^7 Pa ≈ 0	shear 13 elastic modulus
G_{23}	10^7 Pa ≈ 0	shear 13 elastic modulus
ν	0,3	Poisson ratio
$\sigma_{1comp}=\sigma_{2comp}$	221 MPa	compression strength
$\sigma_{1traz}=\sigma_{2traz}$	483 MPa	tensile strength
τ	48,3 MPa	shear strength

Table 2: cfrp300 g/mq		
t	0,4 mm	thickness
ρ	750 Kg/m^3	volumetric mass density
E_1	65 GPa	elastic modulus in direction 1
E_2	65 GPa	elastic modulus in direction 2
G_{12}	4,2 GPa	shear 12 elastic modulus
G_{13}	10^7 Pa ≈ 0	shear 13 elastic modulus
G_{23}	10^7 Pa ≈ 0	shear 13 elastic modulus
ν	0,325	Poisson ratio
$\sigma_{1comp}=\sigma_{2comp}$	500 MPa	compression strength
$\sigma_{1traz}=\sigma_{2traz}$	550 MPa	tensile strength
τ	30 MPa	shear strength

Table 2: aluminum 2024T5		
t	4 mm	thickness
ρ	4165 Kg/m^3	volumetric mass density
E	73,77 GPa	elastic modulus
G	27,73 GPa	shear elastic modulus
ν	0,33	Poisson ratio
σ_{comp}	275,79 MPa	compression strength
σ_{traz}	296,47 MPa	tensile strength
τ	262 MPa	shear strength

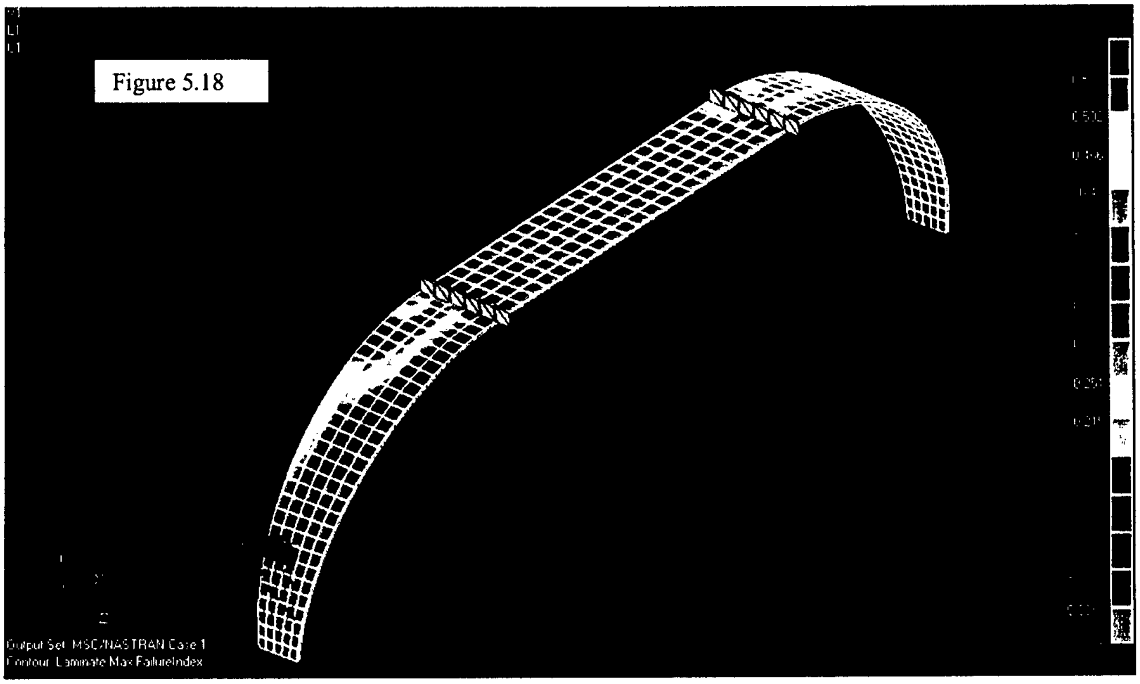


Fig. 5.18

5. 5 Aerodynamics

The Aerodynamic group on the base of wind tunnel test's results and considering the responses of a numerical simulation made using a commercial software for CFD, was able to product final results about lift, drag and efficiency of the whole UAV. In tests and calculation we did not modeled the influence of the propeller. Results are showed below. Figure 1 illustrates the plane drag polar; figure 2, 3 and 4 show the lift and drag coefficients and the efficiency diagrams as a function of the angle of attack alpha. In the figures is considered only the linear part of the Lift function. On the base of such results we can conclude the aerodynamic performances of the UAV designed fit the targets we had individuated.

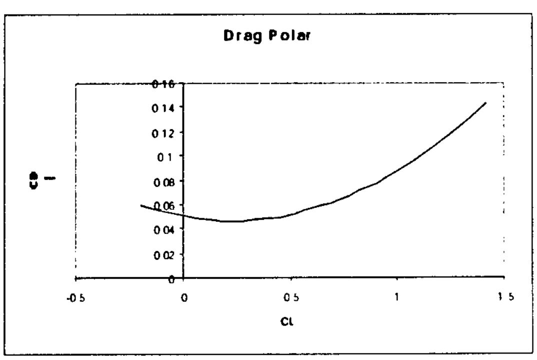


Figure 1

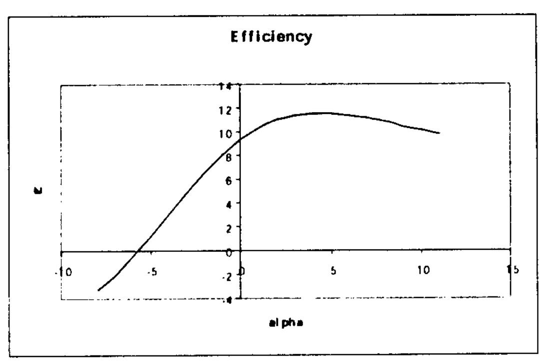


Figure 2

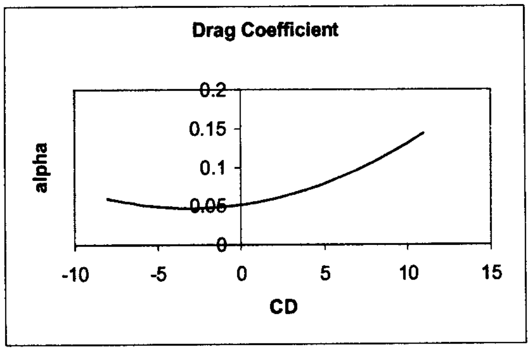


Figure 3

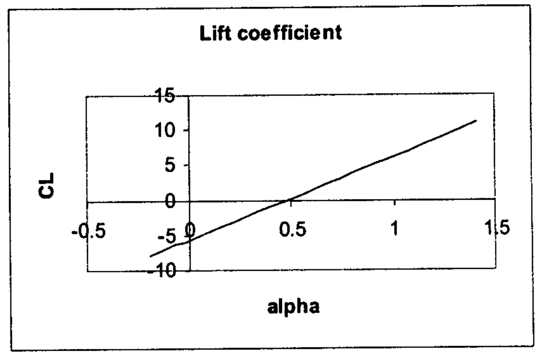


Figure 4

Manufacturing Plan

Like last year Galileo's manufacturing process has been executed in the Lapis laboratories. In the last two years we participated with wooden models made of wood; the need of a much more competitive airplane brought us to use mainly composite materials for structural parts of our model.

For this reason we have achieved a partnership with an important industry in field of composite materials, "Sistema Compositi"; with their facilities and know-how we have been able to get skilled with this kind of manufacturing.

Choosing the right manufacturing process of each component has been fundamental to balance mechanical properties, costs, weights, and manufacturability

To manufacture the composite and wooden parts of aircraft we investigated several production methods. Figures of merit has been used to reach a final choice:

- **Cost:** be have paid particular attention to this aspect because of limitation in budget (except for composite furniture).
- **Know How:** is a value of the skill in a specific manufacturing process.
- **Manufacturability:** is the team capacity to made it.
- **Time:** indicates the time needed to manufacture the piece.
- **Weight:** the lower the best.
- **People:** number of persons needed to manage a particular manufacturing process.

Each FOM has been evaluated by a score from 0 to 3 (worse to better).

We don't care about facilities needed for composites' manufacturing, because of we were allowed to use the factory of "Sistema Compositi".

	Cost	Knowledge	Manufacturability	Weight	People	Time	TOT
FUSELAGE							
Frames	1	2	2	2	1	1	9
Carbon Skin	3	2	1	3	1	0	10
Sandwich Panels	3	2	3	3	1	1	***13***
WING:							
Wooden Ribs	1	2	2	2	1	1	9
Carbon Ribs	3	1	1	2	1	1	9
Foam core Wood skin	0	1	2	3	1	1	8
Foam core Carbon skin	3	2	3	3	1	1	***13***
EMPENNAGE:							
Wooden Ribs	1	2	2	3	1	3	***12***

Carbon Ribs	3	1	1	1	1	1	8
Foam core Wood skin	0	1	2	1	1	1	6
Foam core Carbon skin	3	2	3	1	1	1	11
LANDING GEAR:							
Aluminuim	0	2	1	2	1	2	8
Kevlar	3	2	3	2	1	1	***12***
NOSE:							
Ribs	1	1	1	2	2	1	8
Skin	0	3	3	2	2	3	***13***
Sandwich	0	1	0	2	2	1	6

From this chart we have chosen the way to made every part of new model. So we briefly describe the manufacturing processes to be used in the fabrication and assemblage of major components of the aircraft while the details can be found in other part of this report.

Fuselage.

Sandwich structured laminates will be manufactured assembling a honeycomb core (Nomex) and two cfrp skins.

All fuselage parts will be cut by numeric control machinery. A further cfrp skin will reinforce highly stressed areas, like links of wings and of landing gear, and nose.

The parts will be assembled using epoxy glue. Servos and control linkages will be installed last. The fuselage was covered with oracover.

Nose.

The fiberglass will be laminated on a mould. Cfrp strengthening will be added where motor and battery pack will be placed.

Wings and Empennage.

The wing and empennage are sensitive components of any airplane. It is imperative that the wing and tail provide the calculated aerodynamic characteristics while maintaining their structural integrity. To manufacture the wing we will laminate fiberglass on polystyrene foam, in the same way we will make the nose.

Empennage will be manufactured according to classical configuration with balsa wood ribs and lime wood spars. A carbon rod will be the junction component with fuselage.

Landing gear.

The landing gear will be manufactured by a manual lay-up of Kevlar fabrics (for inner layers) and carbon fabrics (for external layers).

	JANUARY													
	14	15	16	17	18	21	22	23	24	25	28	29	30	31
	M	T	W	T	F	M	T	W	T	F	M	T	W	T
FUSOLAGE:														
mould	■	■												
sandwich preparation			■											
cutting				■										
assambly					■	■	■	■						
cover									■	■				
finish														
WINGS:														
mould											■	■		
lamination													■	
curing														■
finish														
EMPENNAGES:														
ribs														
assambly														
LANDING GEAR:														
mould														
lamination														
curing														
finish														
testing														
NOSE:														
mould														
lamination														
curing														
finish														
TESTING:														
static and dinamic tests														
COMPONENTS:														
servo and control installation														
landing gear e motor														
FINAL PREPARATION:														
test mounts and control linkages														
oracover on components														
FLIGHT OFF														

	FEBRARY																			
	1	4	5	6	7	8	11	12	13	14	15	18	19	20	21	22	25	26	27	28
	F	M	T	W	T	F	M	T	W	T	F	M	T	W	T	F	M	T	W	T
FUSOLAGE:																				

mould
sandwich preparation
cutting
assambly
cover
finish

WINGS:

mould
lamination
curing
finish

EMPENNAGES:

ribs
assambly

LANDING GEAR:

mould
lamination
curing
finish
testing

NOSE:

mould
lamination
curing
finish

TESTING:

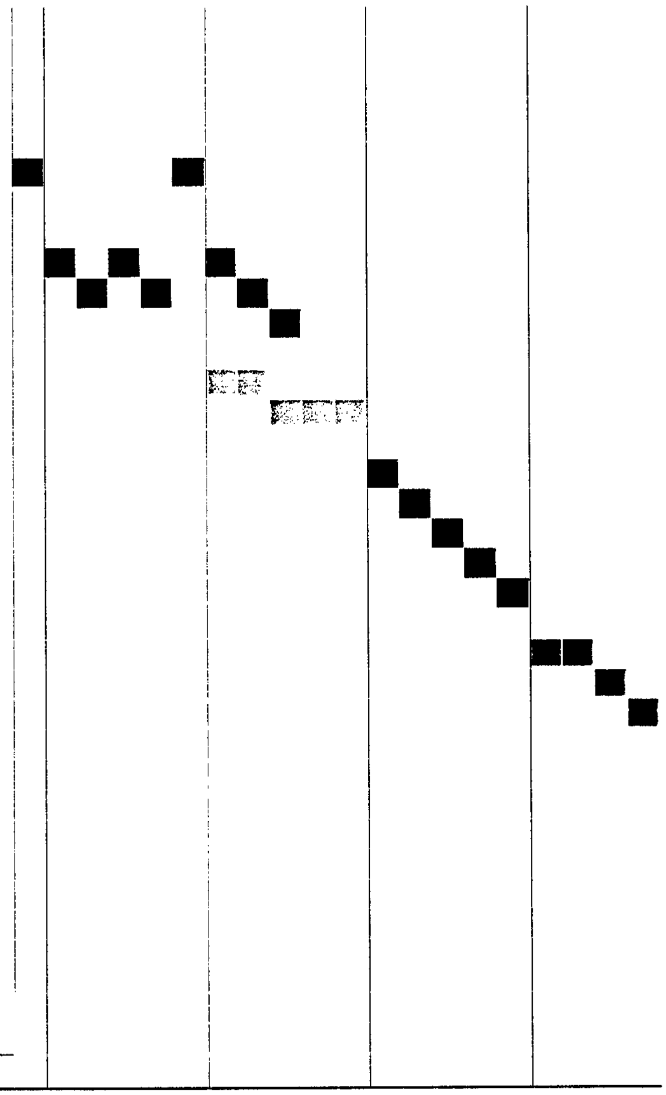
static and dinamic tests

COMPONENTS:

servo and control installation
landing gear e motor

FINAL PREPARATION:

test mounts and control linkages
oracover on components



FLIGHT OFF

	MARCH																			APRIL
	1	4	5	6	7	8	11	12	13	14	15	18	19	20	21	22	25	26	27	28	1
	F	M	T	W	T	F	M	T	W	T	F	M	T	W	T	F	M	T	W	T	M
FUSOLAGE:																					
mould																					
sandwich preparation																					
cutting																					
assambly																					
cover																					
finish																					

WINGS:	
mould	
lamination	
curing	
finish	
EMPENNAGES:	
ribs	
assambly	
LANDING GEAR:	
mould	
lamination	
curing	
finish	
testing	
NOSE:	
mould	
lamination	
curing	
finish	
TESTING:	
static and dinamic tests	
COMPONENTS:	
servo and control installation	
landing gear e motor	
FINAL PREPARATION:	
test mounts and control linkages	
oracover on components	
FLIGHT OFF	

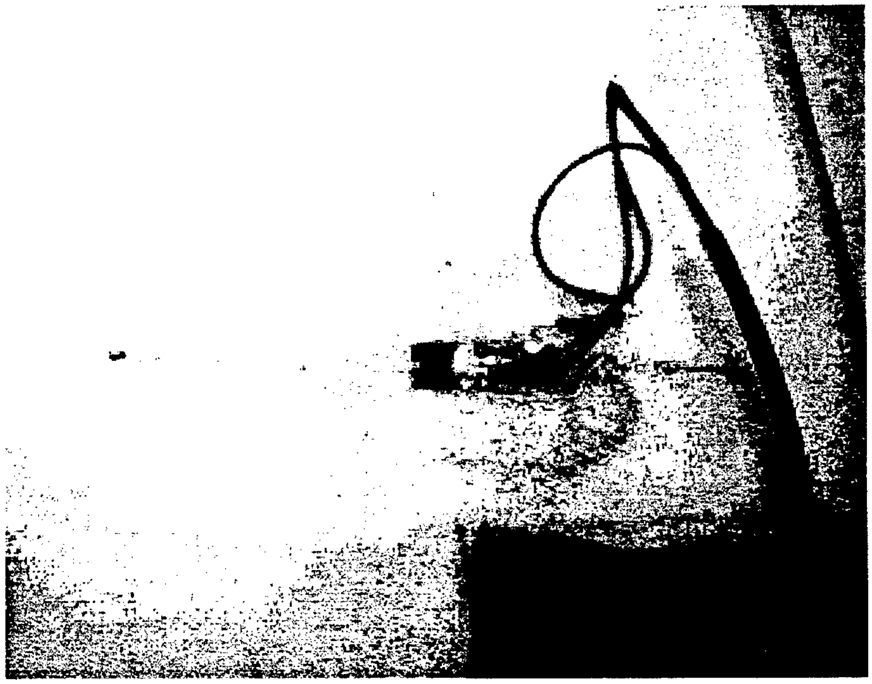


Photo 1. The SG bonded near he root section

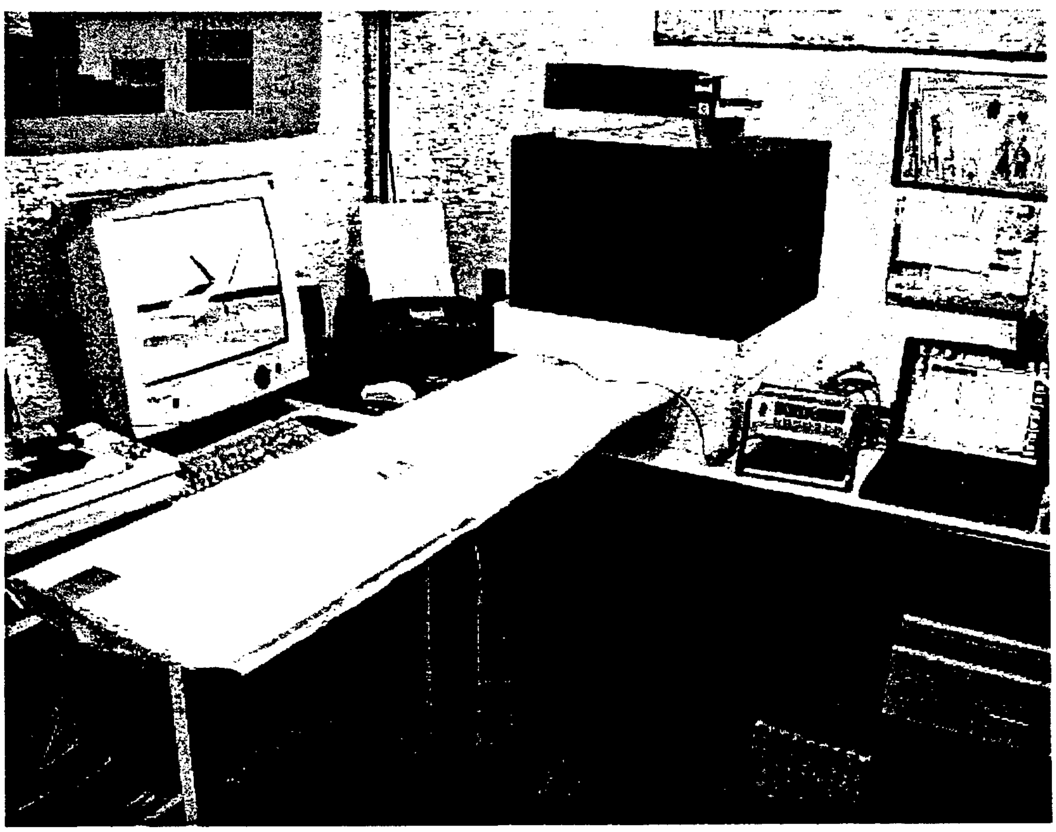


Photo 2. A panoramic view of test facility. Half wing is ready to be tested.

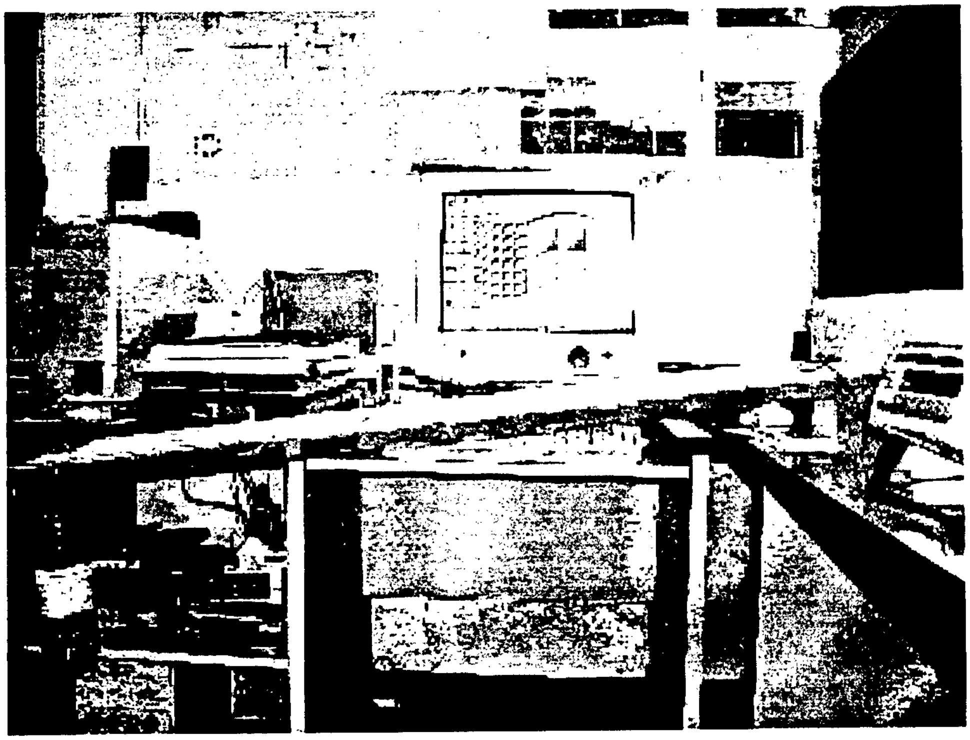
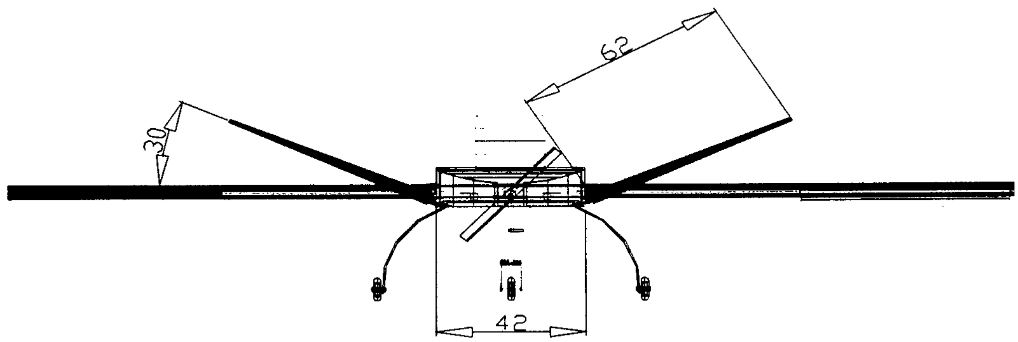


Photo 3. Tip deflection under "water" load condition.

Photo 4. Particular of the failure region.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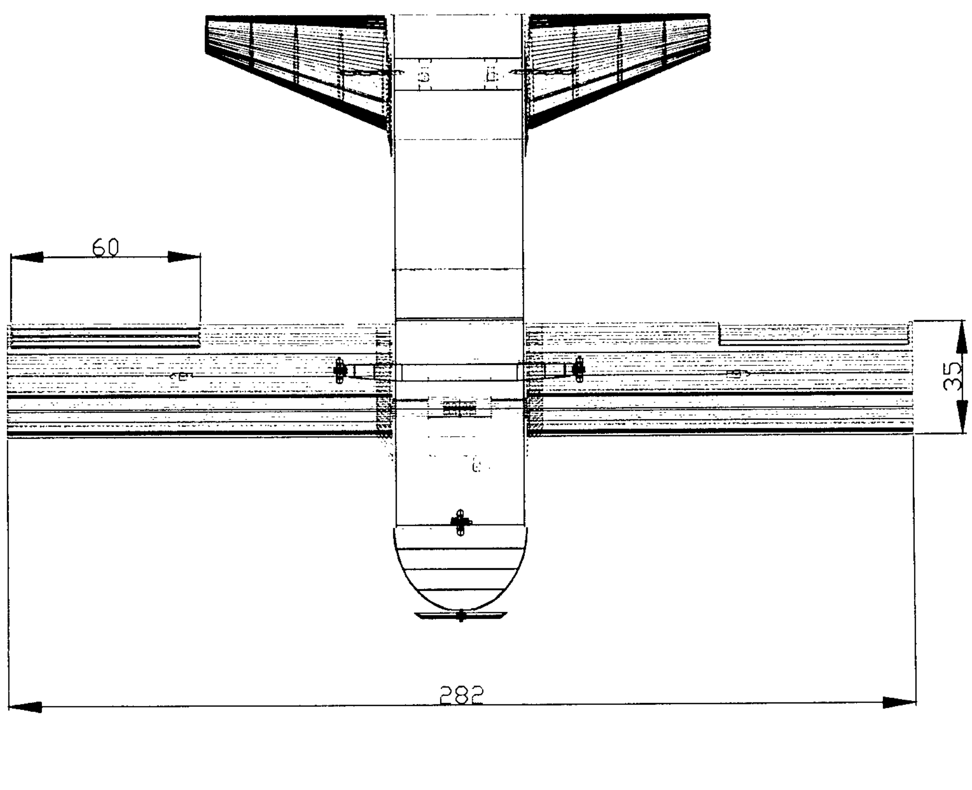
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35
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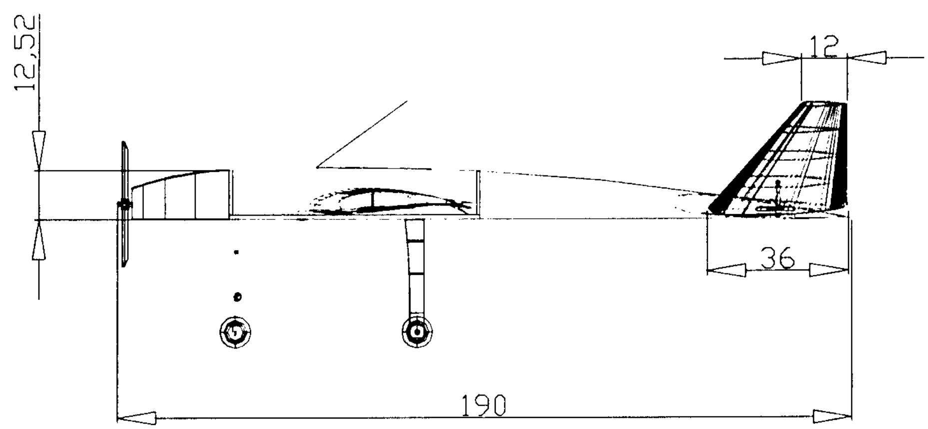
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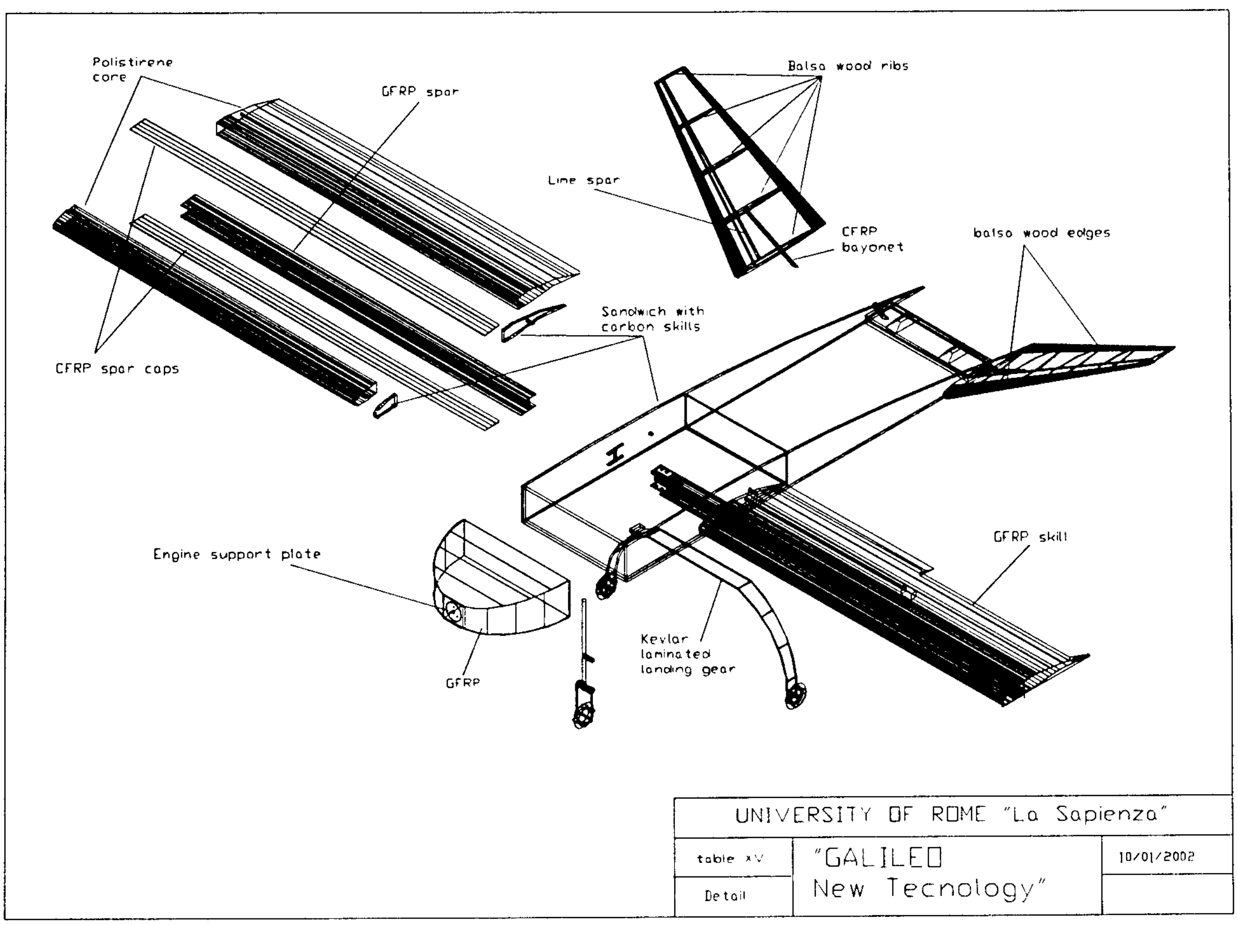
12
36
190


Polistirene core
GFRP spar
Balsa wood ribs
Line spar
CFRP bayonet
balsa wood edges
Sandwich with carbon skills
CFRP spar caps
GFRP skill
Engine support plate
Kevlar laminated landing gear
GFRP
UNIVERSITY OF ROME "La Sapienza"
table XV
"GALILEO
New Tecnology"
10/01/2002
Detail

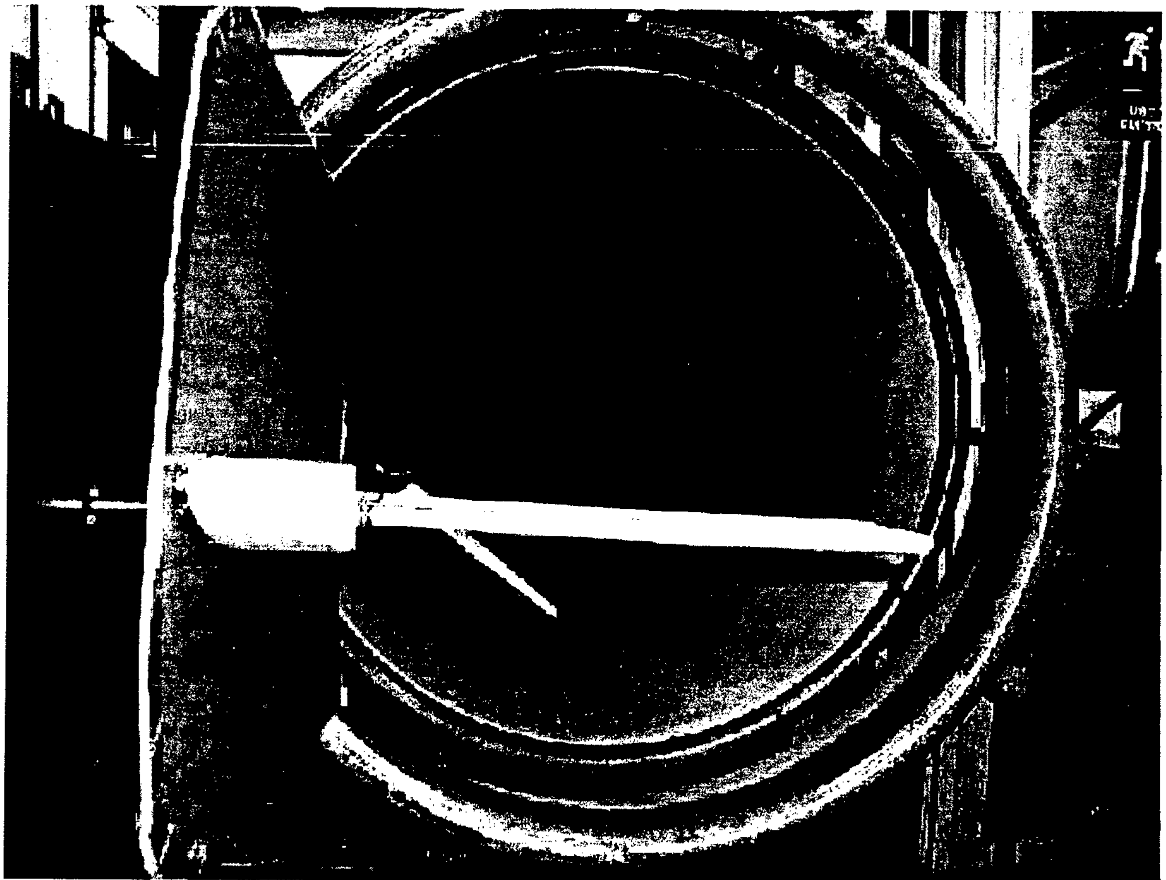
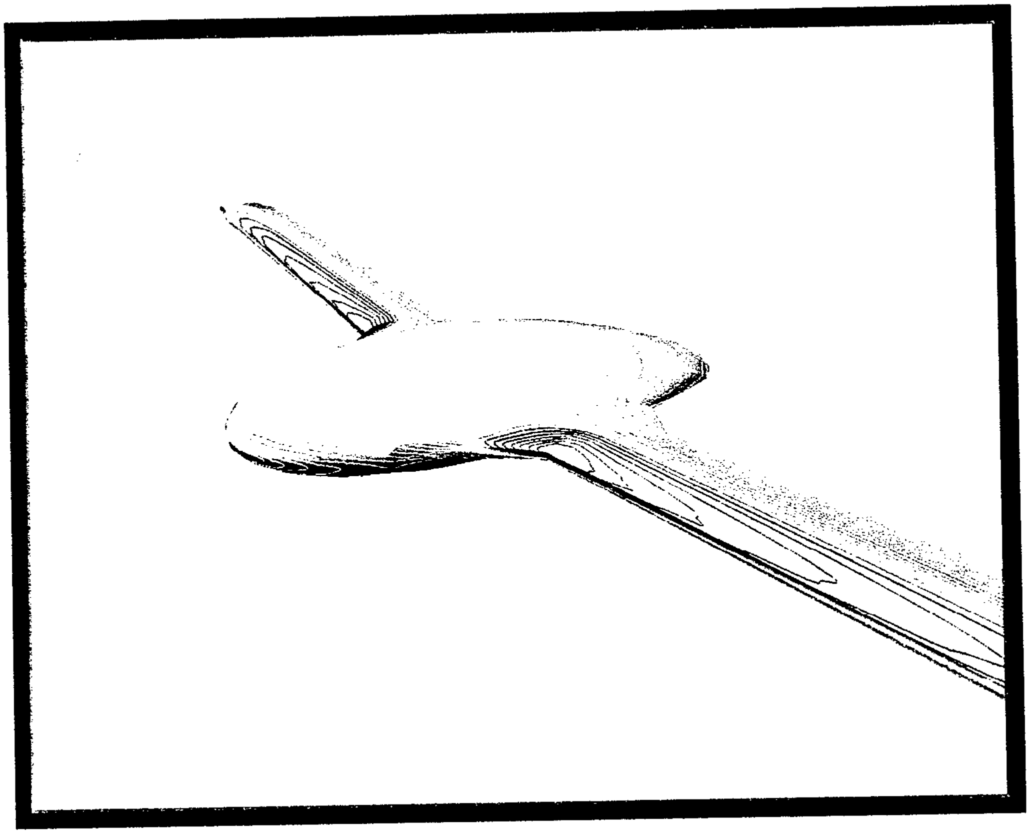


Photo 5. The model in the wind tunnel for aerodynamic tests

2001/2002 AIAA

Cessna/ONR Student Design/Build/Fly Competition



San Diego State University

Monty's Revenge

Table of Contents

1. Executive Summary

Organized by Cessna and ONR (Office of Naval Research), the Design/Build/Fly competition places college students in charge of creating an aircraft capable of performing a specific task. Basic aircraft design requires that it be unmanned, electric-powered, and radio controlled.

This year's task involves three separate missions within a 10-minute flight time. One of the missions requires the airplane to carry up to twenty-four softballs; the other two missions are flown with no payload.

The competition relies not only on the ability to fly a plane, but to use teamwork to design and build a competitive model. Success is measured by achieving the highest possible score at the contest. Scoring involves three main factors: written report score, total flight score, and rated aircraft cost.

In order to accomplish the goal, the team met in June 2001 to begin the process. The design process involved three phases: conceptual design, preliminary design, and detail design. Once designed the construction team can build the airplane.

1.1 Conceptual Design

Throughout the conceptual, phase the entire team was involved in creating ideas that could be used for the final design. During the eight weeks set aside for the conceptual phase, the team met regularly to discuss their ideas. Discussion focused on the conceptual design of the body, wings, and power plant in which no idea was discounted. Each concept was then broken down into its component parts and evaluated for its "figures of merit." Figures of merit uses a number system, defined by the design team, to rate each concept. Any idea that passed the screening process was further evaluated.

1.1.1 Alternative Designs

Various design ideas were considered. Configurations such as number of wings, wing placement, tail design, number of engines, engine location (pusher or tractor), fuselage design, and payload placement, were discussed. Assumptions were made about wing design. Tail options included the standard vertical/horizontal stabilizer, multiple vertical stabilizers, no tail, or V-tail. Number and types of engines were looked at along with the propeller size and pitch.

Each conceptual design was evaluated to maximize the score as much as possible. Actual cost of constructing the design was also considered for a few factors.

1.1.2 Design Tools

Basic designs were created in Pro-Engineer (CAD) to help visualize each concept. Using the basic designs, each was evaluated in order to narrow the search for the best conceptual design.

Maximizing the score by minimizing the Rated Aircraft Cost (RAC) was the first consideration in choosing a preliminary design. An EXCEL spreadsheet was created using the RAC factors so that each configuration could be quickly evaluated.

To determine the power plant requirements, a computer program called ElectriCalc was used. It allowed evaluation of many different types of Astro Flight and Graupner motors with various types of propellers.

Difficultly of construction was evaluated by the construction team in order to determine the ease of building the design. Consultation with the construction team and relying on their airplane building expertise was crucial to finalizing a design.

Actual cost was a major concern for the successful completion of the project. Some of the configurations were looked at from a fiscal standpoint and evaluated as such.

1.1.3 Results of the Conceptual Phase

Ending of the conceptual phase resulted in a configuration for the team to begin further research. Basic plane design was determined to be a low-wing monoplane with a lifting body fuselage, a single motor in a tractor configuration, T-tail design, and a full load of twenty-four softballs.

1.2 Preliminary Design

During the preliminary phase the team split into their respective groups to begin research on designing the best plane using the basic configuration determined during the conceptual phase. Teams created alternative designs or evaluated different equipment and compared the results.

1.2.1 Alternative Designs

Aerodynamics Team:

Evaluation of different planforms and airfoils were performed for the fuselage and wings using various computer programs.

Power Plant Team:

The power plant team obtained various propellers and other support equipment to begin performance tests. An Astro Flight-60 motor was the only motor used for testing. It was mounted to a static thrust bench build by the construction team. Through testing, data was obtained for current, thrust, and RPM. Furthermore, battery performance was evaluated at the same time.

Construction Team:

Preliminary work involved material testing. Tests included making of various components of different materials including carbon fiber, fiberglass, metal, and different types of wood. This gave the construction team an idea of what to use and what not to use when constructing components. The information gained during this period was invaluable when evaluating the FOM for the Manufacturing Plan.

1.2.2 Design Tools

Airfoil and fuselage designs were created in a potential flow program called XFOIL, a FORTRAN 77 based computer program which allows the user to input x and y coordinates to create a 2-D airfoil. XFOIL breaks the shape into a number of short segments and uses higher order panel methods to calculate basic aerodynamic coefficients. XFOIL considers boundary layer formation and can predict separation. It allows the user to modify the plotted pressure distribution and XFOIL redraws the resulting airfoil. Once the wing and body airfoils were optimized in XFOIL the 2-D points were exported to VSAERO. VSAERO is a computer program used to calculate the aerodynamic characteristics of a mesh object in 3D. It requires Cartesian coordinates and some basic geometry of the body. Then a grid or mesh is placed

over the body to define the surface. Calculations produced by VSAERO are sent to an output file where the results are summarized. Omni3D reproduces this data visually in three-dimensional form. From the defined wake location, VSAERO calculates the linearized potential flow external to the body. The user specifies the normal velocity for all parts of the boundary, which represents the body. Wake surfaces have a zero normal velocity and are positioned by VSAERO to be parallel to the local flow. Flow is created by the positioning of singularities (sources and doublets) on the boundary of the object.

Power plant team placed an Astro Flight power meter in the electric circuit for the motor. Also in line was a 40-amp fuse, as specified by the rules. Testing was performed to record current for a given thrust, also battery power consumption (amp-hours) was observed. Testing was performed on various propellers to determine which one would perform the best. The team looked for the highest thrust for the lowest current.

1.2.3 Results of the Preliminary Phase

A final design was established by the end of the preliminary phase. An airfoil was taken from a known source and improved upon. A fuselage was designed to be a low drag lifting body. The power plant team found the best propeller for the motor.

1.3 Detailed Design

End of the preliminary phase marked the beginning of the detailed design phase. All the design specifications were handed over to the drawing team to make working drawings for the construction team to follow. Also, performance data was calculated to determine the operating parameters for the design. All components were weighed or its weight was estimated. Along with the weight, an approximate location was used to ensure that the center of gravity would be maintained at the ¼ chord position.

1.3.1 Alternative Designs

The only alternative design left was the choice between a tail wheel and a nose wheel. This choice is more of a personal preference of the pilot. At the time of the report it was still being decided since either wheel placement will work with the final configuration. Both may be attempted and its performance evaluated.

1.3.2 Design Tools

Using the designs from VSAERO, the drawing team used Pro-Engineer to create parts, assemblies, and drawings; Pro-Engineer is a computer aided drafting program. Cross-section drawings were made in order to print out templates for foam cutting and shaping of the body. A plotter was used for the oversized template drawings. An Excel worksheet was created to keep track of the weights and distances from the center of gravity ensuring a properly balanced airplane.

1.3.3 Results

Completing the detailed design phase marked the beginning of the construction phase. Final design of the airplane is expected to be very competitive. It is a high lift/low drag aircraft able to carry a full load. The power plant gives the plane the ability to perform the task with power and battery capacity to spare. Final aircraft design returned a Rated Aircraft Cost of 13.83 (based on estimated weight).

2. Management Summary

Rules were released for the 2001/2002 Cessna/ONR Design/Build/Fly Competition in June of 2001. A meeting was called immediately for all participants in last year's competition to discuss the organization of the new team. When the fall semester started, the team members advertised the project to other engineering students in order to gain new members. Team composition included one faculty advisor, seven seniors, six juniors, one sophomore, and five freshmen, all aerospace engineering students.

2.1 Architecture of the Design Team

Team member assignments began with Tim Lo, AIAA Student Branch President, appointing a project manager. The project manager organized the initial meetings in order to assess the skills available for the project. Under the direction of the project manager, leaders were chosen for the various teams. Members were then assigned to teams based on each member's wishes and expertise of each individual. Most team members were involved with more than one team, which improved the effectiveness of communication between the teams. Teams are listed below in Table 2-1.

Project Manager: Greg Marien

Treasurer: Thao Tran

Fundraising: Greg Marien, Amado Aviles, Namal De Silva

Team	Aerodynamics	Drawings	Power Plant	Construction	Report
Team Leader	Amado Aviles	Greg Marien	Greg Marien	Andy Bechtel	Thao Tran
Team Members	Ryan Call Leo Rios-Reyes Chad Berman Greg Marien Andy Bechtel Underclassmen	Amado Aviles Thane Coxon	Matt Gregory	Greg Marien Tim Lo Thane Coxon Amado Aviles Matt Gregory Toru Yamasaki	Greg Marien Amado Aviles Thane Coxon

Table 2-1 Team member organization

2.2 Duties and Responsibilities of the Teams

Defining responsibilities is important to ensure a smooth design process. All positions are defined to allow the team to work in an efficient manner.

2.2.1 Project Manager

Regular meetings are called by the project manager and held with the team leaders to ensure that the schedule is maintained and to centralize the decision-making process. Fundraising, presentations, and sponsorship are also the project manger's responsibility. Any spending of funds must be approved through the project manager.

2.2.2 Treasurer

The treasurer maintains an accounting of all funds, including keeping track of all donors and donations. All purchasing must go through the treasurer for reimbursement.

2.2.3 Fundraising

Fundraising team seeks monetary donations and equipment to fund the project. The team also supports the project manager with any presentations in relation to the fundraising.

2.2.4 Team Leaders

Team leaders assign duties to individual team members and organize the team's activities and reports to the project manager.

2.2.5 Aerodynamics

Team is responsible for designing the fuselage and wing and performs aerodynamic analysis of the design.

2.2.6 Drawings

Drawing team converts the aerodynamic team's design into CAD to create working drawings. The team is also responsible for the final drawing package for the report.

2.2.7 Power Plant

Power plant team researches the equipment, tests the motors, propellers, and batteries. Wiring of the plane is also the responsibility of the team.

2.2.8 Construction

Construction team fabricates the plane from the working drawings.

2.2.9 Report

Report team compiles the information and is responsible for the format and printing of the final report.

2.3 Schedule Control

To ensure the schedule is maintained, regular meetings are conducted to solve any problems, which could put the project behind schedule. Project manager can make modifications to the schedule to account for unforeseen circumstances. Milestone schedule is documented in Table 2-2 below.

2.4 Configuration Control

Any modifications to the original design are discussed with all team leaders before the change is made. This is to ensure the design change will not affect the overall performance of the airplane. Project manager makes the final decision of the configuration.

Milestone Schedule for Monty's Revenge																																												
	Jul					Aug				Sep				Oct					Nov				Dec					Jan				Feb				Mar				Apr				
Milestone (week of)	2	9	16	23	30	6	13	20	27	3	10	17	24	1	8	15	22	29	5	12	19	26	3	10	17	24	31	7	14	21	28	4	11	18	25	4	11	18	25	1	8	15	22	29
Design Phases																																												
Conceptual (scheduled)																																												
Conceptual (actual)																																												
Preliminary (scheduled)																																												
Preliminary (actual)																																												
Detail (scheduled)																																												
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report due																																												
Other																																												
entry form due																																												
entry form submitted																																												
competition																																												

Table 2-2 Milestone schedule

3. Conceptual Design

The conceptual design phase is the first phase of the design process. It is the phase in which all concepts and configurations are created and investigated. Concepts are evaluated by different parameters using figures of merit (FOM) to determine which concept has the best qualities. Included in the figures of merit is the Rated Aircraft Cost (RAC), which determines one of the major factors in the competition scoring. Once the basic configurations with the lowest RAC's are found, further evaluation is performed to determine the final conceptual design.

3.1 Figures of Merit with Assumptions and Justifications

Using figures of merit to simplify the selection process, some assumptions had to be made. One assumption made is that the final configuration is a monoplane. Reason for the monoplane is that it will be much easier to construct. Along with the monoplane, a choice had to be made on either a high wing or a low wing design. The obvious choice was a low wing since rapid loading and unloading of the softballs is preferred through the top and a high wing structure would be in the way. Another assumption is that the power plant is assumed to be a tractor, not a pusher. The reason for this is that the large propellers that are used would require an oversized landing gear to prevent a pusher from hitting the runway during the aircraft's rotation. Furthermore, the center of gravity would be moved too far back to be compensated for. These two configurations are therefore removed from any further consideration. Another assumption is that there are twenty-four softballs and five pounds of batteries. Twenty-four softballs will be needed to maximize the score and five pounds of batteries is the limit for the competition. Manufacturer's Empty Weight assumption is thirteen pounds; this is based on seven pounds of propulsion system and five pounds for fuselage, wings, and tail. Other assumptions needed to calculate an approximate RAC are as follows:

Wing span	9 ft.
Chord length	.67 ft.
Number of wing control surfaces	2
Long and skinny fuselage (max length)	80"
Short and wide fuselage (max length)	60"

All vertical empennages have control surfaces.

Using these assumptions, an evaluation was performed to narrow the selections for the airplane configuration. The screening process concentrated on finding the lowest possible RAC and therefore maximizing overall score. Configuration possibilities included the following: number of motors, fuselage type, and stabilizer configuration. "Number of Motors" category was given a 1 or a 2. "Fuselage Type" was either long/skinny (LS) or short/wide (SW); this coincides with possible softball configurations and resulted in a different maximum length for the RAC. "Stabilizer Configuration" had four options: one or two vertical surfaces (with control), a V-tail, or no tail. Table 3-1 below shows the results of sixteen different configurations. Evaluation continued with the four lowest Rated Aircraft Costs and is documented in section 3.3

concept number	Number of Motors	Fuselage Type	Stabilizer Configuration	RAC	Decision
1	1	LS	1	12.90	[illegible]
2	1	LS	2	13.10	[illegible]
3	1	LS	V	12.80	[illegible]
4	1	LS	NT	12.50	P
5	1	SW	1	12.57	P
6	1	SW	2	12.77	[illegible]
7	1	SW	V	12.47	P
8	1	SW	NT	12.17	P
9	2	LS	1	14.98	[illegible]
10	2	LS	2	15.18	[illegible]
11	2	LS	V	15.08	[illegible]
12	2	LS	NT	14.58	[illegible]
13	2	SW	1	14.64	[illegible]
14	2	SW	2	14.84	[illegible]
15	2	SW	V	14.54	[illegible]
16	2	SW	NT	14.24	[illegible]

Table 3-1 Rated Aircraft Cost of Various Concepts

3.2 Alternative Configurations

Concepts 4, 5, 7, and 8 received the lowest RAC. All four had one motor; one had a long slender body while the others had the short and wide body. Various tail styles passed and will be discussed in section 3.3. Winning concepts are pictured in Figure 3-1.

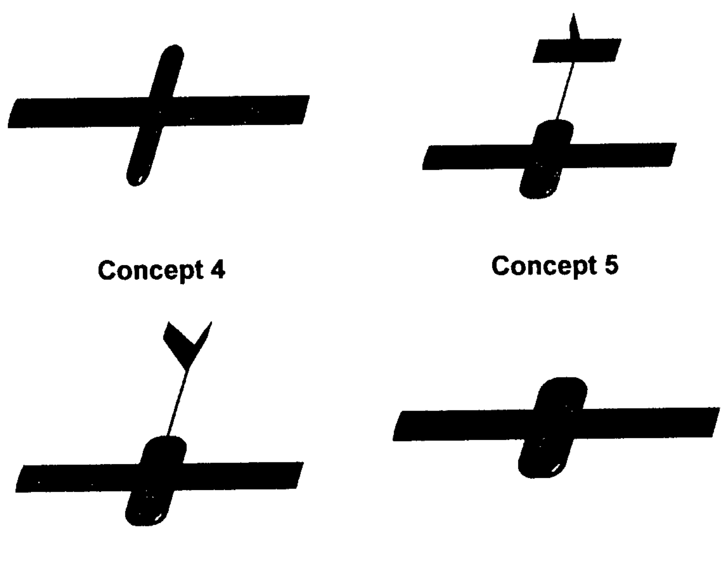

Figure 3-1 Winning Concepts

3.3 Design Parameters Investigated

Four concepts had to be narrowed further until the final configuration could be chosen. In order to choose the final configuration the team held a meeting to discuss the results. The decisions were as follows:

Power Plant:

Only single engine configurations survived the screening process. This was because of the higher Rated Aircraft Cost for multiple engines; therefore, no further evaluation was needed. Multiple propellers were discussed, but the lack of experience in building a gearing or pulley system for that type of configuration prevented any further evaluation.

Fuselage:

Two different fuselages were discussed. One was the long slender fuselage and the other was short and wide. The long slender fuselage looked like a typical passenger airplane, which may have been competitive, but the team decided to go with something different - the short and wide body. Attempting to make the short and wide fuselage led to the decision to create a lifting body. A flying wing was also possible, but the team wanted to avoid the instability issues.

Stabilizer Configuration:

Three types of tails passed the screening process. The tail that returned the lowest RAC is no tail at all. Stability is a big issue in airplane design, and the lack of a tail removes too much stability and control. Because of lack of stability the "no tail" configuration was dropped from consideration. The next best possible choice would have been the V-tail. With the guidance of the construction team and the pilot, this concept was dropped. The reason for this is that the team had no experience with the V-tail and there was only a slight advantage in RAC. The final decision was to use a single vertical/horizontal stabilizer (T-tail).

3.4 Final Configuration Selection

- Monoplane
- Low wing
- Short and wide fuselage
- Lifting body fuselage
- Single engine at the front of the plane
- Twenty-four softball load
- Single vertical and single horizontal stabilizer (T-tail)

4. Preliminary Design

During the preliminary phase, the aerodynamics team determined the sizing and design parameters for final aircraft design. Parameters from the conceptual phase were used as the foundation to be designed around. Instead of using figures of merit, parameters were evaluated separately for the best configuration of each. Also, wing and power loading requirements were used to determine the final design specifications. Assumptions had to be made and justified to complete the design. Assumptions were then compared to their respective conceptual phase values.

4.1 Design Parameters (Figures of Merit)

Design parameters evaluated include the payload configuration (ball tray), wing and fuselage design, power plant, and tail sizing.

4.1.1 Ball Tray

Competition requirements restricted the softball configuration to a single level, at least two abreast, and no overlapping. Efficient placement of the softballs was crucial to the lifting body design. Maximum loading of twenty-four softballs was the basis for the conceptual phase and is part of the preliminary design.

Two design ideas were used for ball placement, each with twenty-four softballs. Since a sphere has the best surface to volume ratio, an attempt was made to closely simulate the cross section of a sphere using the balls to fill it. Best placement resulted in a circular 2-4-6-6-4-2 configuration as shown in Figure 4-1(a) below. Although this would have given a good use of surface to volume ratio, this design would have required a very wide body, which is undesirable due to the increased drag. Another reason is the rule for blended wing configuration requires the balls to be within 18", and the team wanted to avoid a possible rule violation. The next best design is shown in Figure 4-1(b). With the final configuration of 5-7-7-5, an airfoil-shaped fuselage could now be designed around the softballs.

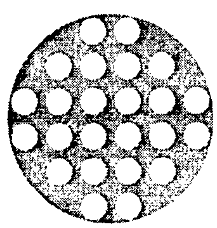

Figure 4-1(a) Round Body (2-4-6-6-4-2) Configuration

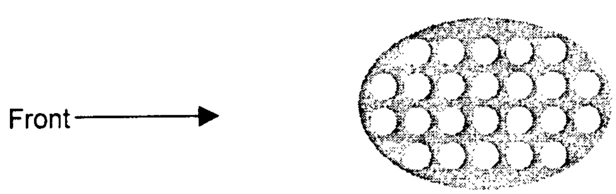

Figure 4-1(b) Elliptical Body (5-7-7-5) Configuration

4.1.2 Wings and Fuselage

Using the ball tray in Figure 4-1(b), a lifting body was designed around it. Although the shape would give lift, the goal was to reduce drag.

Assumptions were made to create preliminary designs for the wings. Total weight of the plane along with expected speed and calculated Reynolds numbers were assumed. The aerodynamics team researched many different airfoils. An airfoil from a model airplane book caught the attention of the team. Taking the basic airfoil shape, the team improved on it giving the airplane a distinct advantage compared to the original airfoil.

4.1.2.1 Assumptions

Preliminary tests from the Cobalt 60 Astro Flight electric motor with a 22X10 propeller indicated an average of 11 lb. of thrust. Limited to a 200 ft. runway the minimum velocity at take off was estimated to be 50 ft/s. The weight of the aircraft last year was adjusted for this year's payload, giving a net take off weight estimated to be 22 lb. Assuming the aircraft will be constructed using the same material as last year, this estimation is expected to be fairly accurate. At sea level conditions the Reynolds number was calculated to be 212,000. Finally, prop wash was neglected during the design phase. With these assumptions, the first course of action was to design a two-dimensional airfoil that would support a relatively slow flying heavy plane.

4.1.2.2 Choosing an Airfoil

Main Wing:

After balancing drag and lift characteristics, a high camber airfoil was found, designated the G8 (Grant 26). Data obtained from XFOIL for the G8 is shown below in Figure 4-2. Improvement began with an effort to smooth out the pressure recovery on the top surface. Reducing the pressure gradient toward the trailing edge helped to keep the flow laminar. This is desirable as it reduces drag and tends to delay stall. XFOIL was then utilized to modify the pressure distribution.

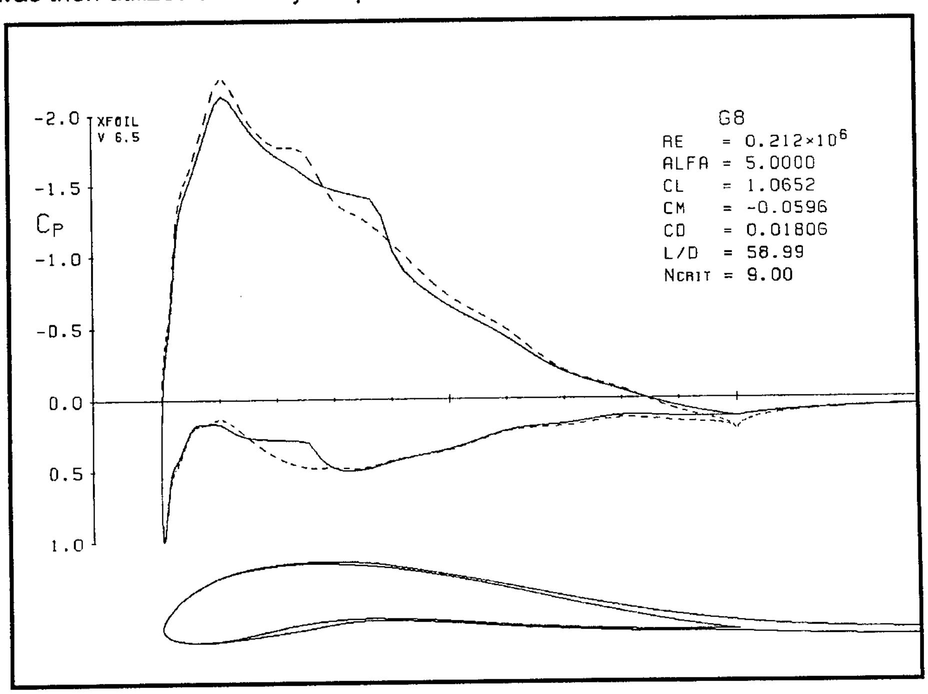

Figure 4-2 Pressure Distribution of an Unmodified G8 Airfoil

Modifications were optimized for a 5-degree angle of attack. It turns out that 5 degrees is the cruise angle for the wing; the reasoning for this is discussed later. Leading edge of the G8 was rounded so the flow over the top surface would satisfy the above conditions. Some of the camber of the G8 was removed to reduce the separation on the lower surface. Just past the leading edge the curve on the lower surface was smoothed out to reduce the separation bubble as well. During modifications constant attention was given to C_L, C_D and L/D values. Changes to the airfoil were only allowed if they improved these numbers, but sacrifices were allowed when there was a trade off. The main objective for the modification of the G8 was to increase the lift coefficient. From the modifications, L/D increased by 16% to 70.2. The new airfoil was designated the G8-AR and its pressure distribution is shown in Figure 4-3.

Along with the G8-AR airfoil, a NACA 0009 airfoil will be incorporated into the wing tip. Due to extensive testing preformed by NACA, the airfoil results were not reproduced.

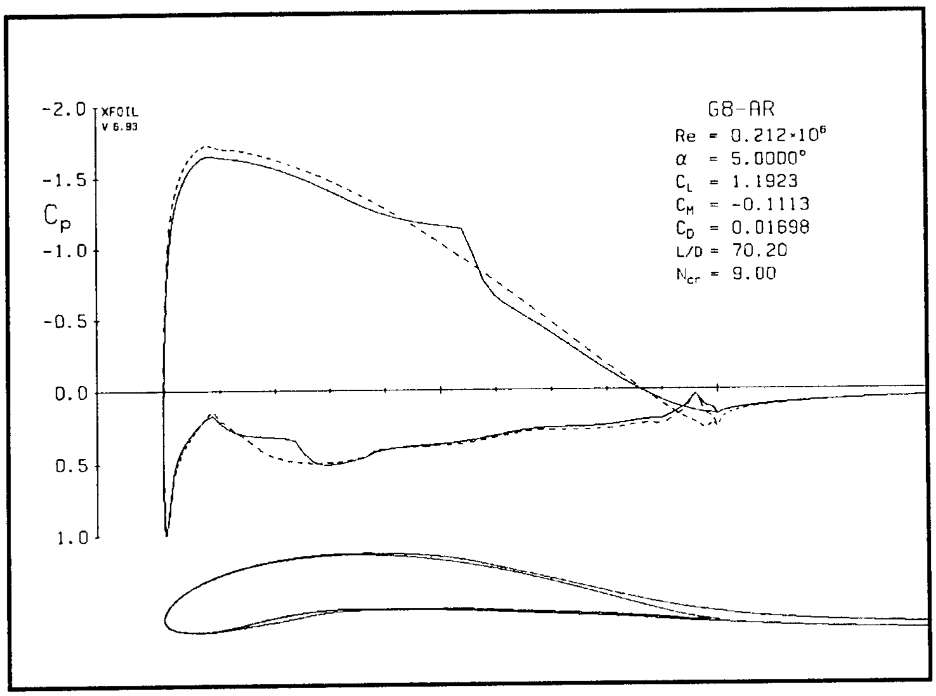

Figure 4-3 Modified G8 airfoil-designated the G8-AR

Lifting Body Fuselage:

After some debate in the conceptual phase, the DBF team decided to incorporate a lifting body design to improve flow around a large payload area. An airfoil body would reduce drag and produce some lift compared to an otherwise blunt body. Arrangement of the payload and engine structure had been decided in the early part of the preliminary phase, so the best solution was to build up from the minimum area of the payload. A profile of seven softballs was entered into XFOIL. A 3.82" high (softball diameter) by 28" long cross-section (approximate length of seven softballs) was non-dimensionalized as thickness and chord respectively as illustrated in Figure 4-4. By modifying the pressure distribution at a 3-degree angle of attack, a C_L of 0.864 was achieved; Figure 4-5 is the improved airfoil shape for the body and is designated the DBF2002-AR airfoil.

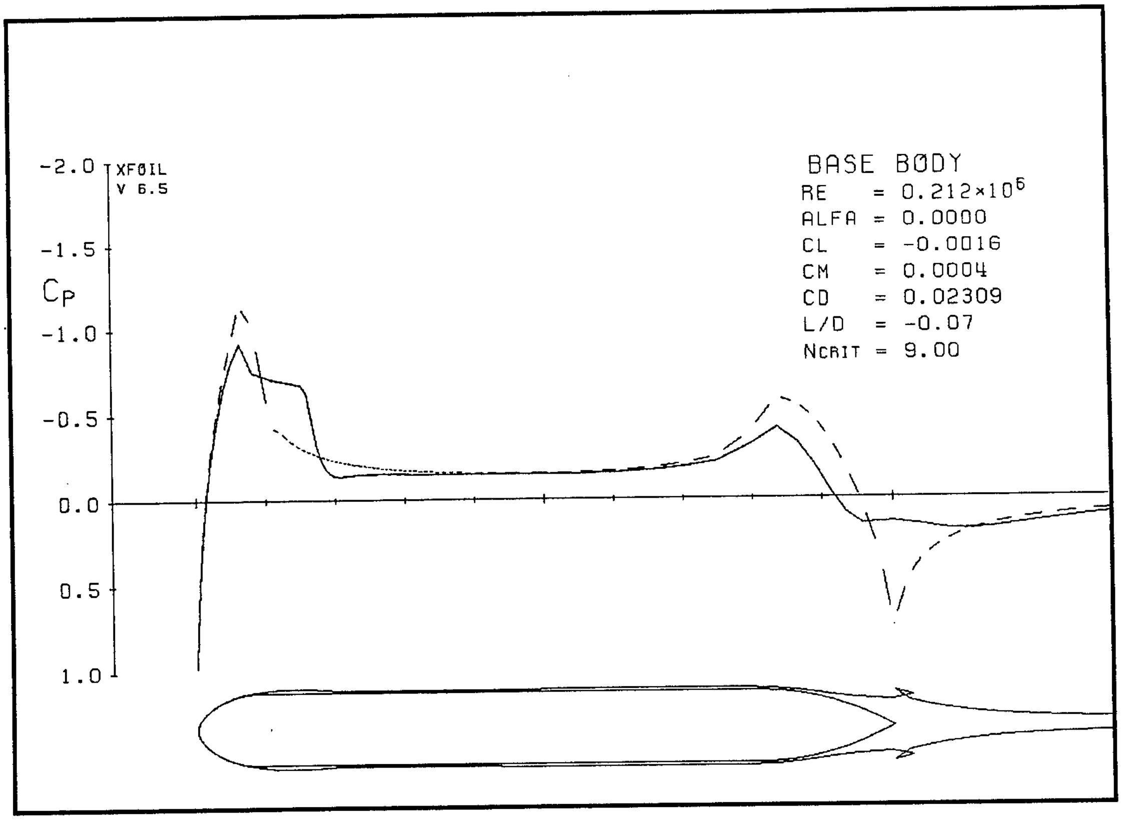

Figure 4-4 Preliminary Fuselage Cross Section

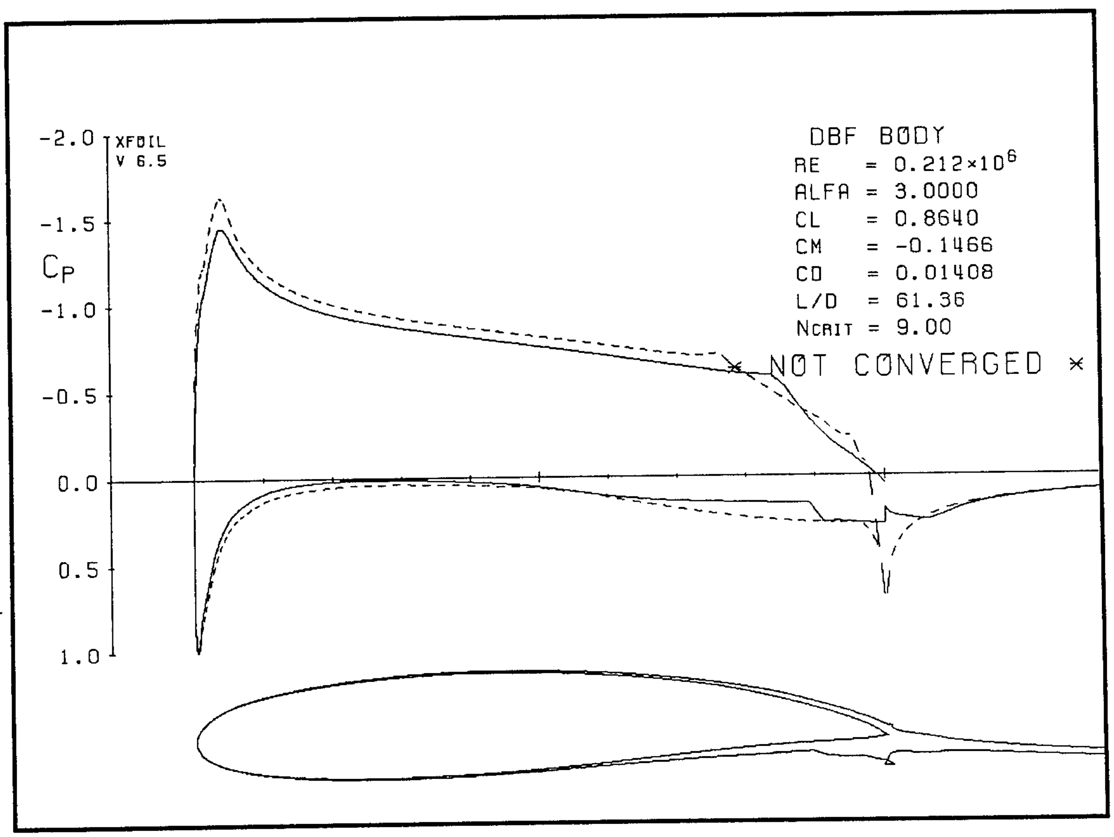

Figure 4-5 Modified Fuselage Cross Section

4.1.2.3 Lift and Drag Coefficient Results

XFOIL was used to obtain the lift and drag coefficients over a range of angles for both the wing and lifting body. Parameters investigated were drag versus lift and lift versus angle of attack

Main Wing:

From Figure 4-6 the minimum drag point was found to be 0.01781 with the corresponding C_L point at 1.0541. Corresponding value for the angle of attack was found to be 5 degrees, see Figure 4-7. In order the keep the induced drag to a minimum the incidence angle for the tips was set to an angle of attack of zero; therefore, giving the tip of the wing a downward twist reducing the possibility of tip stall.

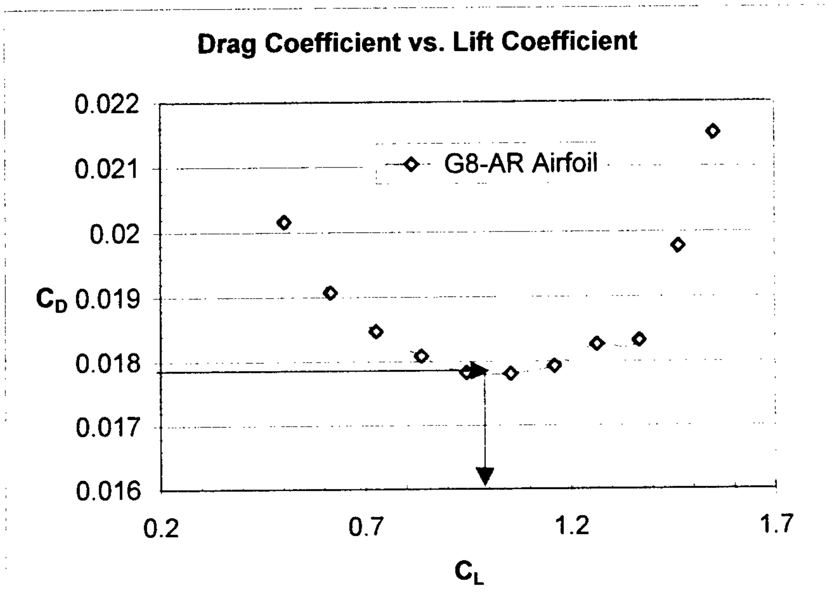

Figure 4-6 G8-AR Airfoil Drag Coefficient versus Lift Coefficient

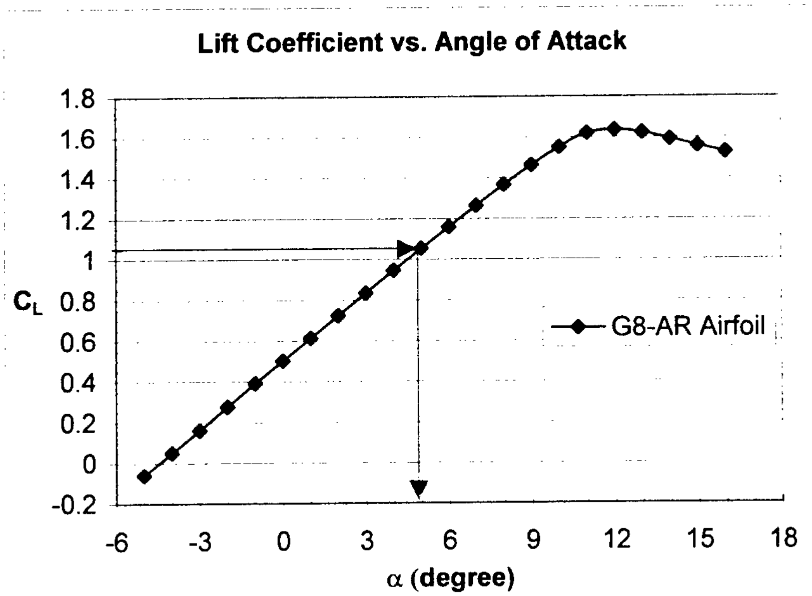

Figure 4-7 G8-AR Airfoil Lift Coefficient versus Angle of Attack

Lifting Body Fuselage:

Figure 4-8 shows the lowest drag point, which occurs at 0.01238, and the corresponding C_L value is 0.6414. This C_L value is referred to as C_L minimum. After obtaining $C_{L\ min}$, Figure 4-9 was used to return a corresponding angle of attack, which is approximately 5 degrees. This in turn is the angle at which the airfoil will have the lowest drag; therefore, the lifting body design will have an angle of attack of 5 degrees.

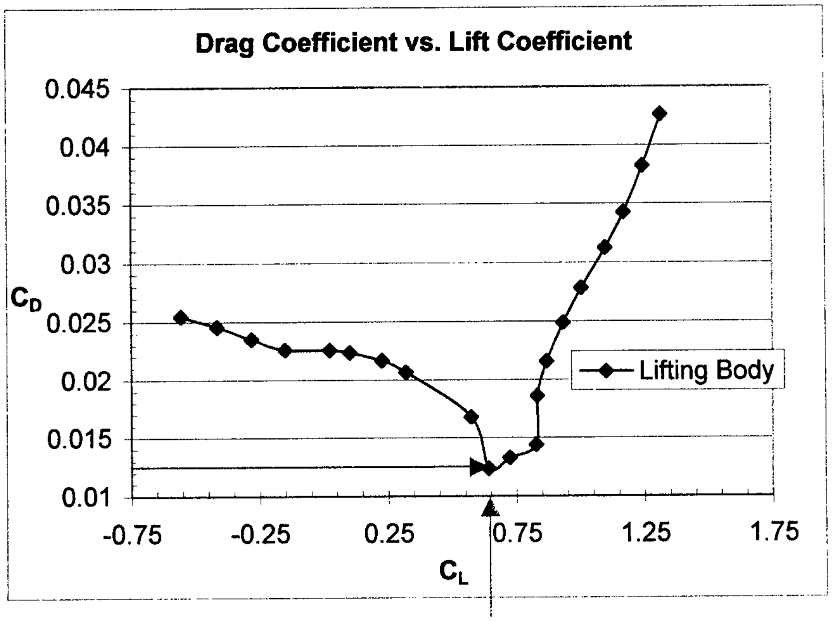

Figure 4-8 DBF2002-AR Drag versus Lift Coefficient

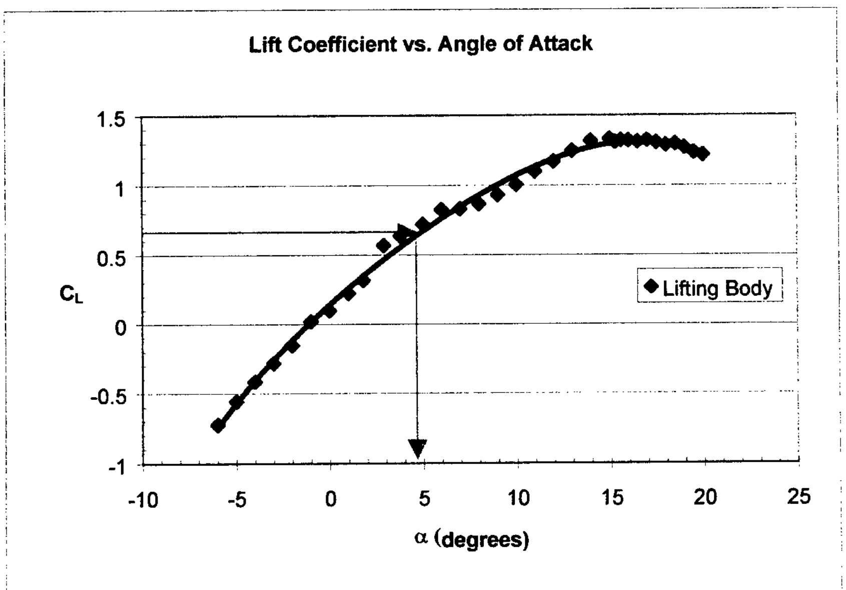

Figure 4-9 DBF2002-AR Lift Coefficient versus Angle of Attack

4.1.2.4 Optimizing the Wing Shape

After determining the incidence angle of the wing and lifting body airfoils, a preliminary model of the wing was built in VSAERO where body and wing components were blended together. Initial modification began with tapering the wings to reduce induced drag and approximate an elliptical shape. Taper was limited to the last foot near the tips, thereby maximizing the lifting surface. Sweptback wings were not used since this would reduce the angle at which the wing passed through the free steam and therefore reduce lift.

Wing Span and Aspect Ratio:

After creating the aircraft in VSAERO, a test was performed to determine an Aspect Ratio that would produce a lift of at least 22 lb (fully loaded aircraft). Planform area was estimated to be approximately eight square feet and the wing span was varied from 4 feet to 12 feet and the corresponding values of the lift coefficient were obtained. Results are shown in Figure 4-10. Minimum lift coefficient needed to lift 22 lb. was calculated to be 0.9582. Using Figure 4-10 the corresponding aspect ratio was found to be 10.31, which in turn gives the wing span of approximately 9 feet with the main chord of 0.6667 feet.

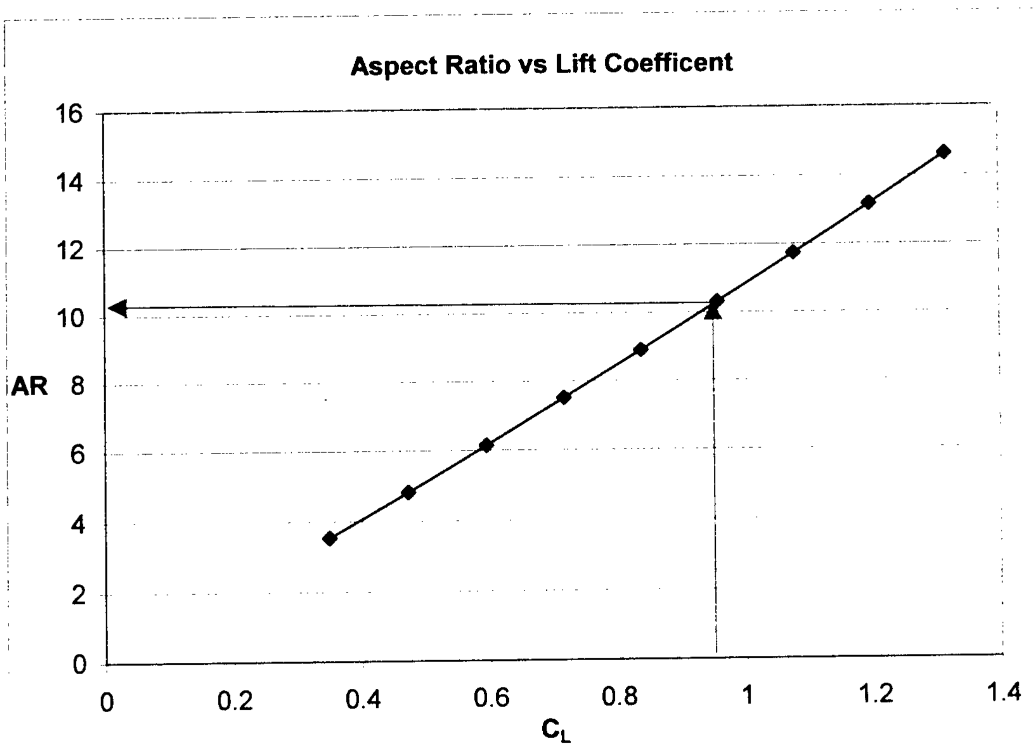

Figure 4-10 Aspect Ratio versus Lift Coefficient for Main Wings

Dihedral Angle and Taper Ratio:

The two design aspects that were varied were dihedral angle and taper ratio (for the last twelve inches of the wing). A minimum dihedral angle of 3 degrees was calculated to raise the center of pressure of the wing just above the center of gravity, which is needed to stabilize the aircraft. To aid in choosing two possible dihedral angles for evaluation, a test was run in VSAERO varying the dihedral angle from 0 degrees to 10 degrees. For each run the values of the lift coefficient were obtained, see Figure 4-11. It is known that as the dihedral increases the C_L decreases, due to the decrease in planform area. It is interesting to note that the data obtained from VSAERO for the wing is not linear and that a zero angle of dihedral is not the point of maximum C_L. The reason for this was determined to be from wing and body interaction. From the data obtained, the maximum C_L of 0.959 gives a dihedral angle of approximately 2.5 degrees. Although a higher C_L exists at 2.5-degrees evaluation was performed from 3 to 5 degrees. This was because a minimum of 3 degrees was required for the design. Since the object was to increase stability and not lift, the difference of 0.001 in the C_L value did not matter.

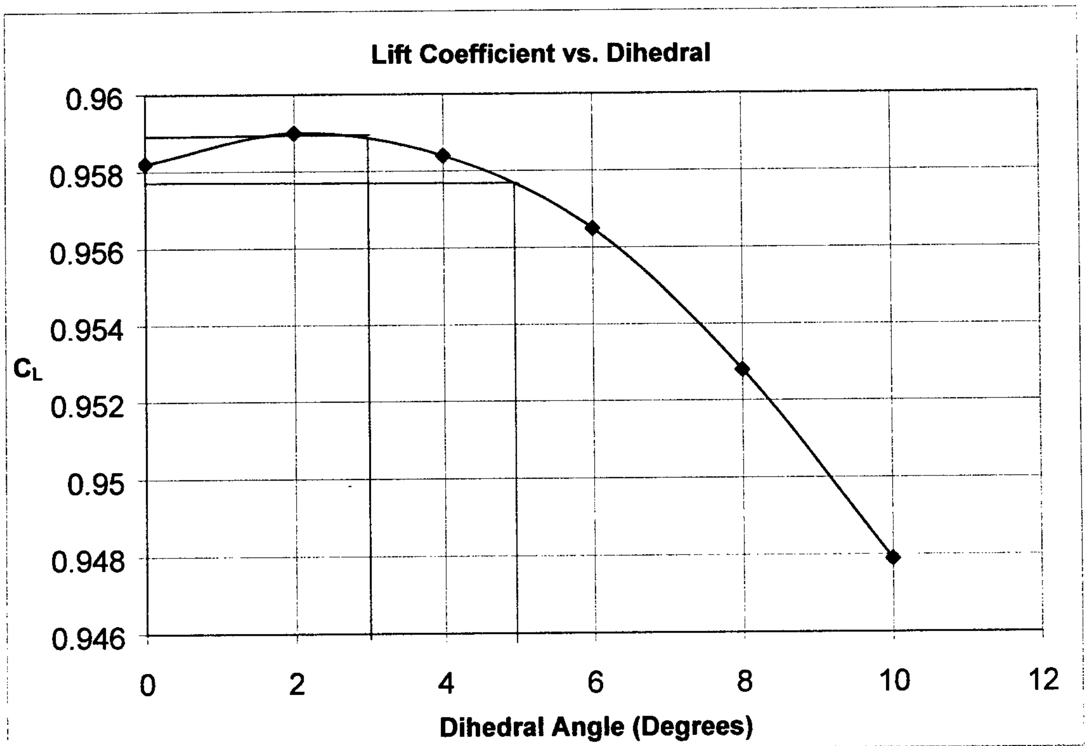

Figure 4-11 VSAERO Data for Effects of Lift Coefficient versus Dihedral Angle

Taper ratio was also varied to help increase the efficiency of the wing. Lift and efficiency of a tapered wing is 10% to 15% greater than a uniform chord and equal area (Grant 32). Grant suggests that a taper ratio of 0.5 is useful for model aircraft; therefore, taper ratios of 0.5 and 0.25 were chosen for the test models.

With the optional configurations, four models were designed. All of the models have the same wingspan, lifting body configuration, wing configuration, and wing placement with respect to the body. Incidence angles were maintained the same for all tests. Dihedral angles of 3 and 5 degrees and taper ratios of 0.5 and 0.25 had been chosen resulting in the following four models:

Model D1T1: Dihedral 3 degrees, Taper Ratio 0.5

Model D2T1: Dihedral 5 degrees, Taper Ratio 0.5

Model D1T2: Dihedral 3 degrees, Taper Ratio 0.25

Model D2T2: Dihedral 5 degrees, Taper Ratio 0.25

These four models were created and run in VSAERO. Data was collected and compared to determine the best design for the mission.

4.1.2.5 Results

Comparison of the four models, with respect to span wise loading, showed little variation. Using Figure 4-12, the following general observations can be noted. Load across the wing is distributed higher at the roots dropping toward the tips. The general distribution resembles the desired elliptical shape. A significant amount of the work is performed at the wing root and wing-body blend where C_L values peak at 1.464. The fuselage bears a small yet significant amount of the loading. A C_L of 0.4645 was still achieved at the lowest point on the curve. An interesting anomaly occurred at the wing tips. Excessive taper had increased loading at the ends. This is highly undesirable since it counteracts the extensive efforts to reduce induced drag.

A similar comparison could be made across the wing with respect to the span wise moment distribution. See Figure 4-13 below. As expected the largest forces are found near the root of the wing. These forces tend to bend the wing up around the fuselage. Large forces are allowed since the moment arm is small and localized to a small region. Once again, excessive taper created unfavorable conditions at the tips. Moments from the body create unwanted stresses and structural problems.

An evaluation of the drag across the wing can be made from Figure 4-14. Again there was very little deviation between models. However the distribution is quite different from the previous comparisons due to the interaction between components and wide variation in surface area. Symmetry again plays an important roll for stability by balancing drag distribution across the span. Taper drastically reduces drag at the tips by .0244 a reduction of 43%. Drag falls linearly across the main wing traveling toward the root from .0429 to .0172. At the wing-body blend the drag drops significantly and is attributed to wing and body interaction. Attention was paid to the trend not the negative values, which have been concluded to be erroneous points. Moving toward the center of the lifting body the drag increases almost elliptically leveling out at 0.0479.

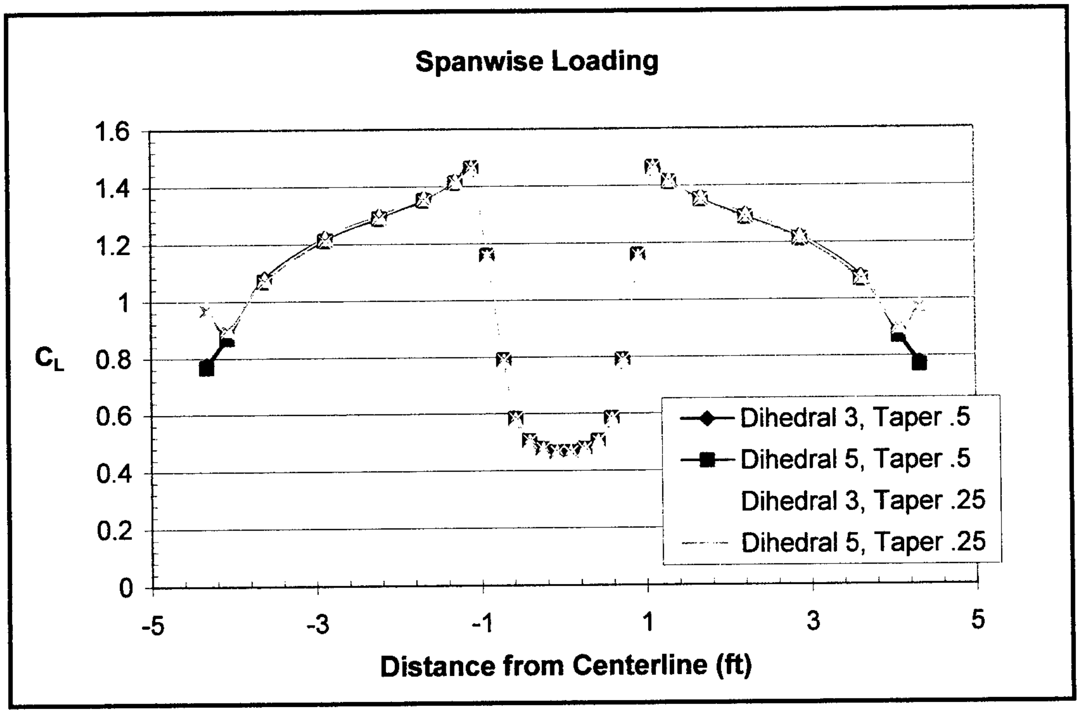

Figure 4-12 Span Wise Loading Distribution of Four Different Models

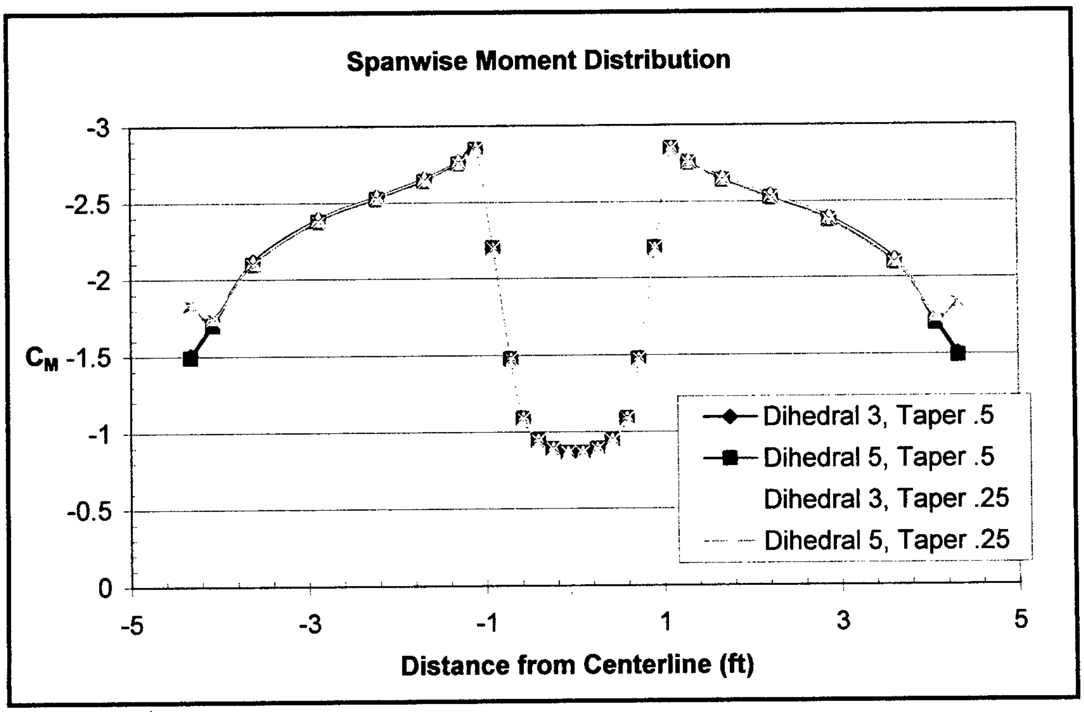

Figure 4-13 Span Wise Moment Distribution of Four Models

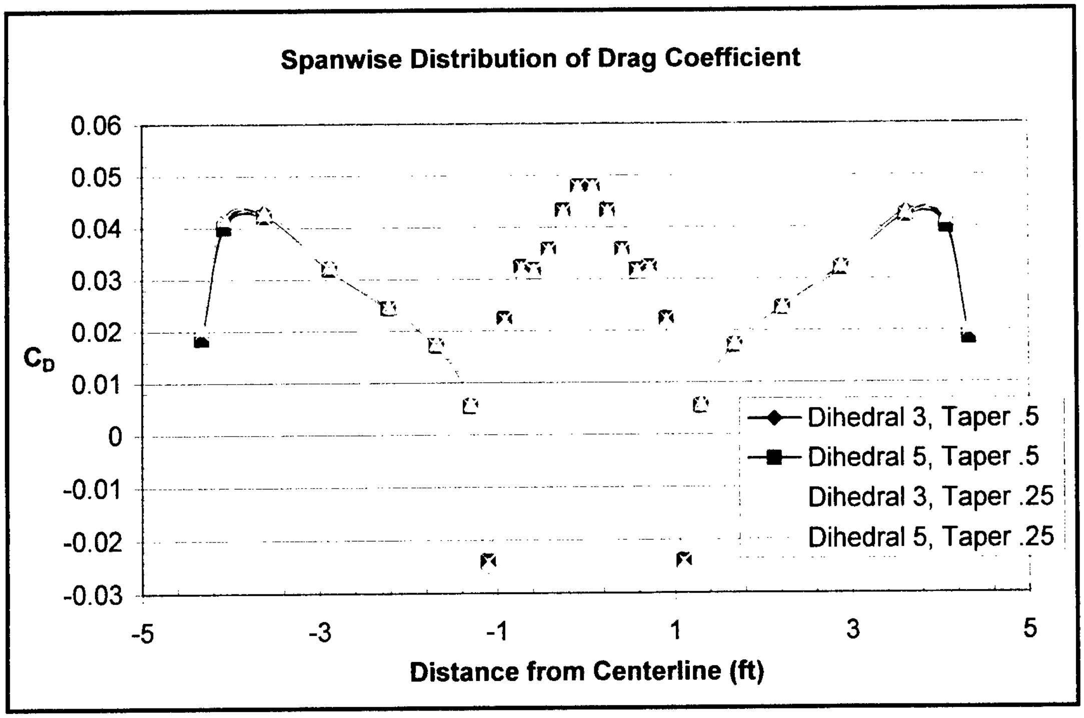

Figure 4-14 Span Wise Drag Coefficient Distribution of Four Models

A result of the aerodynamic characteristics of the different aircraft configurations is compiled in Table 4-1. The data here represents the average forces for the entire wing-body configuration. Model D1T1 was chosen from the four models based on the higher values obtained for the C_L and the smoother load distribution at the tips.

Model	C_L	C_M	C_D
D1T1	0.9587	-1.850	0.0331
D2T1	0.9571	-1.849	0.0330
D1T2	0.9563	-1.844	0.0330
D2T2	0.9548	-1.843	0.0329

Table 4-1 Solution Summary of Various Models

Analysis of the D1T1 design started with making 14 butt-line cuts along the span of one side of the body, assuming that the opposing half will have exactly the same data. Three of the fourteen sectional centers of pressure and the pressure distributions are shown in Figures 4-15 through 4-18. In each of the figures the model and the span wise loading are displayed to help visualize each butt-line cut. It is interesting to note that the centers of pressure for the lifting body are much further aft, with respect to the airfoil than that of the main wing. Figure 4-19 shows the location of the center of pressure at each section along the span, showing an average of all the centers of pressure at approximately 1.3 ft from the nose of the lifting body. Having the center of pressures at the same location along the entire span helps in the balancing of the aircraft.

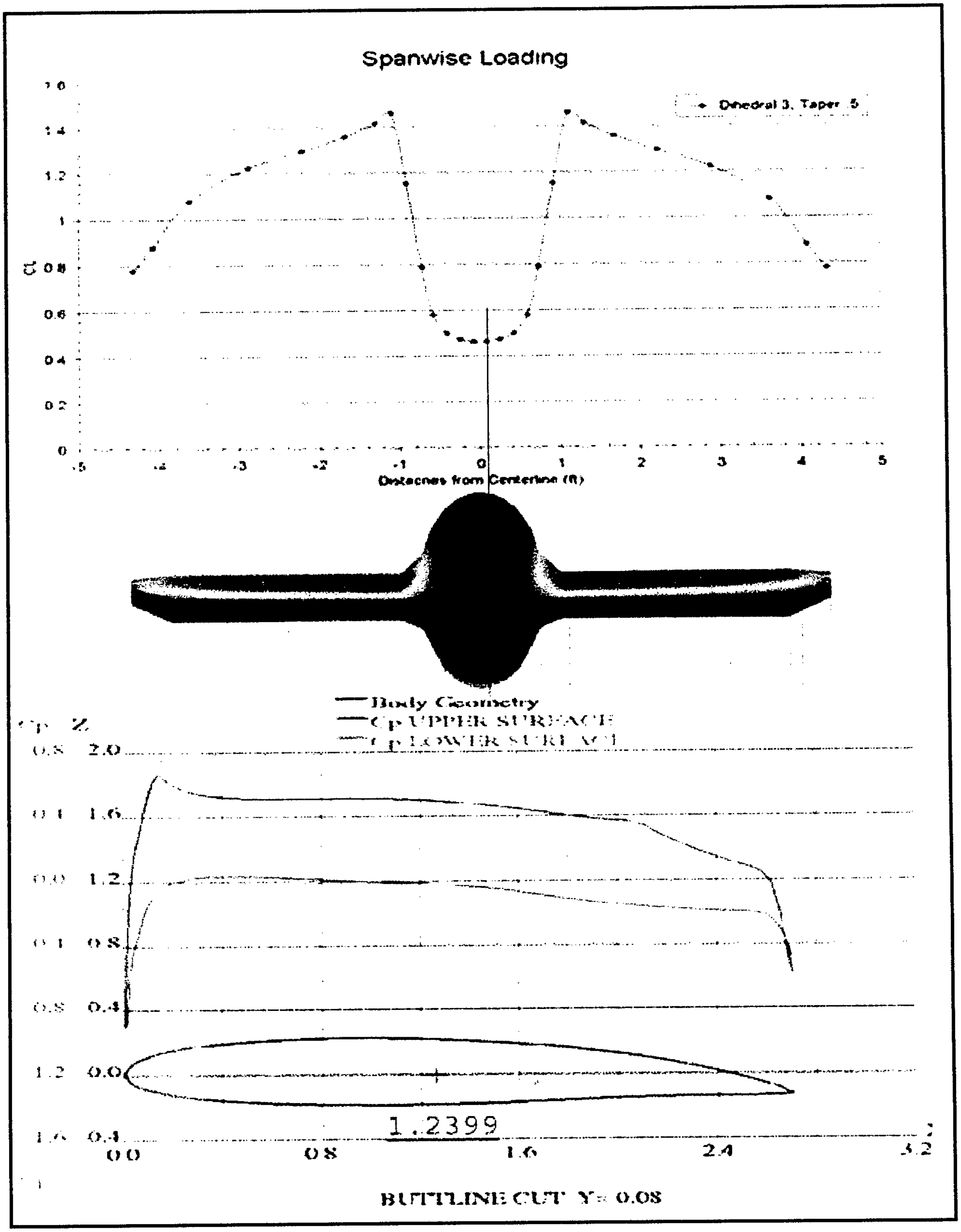

Figure 4-15 Span Wise Loading at 1" from Centerline

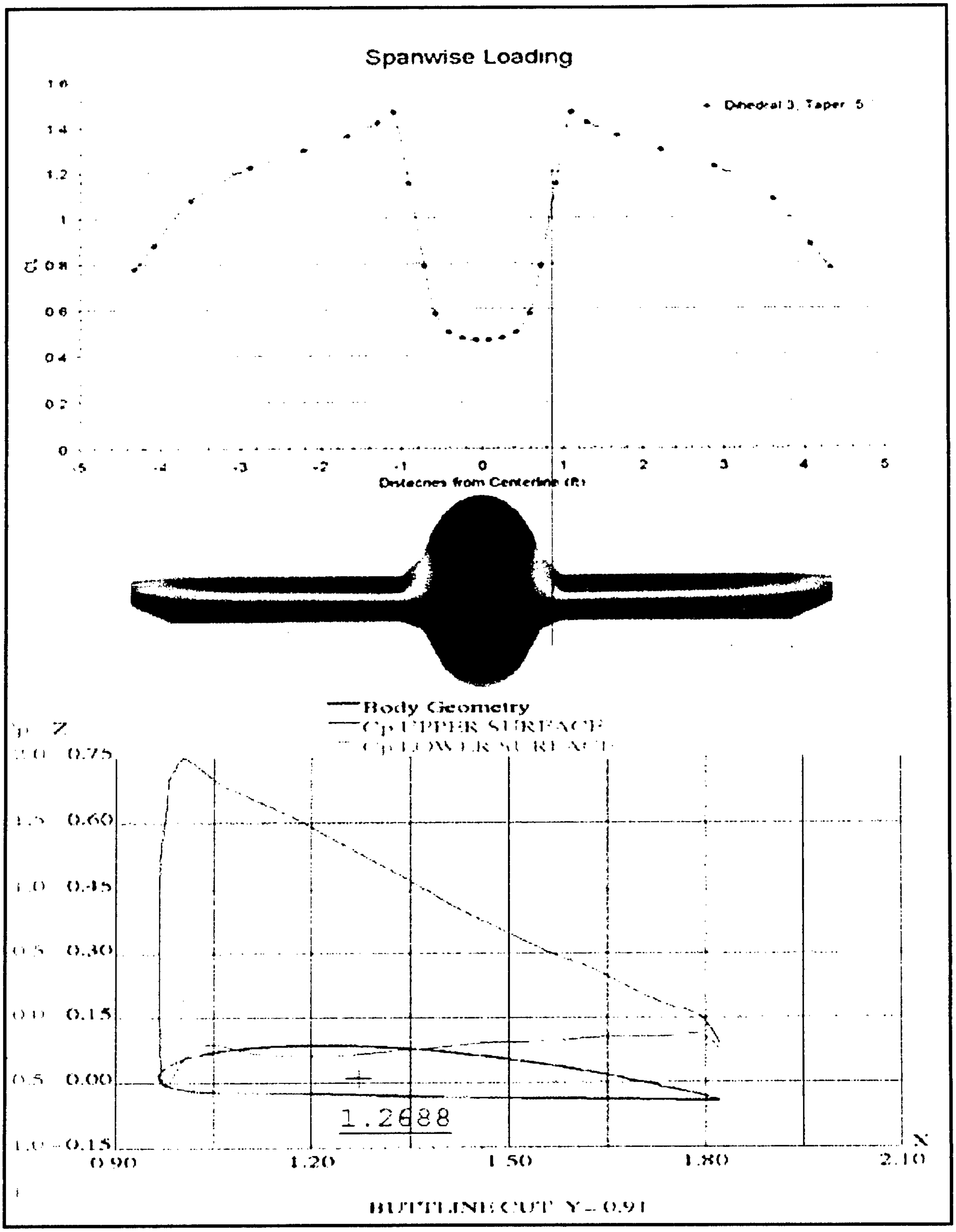

Figure 4-16 Span Wise Loading at 10" from Centerline

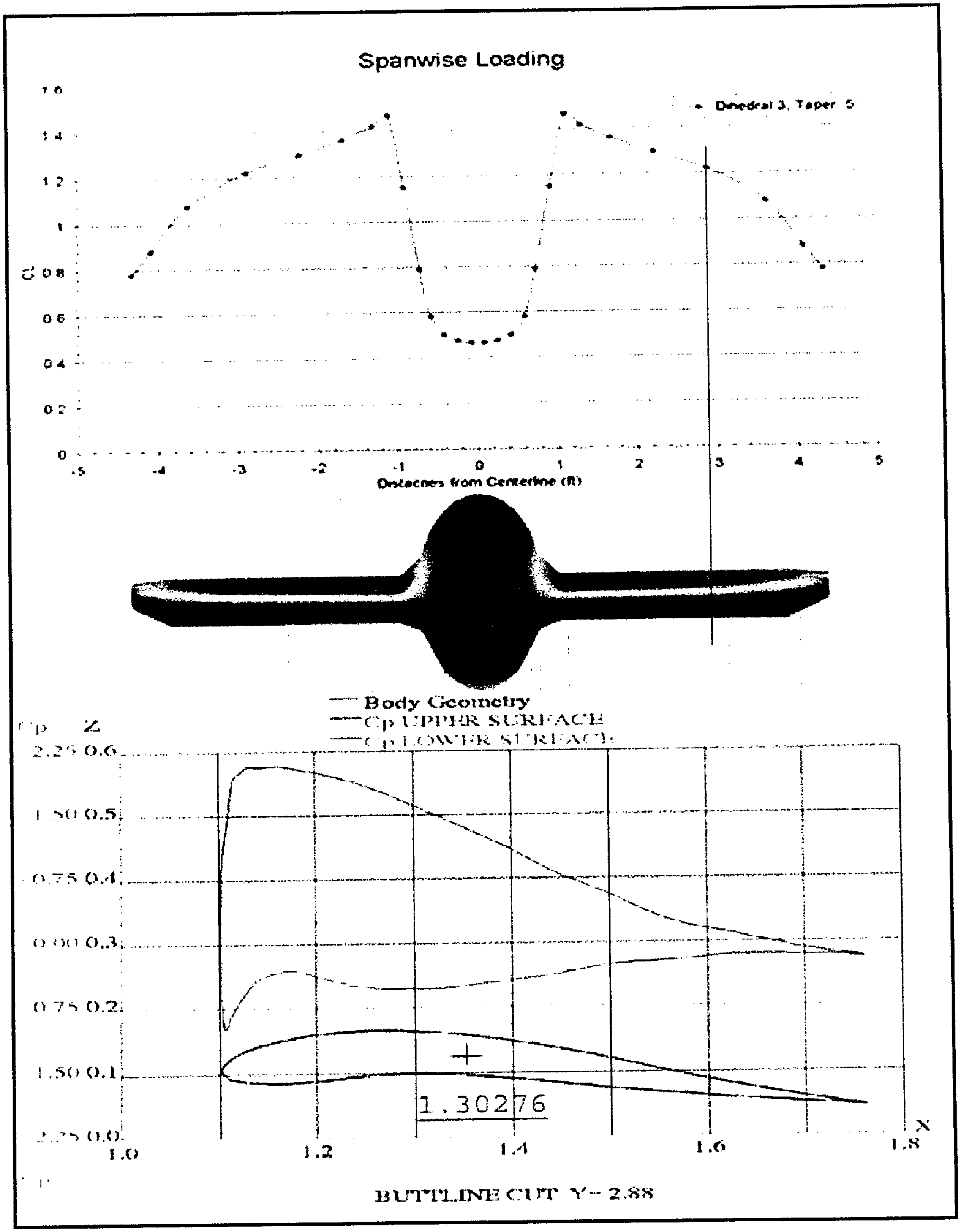

Figure 4-17 Span Wise Loading at 34” from Centerline

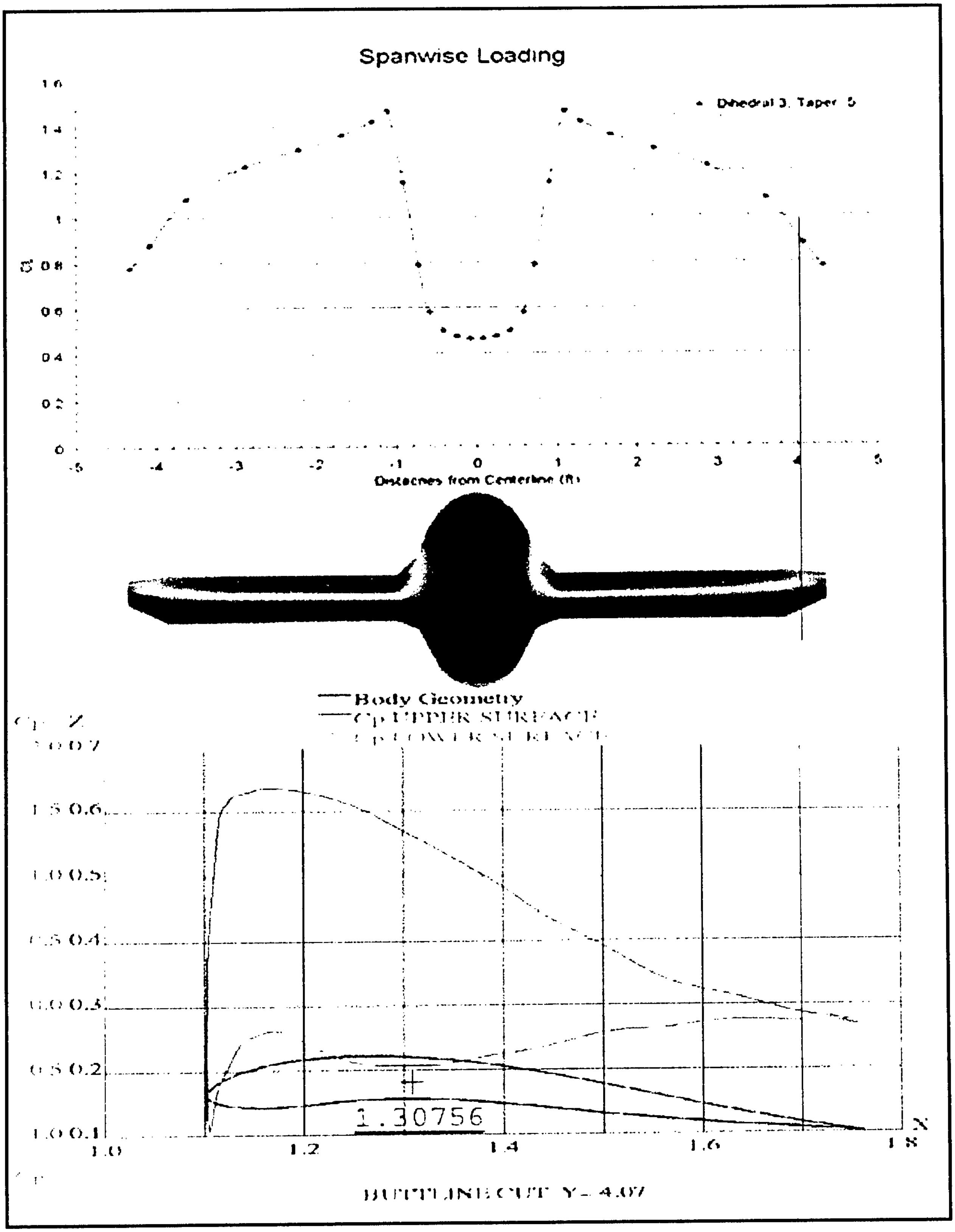

Figure 4-18 Span Wise Loading at 48" from Centerline

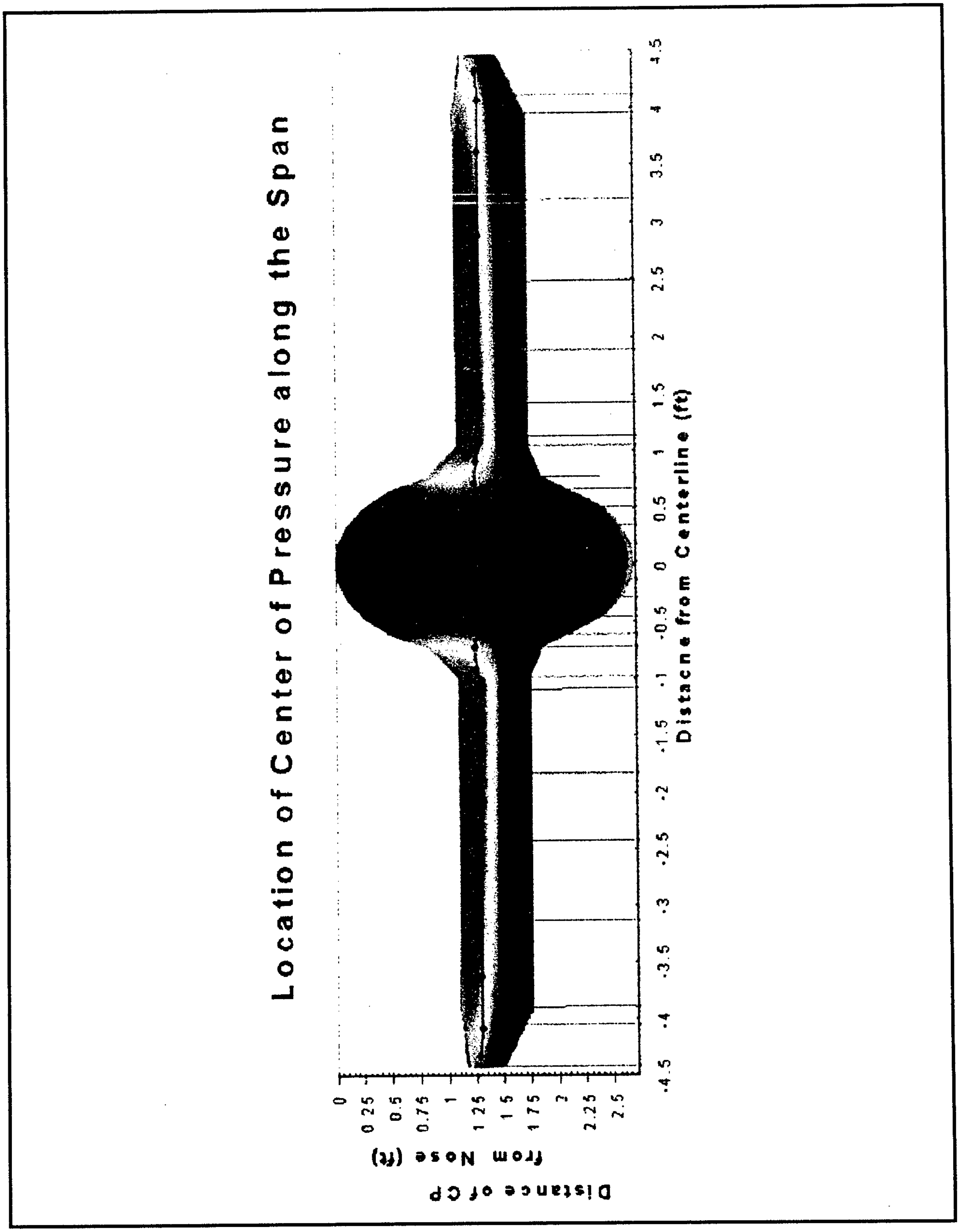

Figure 4-19 Center of Pressure Location Along the Entire Span

4.1.3 Power Plant

Conceptual design required only one motor; therefore, the on-hand AstroFlight-60 motor was used to determine all further data. Due to lack of funding no other motors were evaluated. Various propellers were tested to determine the most efficient configuration. Power source for the motor was three NiCd battery packs, each containing 12 batteries for a total of 36. Batteries are rated for 2400 mah and are connected in series.

Using ElectriCalc, the best size propeller for the AstroFlight-60 motor was evaluated to be a 2-bladed 22x12 Zinger. Using the results from the ElectriCalc program, multiple propellers, small and large, were tested on a static test platform. Propellers tested included the following types: 20x10 and 22x10 Zinger, 22x12 Moki, 24x12 Master Airscrew, and a 3-bladed 16" propeller. Testing included observing the current, thrust, and RPM in order to make comparisons between the different sizes of propellers.

First, the 16" 3-bladed propeller was tested and was immediately disqualified from further testing since the motor was too powerful for it, which makes it too easy to exceed the RPM limits of the motor. Next, the 20x10 Zinger was tested and was also disqualified because of excessive RPM readings. Master Airscrew's 24" propeller resulted in plenty of thrust but the torque was too great as observed in the current draw for a given thrust. Finally, the 22" propellers were tested and resulted in a close decision between the 22x12 Moki and the 22x10 Zinger. Both would allow full throttle, which put the motor at its design limits, and both gave the required amount of thrust. Figure 4-20 below shows one of the four trials of propeller comparison and is representative of the other tests performed. The Zinger used less current than the Moki for a given thrust - 4 to 6 amps less. Since the ElectriCalc program recommended a 22X12 Zinger, an attempt was made to locate one for comparison, but none were found. It was decided to use the 22x10 Zinger until a 22x12 could be purchased and tested.

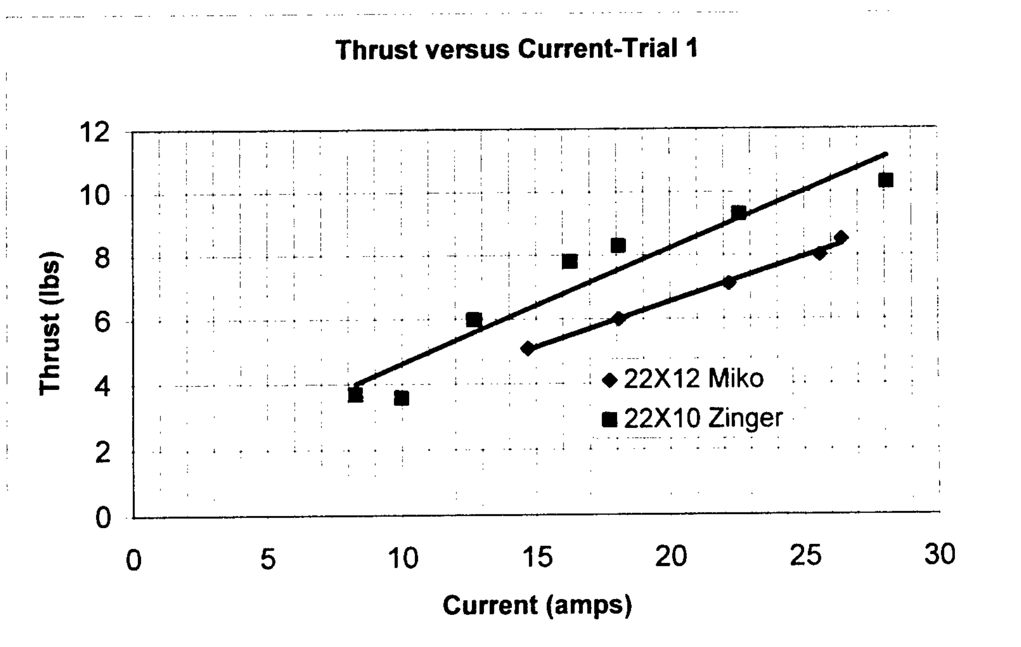

Figure 4-20 Thrust versus Current: Comparison Between Different Propellers

4.1.4 Tail Sizing

The tail type was determined to be a single vertical and single horizontal stabilizer (T-tail). Sizing for the tail was calculated using two different sources, Nicolai and Grant, and compared.

4.1.4.1 Horizontal Tail

Preliminary sizing of the horizontal tail was done using horizontal tail volume. According to Nicolai, the horizontal tail volume can range from 0.3 to 0.5; a value of 0.4 was used for the preliminary sizing (Nicolai 12-1). Using a wing surface wing area of 927 in^2, a mean aerodynamic chord of 9.4", and an estimated distance of 35.2" between the mean aerodynamic centers of the wing and tail, the horizontal tail area was calculated to be 96 in^2. This value seemed to be small when compared with the wing. However, from Grant, it was suggested that the area of the horizontal tail should be about 30% to 45% of the wing area. Grant indicated that for a good glide ratio, the ratio between the horizontal tail and wing area should be 35%. Since the design was to have a good glide ratio 35% was used. Calculation of the second method returned a value of 325 in^2 which was used in the final configuration.

4.1.4.2 Vertical Tail

Nicolai states that vertical tail volume should range between 0.05 and 0.08; therefore, using a vertical volume of 0.06 initial size of the vertical tail was calculated (Nicolai 12-1). Using a value of 108" for the span and chord and the distance between the mean aerodynamic centers of the wing and tail, the fin area was computed to be 169 in^2. Grant specifies a range of 12% to 25% of the fin to wing area ratio. It also indicated that best glide performance was obtained at 18% for the model configuration. Using 18%, the fin surface area was calculated to be 166 in^2, which was consistent with Nicolai. For the final configuration, a fin surface area of 210 in^2 was used to have a margin of safety.

4.2 Comparison of Assumptions with Conceptual Phase Assumptions

Conceptual design assumed a nine-foot wingspan, which was exactly what the aerodynamics team came up with. A weight of 22 pounds is still the same, and the payload is still twenty-four softballs. Prop wash was neglected in order to simplify gathering of design parameter data. In reality, prop wash will result in a higher Reynolds number and turbulent flow, which may help to prevent flow separation and improve the lifting body characteristics.

5. Detail Design

The Detailed Design phase is the final step before construction can begin. Performance data is evaluated, such as take-off, stability, g-load capability, and predicted mission performance. Detailed drawings and aircraft weight/balance worksheets are also part of the final phase.

5.1 Performance Data

5.1.1 Take-off Performance

Take-off has to be within 200 feet. Calculation of the take-off distance is as follows (Nicolai 10-8):

$W := 22 \cdot lb$

$S := 7.9 \cdot ft^2$

$g = 32.174 \frac{ft}{s^2}$

$\rho := .002379 \frac{lb \cdot sec^2}{ft^4}$

$T := 11 \cdot lb$

$L := 9.78 \cdot lb$

$D := 2 \cdot lb$

$\mu := .03$

$C_{Lmax} := 1.7$

$$S_G = \frac{1.44\left(\frac{W}{S}\right)}{g \cdot \rho \cdot C_{Lmax}\left[\frac{T}{W} - \frac{D}{W} - \mu \cdot \left(1 - \frac{L}{W}\right)\right]}$$

$Sg := 78.4693 ft$

Where:

W = weight

S = wing area

T = available thrust

L = lift at 70% of take-off speed

D = estimated drag

μ = rolling coefficient of friction

Conclusion is that the airplane will be able to meet the requirements of the contest.

5.1.2 Stability

Using the assumption of 50 ft/s, the Phugoid was determined to be 6.9 seconds. Since stability derivatives could not be calculated at the time of this printing, tail volume was increased by about 20% in order to give a wide design margin.

5.1.3 Load Capability

To simulate a 2.5 g-load, the fully loaded aircraft is to be lifted by the wing tips at the competition. To ensure the structure is strong enough to handle the load a prototype of the wing will be made to perform a structural test. Using 30 lb. for the weight of the plane (includes extra for a design margin) and 39.15" for the distance from the tip to the root, a bending moment of 48.9 lb-ft was calculated. Since the root is where the bending moment is estimated to be the highest, the test will be sufficient to determine the structural strength of the wing.

5.1.4 Predicted Mission Performance

Endurance was key in completing all three missions in the 10 minutes. Assuming a total of 80 seconds for the two payload changes leaves 8.7 minutes for flight time. Assumptions made for the required travel distance is based on the size of the course. Turns at the ends of the course are estimated to be 100 feet each. Each 360-degree downwind leg turn is estimated to be 200 feet. With 1,000 feet of travel between the turns, total travel for the first two missions is 9,600 feet. The final mission is only 4,400 feet because there is no 360-degree turn. Total travel distance of the three missions is 14,000 feet. With the 8.7 minute time frame the airplane must fly at least 26.8 ft/s (18.3 mph). This does not account for the landing and take off times, but with an estimated speed of almost twice the minimum required, time will not be a problem. Battery performance has been evaluated experimentally which shows the batteries have enough energy to accomplish all three missions.

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5.2 Weights and Balance Sheet

To ensure the airplane is balanced around the center of gravity, components must be placed in strategic locations. All available components were weighed and given an approximate location from the center of gravity. Components not yet constructed or not available at this time were estimated. Results of the weight and balance are documented in Table 5-1. The resulting moment was calculated to be zero; therefore, the airplane is balanced. Many item weights were estimated, but the design of the aircraft allows for moment compensation by moving the battery packs forward or aft.

negative number indicates a moment around the trailing edge			
Component	**weight (lb)**	**distance from center of mass to CG**	**moment around CG**
fuselage	4.00	-1	-4.00
wings	2.00	-1	-2.00
tail	0.75	-42	-31.50
tail boom (50 in)	0.40	-24	-9.60
tail boom mount (fwd)	0.06	2	0.12
tail boom mount (aft)	0.04	-14	-0.56
main gear	0.50	-2	-1.00
main wheels (X2)	0.12	-2	-0.24
main wheel axles (X2)	0.08	-2	-0.16
nose gear	0.25	14	3.50
nose wheel	0.06	14	0.84
nose wheel axle	0.04	14	0.56
Electronics			
PCM receiver	0.10	13	1.30
rudder servo	0.08	-42	-3.36
elevator servo	0.08	-42	-3.36
right aileron servo	0.12	0	0.00
left aileron servo	0.12	0	0.00
nose gear servo	0.07	4	0.28
Power Plant			
motor/mount	1.90	17	32.30
battery pack #1	1.65	2	3.30
battery pack #2	1.65	2	3.30
battery pack #3	1.65	2	3.30
speed controller	0.12	13	1.56
fuse/fuse holder	0.08	13	1.04
propeller	0.22	20	4.40
Payload			
24 softballs	9.64	0	0.00
total weight	25.78		
moment summation	0.02		

Table 5-1 Weights and Balance Sheet

5.3 Component Selection and Systems Architecture

Final component design is documented in Table 5-2. Drawings of the final configuration are included at the end of this section.

Fuselage airfoil	DBF2002-AR
Empty weight	16.14 lb.
Full weight	25.78 lb.
Payload (# of softballs)	24
Landing gear type	undecided
Main Wing Airfoil	G8-AR
Wing span	9 ft.
Wing:root chord	0.66 ft.
Wing: tip chord	0.42 ft.
Wing area	7.9 ft.
Tail area (horizontal)	210 in^2
Tail Area (vertical)	325 in^2
Motor	Astro Flight-60
Batteries (# of cells)	36
Battery capacity (each)	2400 mah
propeller	22X10

Table 5-2 Final Design Configuration Summary

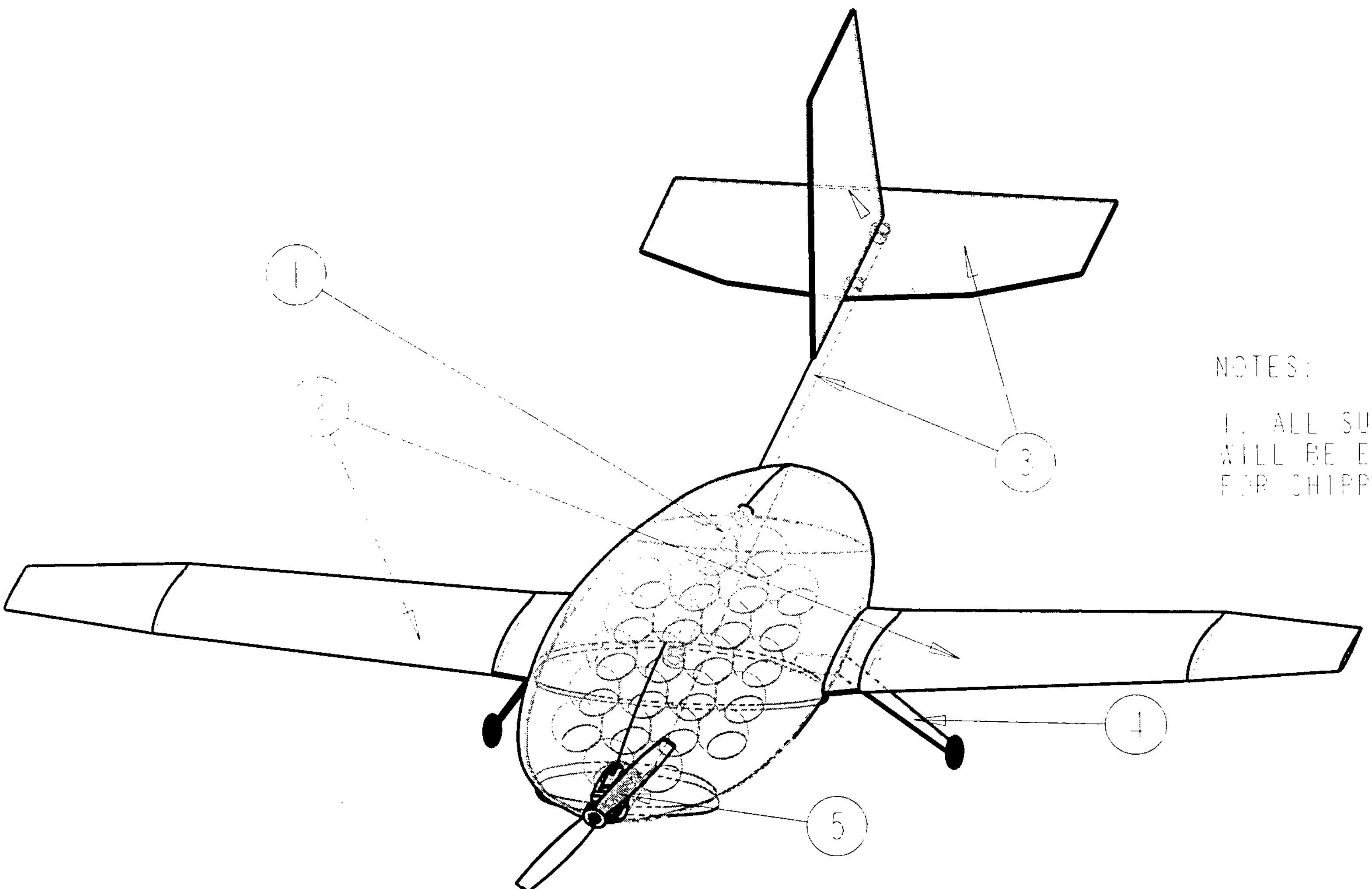
5.4 Rated Aircraft Cost

	Work Breakdown Structure (WBS)			**multiplier**	**WBS subtotal**
1.0 Wing	Wing span	9	ft	8 hr/ft	72
	Chord length	0.67	ft	8 hr/ft (max chord)	5.36
	Number of control surfaces	2		3 hr/control surface	6
2.0 Fuselage	Body maximum length	6.00	ft	10 hr/ft max length	60
3.0 Empenage	Number of vertical surfaces	0	(no control)	5 hr/vertical surface	0
	Number of vertical surfaces	1	(with control)	10 hr/vertical surface	10
	Number of horizontal surfaces	1	**note 1**	10 hr/horizontal surface	10
	V-tail		**note 2**		0
4.0 Flight System	Number of servos	5	**note 3**	5 hr/servo	25
5.0 Propulsion	Number of engines	1		5 hr/engine	5
	Number of propellers	1		5 hr/propeller	5
	Battery weight (lb)	4.88			

Coefficient			
A	Manufactures Empty Weight	100	$
B	Rated Engine Power Multiplier	1500	$
C	Man. Cost Multiplier	20	$/hr
MEW	Manufacturers.Empty Weight	26	lb
REP	Rated Engine Power	4.88	lb
MFHR	Manufacturing Man Hours	198.36	hr

Rated Aircraft Cost

13.89	$ (Thousands)



NOTES:

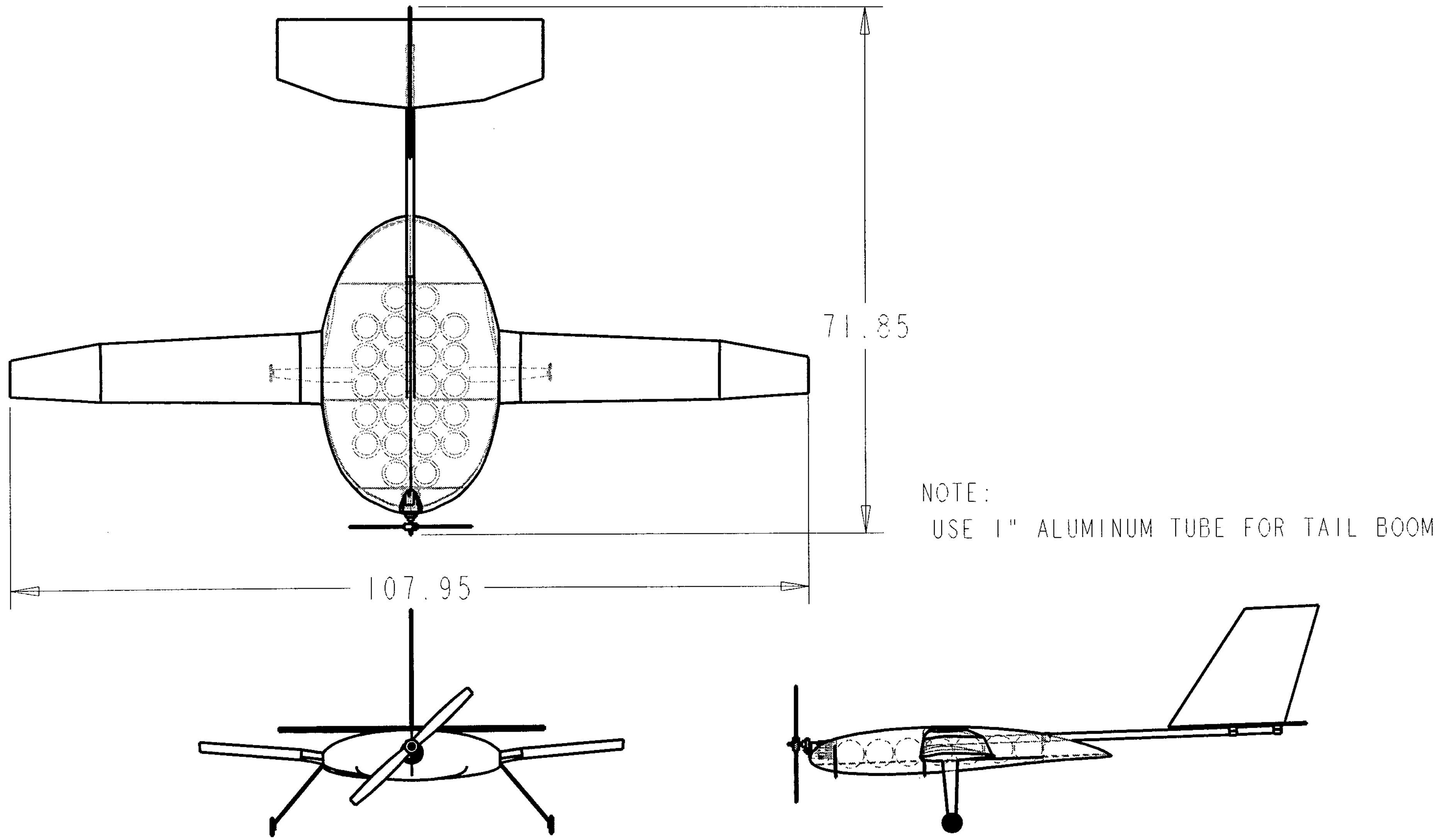
1. ALL SUBASSEMBLIES WILL BE EASILY REMOVABLE FOR SHIPPING PURPOSES

ITEM	DESCRIPTION	QTY
5	MOTOR/PROPELLER	1
4	LANDING GEAR	2
3	TAIL ASSEMBLY	1
2	MAIN WING	2
1	FUSELAGE	1

SDSU AEROSPACE ENGINEERING

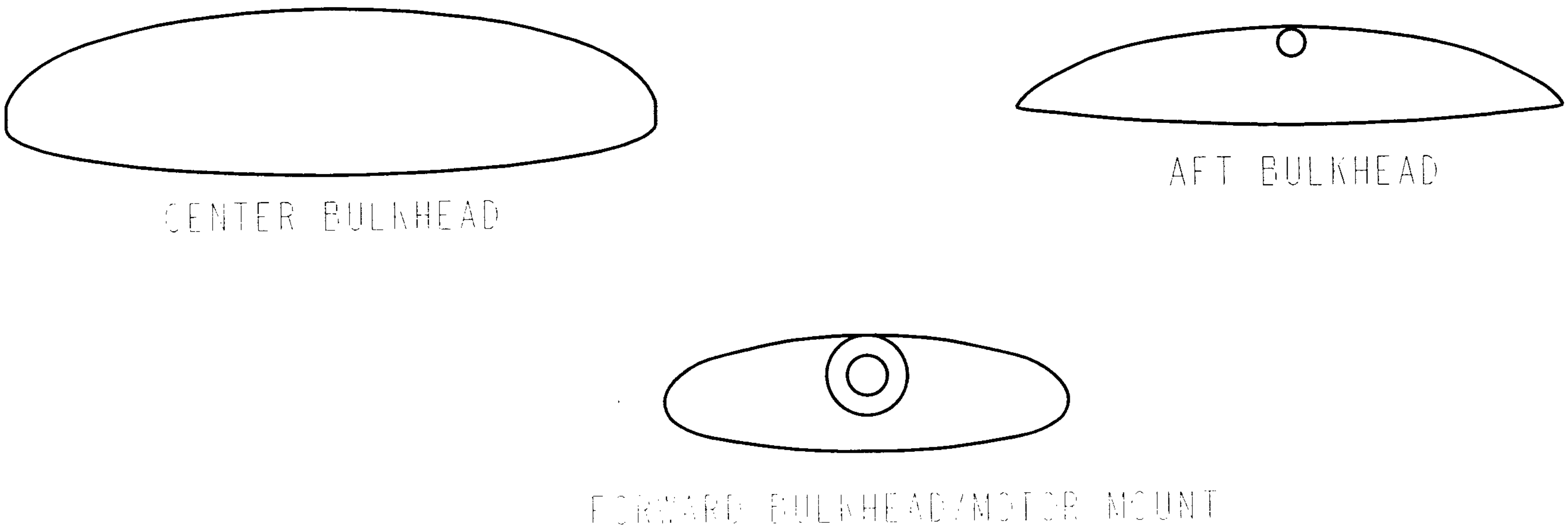
FULL ASSEMBLY

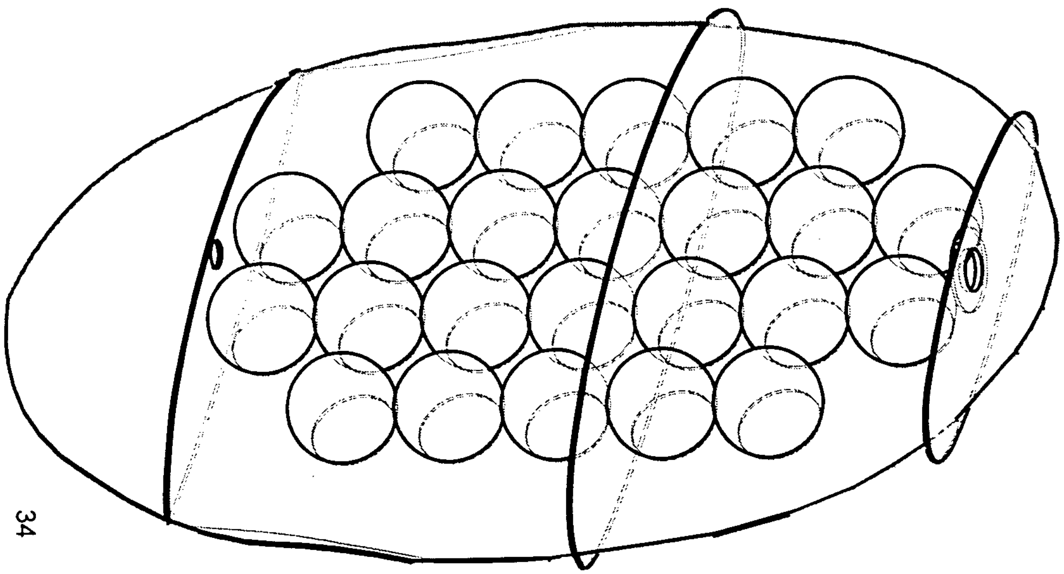
DRAWN BY GREG MARIEN		DATE 1/15/02
SIZE A	DWG NO MONTY'S REVENGE - DBF 2002	
SCALE 0.080		SHEET OF 5



FULL ASSEMBLY		
SIZE A	DWG NO MONTY'S REVENGE - DBF 2002	REV
SCALE 0.050		SHEET 2

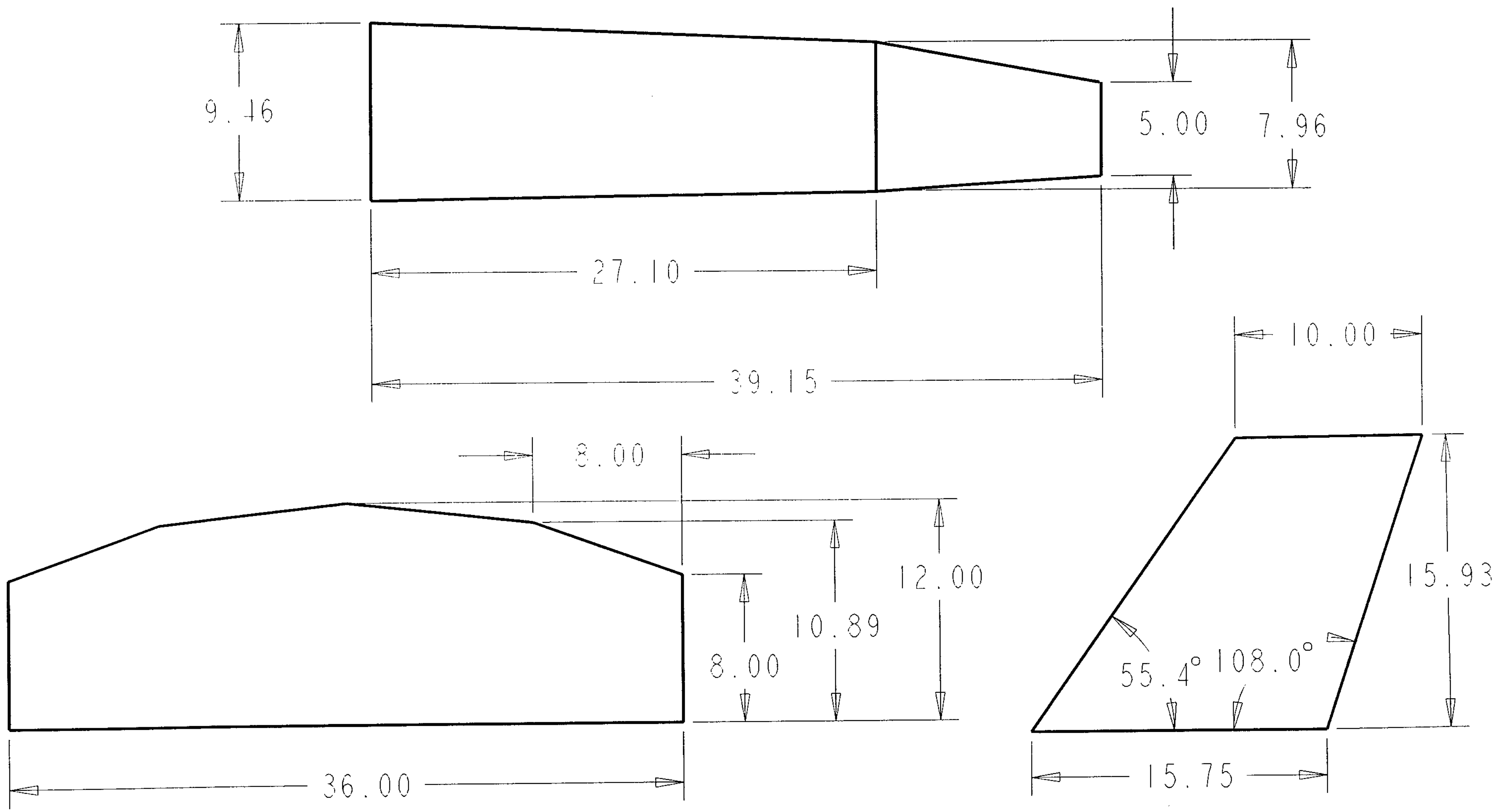
PRO/E





NOTES:
1. BALL TRAY AND BULKHEADS ARE 1/8" BALTIC BIRCH PLYWOOD
2. BULKHEADS ARE GUSSETED WITH BALSA WOOD
3. BALL TRAY CONSISTS OF 24 3" HOLES
4. DRILL HOLES IN BULKHEADS AND BALL TRAY AS NEEDED TO REMOVE WEIGHT

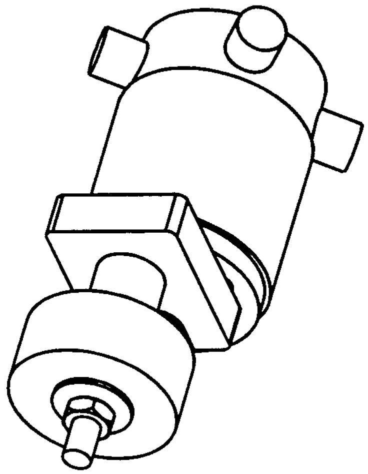
BALL TRAY SUBASSEMBLY			
SIZE A	DWG NO MONTY S REVENGE - DBF 2002		REV
SCALE 0.080		SHEET	3



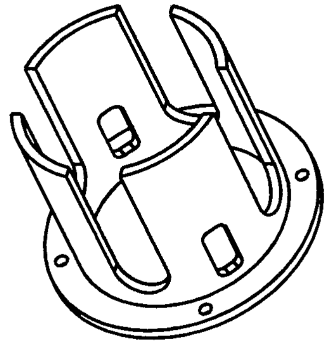
NOTES:
1. MAIN WINGS CONSTRUCTED OF FOAM CORE, 8.9 OZ. CARBON FIBER, AND JEFFCO EPOXY WINGS VACUUM BAGGED
2. BUILT UP TAIL WITH BALSA FRAME AND MINOR USE OF SPRUCE WOOD TO IMPROVE STIFFNESS
3. MAIN WING USE HITEC HS-945MG SERVOS
4. RUDDER AND ELEVATOR USE HS-77BB SERVOS

WING AND TAIL PLANFORMS			
SIZE A	DWG NO MONTY'S REVENGE - DBF 2002		REV
SCALE 0.125		SHEET	4

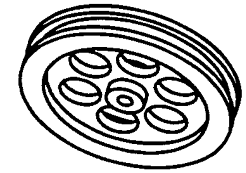
PRO/E



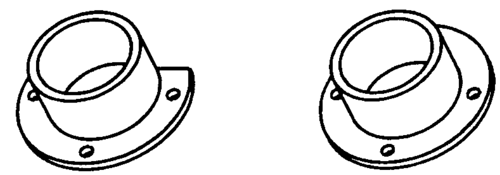
[illegible] FLIGHT 60 MOTOR



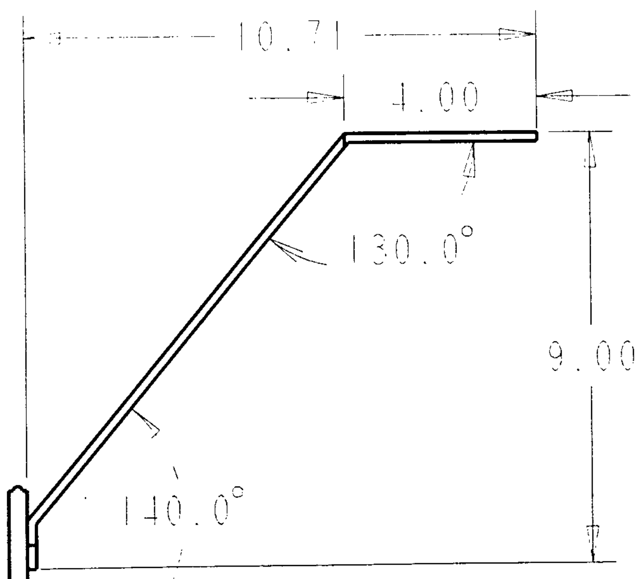
MOTOR MOUNT



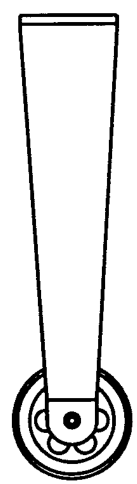
WHEEL



TAIL BOOM MOUNTS



SCALE 0.250
LANDING GEAR



NOTES:
1. ALL PARTS 0.5 SCALE UNLESS OTHERWISE NOTED
2. ALUMINUM MOTOR MOUNT COMMERCIALLY OBTAINED MODIFY AS NEEDED FOR MOTOR WIRING
3. ALL ALUMINUM PARTS MADE FROM 6061-T6 ALUMINUM
4. MACHINED WHEELS FROM 2.25" ALUMINUM STOCK.
5. USE 3/16" #328 O-RING FOR TIRES
6. USE 5/32" SPRING STEEL AXLES FOR WHEELS
7. USE 6-32 CASE HARDENED SCREWS FOR ALL FASTENERS
8. MACHINED TAIL BOOM MOUNTS FROM ALUMINUM STOCK
9. CONSTRUCT LANDING GEAR USING 3/16" THICK ALUMINUM PLATING
10. COMMERCIALLY OBTAIN NOSE GEAR/TAIL GEAR

VARIOUS COMPONENTS			
SIZE A	DWG NO MONTY'S REVENGE - DBF 2002		REV
SCALE NTS		SHEET	5

PRO/E

6. Manufacturing Plan

With the completion of the detailed design, choices had to be made regarding types of materials that were to be used for the component parts. A timeline had to be created to facilitate the order of component construction to ensure completion of the final product in a timely manner; Table 6-3 is a timeline of the manufacturing schedule.

6.1 Manufacturing Processes Investigated

Processes of manufacture were determined by evaluating all the major component parts with each of the possible materials. Each process was rated on a number scale (figures of merit) in which high numbers were desirable characteristics and low numbers were undesirable characteristics. Evaluation involved seven characteristics: availability, required skill levels, cost, time to manufacture, ease of manufacture, reliability, and weight. Component parts evaluated included the ball tray, fuselage, wings, landing gear struts, wheels, tail, tail boom, and motor mount.

6.1.1 Details of the Characteristics Evaluated

Each of the characteristics below has been broken down and is described detailing the thought process of the construction team.

6.1.1.1 Availability

Scaling is from 0 to 2. If the materials were on hand, it received a 2. If they had to be manufactured or purchased, it received a 1. If the material was not available due to reasons beyond the team's control, it received a 0.

6.1.1.2 Required Skill Levels Needed

A number of 5 was issued if the construction technique was a relatively simple process. A low number or a 1 indicated a prohibitively difficult process or if manufacturing equipment was not available. Skills of team members were also used in order to make an informed decision about the manufacturing process within each of the categories. Each member who posses the skill was given a value of 1, and others were given a zero. The results are in Table 6-1.

	Skills						
	Model plane building	Wood working	CNC	lathe	composites	hotwiring	electric motor power plants
Number of personnel	2	4	2	2	5	5	2

Table 6-1 Skills Matrix

6.1.1.3 Cost

A value of 1 was assigned if the cost of material was beyond the means of the project's budget. A high number of 5 indicated inexpensive construction costs. A special case for this category was that if high value material was already on hand or if it was to be donated, it was given a high number (low cost).

6.1.1.4 Time to Manufacture

A high value of 2 was given to materials that would allow for a relatively rapid construction; a low value of 1 was placed on materials that would need extended time.

6.1.1.5 Ease of Manufacture

If the component to be manufactured would be easy to create, it was given a high number up to a maximum of 5. If the process was beyond the skill of the members, a 1 was given.

6.1.1.6 Reliability

From past experience or from technical expertise of some of the members, each component was rated by the reliability of the materials. A 5 indicated a high amount of reliability while a 1 indicated a low reliability.

6.1.1.7 Weight

Since weight is always a major factor in airplane construction, it was considered in a way to make the weight value relative to its strength. A high value of 5 was issued for low weight materials, and a low value of 1 was given for heavy materials. Special considerations were given if the minor use of heavy materials was an advantage over the major use of lightweight materials.

Intentionally left blank

6.1.2 Figures of Merit

Evaluation of the parts is summarized in Table 6-2 below. The values for each material were averaged returning a number in which the highest number was chosen as the desired manufacturing process.

		Availability 6.1.1.1	Required Skill Levels 6.1.1.2	Cost 6.1.1.3	Time to Manufacture 6.1.1.4	Ease of Manufacture 6.1.1.5	Reliability 6.1.1.6	Weight 6.1.1.7	Average
Ball Tray	carbon-fiber	2	3	5	1	1	5	5	3.14
(internal frame)	fiberglass	2	3	5	1	1	5	3	2.86
6.2.1	wood	1	5	3	2	4	4	4	3.29
Fuselage	carbon-fiber	2	3	5	1	3	5	5	3.43
	fiberglass	2	3	5	1	3	5	3	3.14
6.2.2	wood[2]	1	5	3	2	4	2	4	3.00
Wing	carbon-fiber[1]	2	3	5	2	3	5	5	3.57
	fiberglass	2	3	5	2	3	5	3	3.29
6.2.3	wood[2]	1	5	3	2	4	2	4	3.00
Landing Gear	carbon-fiber	2	3	5	1	1	3	5	2.86
6.2.4	aluminum	1	4	4	2	3	5	3	3.14
Wheels	commercial	2	5	2	2	5	2	1	2.71
	plastic	0	5	2	2	5	1	5	2.86
6.2.5	aluminum	1	3	4	1	4	5	3	3.00
Tail	carbon-fiber[1]	2	3	5	1	3	5	5	3.43
	fiberglass	2	3	5	1	3	5	3	3.14
6.2.6	wood[2]	1	5	3	2	5	4	5	3.57
Tail Boom	carbon-fiber	0	1	1	1	3	3	5	2.00
	fiberglass	0	1	1	1	3	3	3	1.71
	wood	1	5	3	2	4	2	1	2.57
6.2.7	aluminum	1	4	3	2	2	5	3	2.86
Motor Mount	aluminum	1	3	3	1	2	5	3	2.57
6.2.8	plastic	2	1	0	2	1	2	2	1.43

[1]Foam core

[2]Balsa wood framing covered with plastic.

Table 6-2 Figures of Merit Summary for the Manufacturing Process

6.2 Major Component Manufacturing Process

Manufacturing of the major components will use a wide range of processes including composites, foam, wood, plastic, metal, and commercially obtained parts. Detailed below are the results of the FOM evaluation, describing the basic processes to be used by the construction team.

6.2.1 Ball Tray

Wood was the best choice for the ball tray. A single sheet of 1/8" plywood will comprise the tray. Holes, with a 3" diameter, will be drilled to hold the softballs in the prescribed pattern. Bulkheads will be made with the same type of plywood for extra structural strength.

6.2.2 Fuselage

The fuselage will be constructed using the ball tray as the frame. The motor mount base will be constructed and be an integral part of the frame before laying on the carbon fiber. Foam will then be placed within the wood frame and shaped into the form of the fuselage. Templates will be used to ensure close tolerances with the original design. Mountings for the wings will also be added to the fuselage's foam core and shaped to transition the fuselage cross-section to the root airfoil shape. Wood end pieces (1/8" plywood) in the shape of the wing root will be glued to the wing mountings to provide for a durable end plate. A hole will be drilled in the end plate for the passage of the brass tube, which acts as a sleeve for the carbon fiber connecting rod. It will also allow an area for the carbon fiber/epoxy to adhere to. Once shaping is complete, carbon fiber (5.8-oz/sq. yd.) and epoxy will be used to form the outer shell. Upon curing of the epoxy the fuselage is sanded and another coat of epoxy will be added to create a smooth finish. The payload bay hatches will be cut and most of the foam removed to minimize weight. Some foam may remain in order to maintain some stiffness of the carbon fiber skin.

6.2.3 Wing

The wing will be constructed with a hotwired foam core and unidirectional carbon fiber (8.9-oz/sq. yd.) skin. Wood (1/8" plywood) end pieces will be placed on the roots to match the airfoil shape; these end pieces are identical to the ones made for the fuselage wing mountings. To ensure a smooth surface on the wings, they will be vacuum bagged with a Mylar surface during the epoxy curing process.

6.2.4 Landing Gear

Main landing gear struts will be machined from sheet aluminum and bent into the required shape. Since the choice of the landing gear configuration (nose or tail wheel) was undecided at the time of this report the manufacturing process could only be speculated. Either choice will rely on commercially purchased gear and will be steerable by a servo.

6.2.5 Wheels

After evaluating the wheel type options, aluminum rims will be machined from 6061-T6 aluminum and a 3/16" o-ring used for the tire. Axles will be commercially obtained.

6.2.6 Tail

Three versions of the tail were built, so that they could be evaluated for weight considerations. The lightest version was the balsa frame with plastic coverings. Balsa will be reinforced with spruce and lightweight fiberglass in the critical areas.

6.2.7 Tail Boom

The decision to go with an aluminum tail boom resulted from the lack of 1" commercially available composite rods. Aluminum rods have the advantage of stiffness, which was also a consideration in the reliability category.

6.2.8 Motor Mount

Two types of commercial motor mounts were located for the Astro Flight motors. One mount was a plastic mount made for gas engines, which was modified for the electric motor. The other type was a machined aluminum mount. Evaluation of the weight of both types determined that the aluminum version was better. Although the aluminum mount was made specifically for the motor, some modifications had to be done on the CNC machine, specifically mounting holes and holes for the brush wirings.

	Manufacturing Milestones																																											
	Jul					Aug				Sep				Oct					Nov				Dec					Jan				Feb				Mar				Apr				
Milestone	2	9	16	23	30	6	13	20	27	3	10	17	24	1	8	15	22	29	5	12	19	26	3	10	17	24	31	7	14	21	28	4	11	18	25	4	11	18	25	1	8	15	22	29
ball tray (scheduled)																																												
ball tray (actual)																																												
fuselage (scheduled)																																												
fuselage (actual)																																												
wing (scheduled)																																												
wing (actual)																																												
landing gear (scheduled)																																												
landing gear (actual)																																												
wheels (scheduled)																																												
wheels (actual)																																												
tail (scheduled)																																												
tail (actual)																																												
tail boom (scheduled)																																												
tail boom (actual)																																												
motor mount (scheduled)																																												
motor mount (actual)																																												

Table 6-3 Manufacturing Milestones

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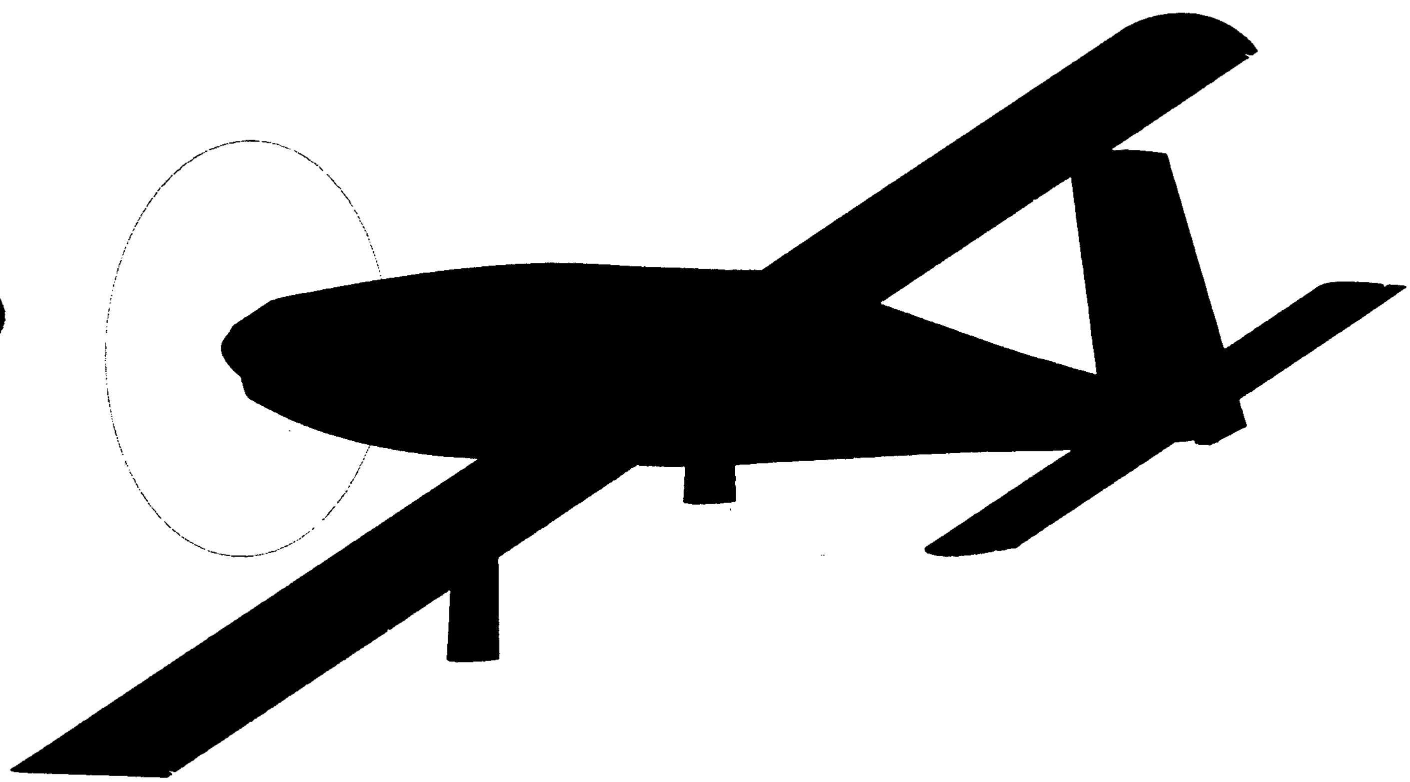
2001/2002 AIAA Foundation Cessna/ONR Student Design/Build/Fly Aircraft Team Competition

Design Report



Marooned

Virginia Polytechnic Institute and State University

March 2002

Table of Contents

1.0 Executive Summary

This year the Cessna/ORN Student Design/Build/Fly (DBF) Competition is going to be hosted by the Cessna Aircraft Company's Filed test facility in Wichita, Kansas. The mission for competitions changes from year to year. This year's mission profile is that competing team are going to fly three sorties, which are parts of the flight mission. The mission is restricted with time, number of softballs for the payload, and five flight attempts.

Virginia Tech (VT) Student DBF team set a number of short and long-term goals for this year aircraft design. Long team goals were: to successfully compete in the 2001-2002 Cessna/ORN Student DBF Competition, and to build a solid program at Virginia Tech for next year's competition. Short-term goals were finishing everything in a timely manner to accomplish long-term goals.

Looking from real-life perspective, the aircraft design is going to find use among large and small business corporations to be used for business trips, short between cities express shuttle services, and private use. The aircraft is gong to have the capacity of 24 passengers including the pilot and the crew.

1.1 Conceptual Design

The Conceptual Design phase was broken into several subroutines: Wings, Tails, Landing Gear, Fuselage, and Propulsion. Various aircraft components and aircraft design configurations are evaluated. Each aircraft configuration and component was rated based on criteria referred to as the figures of merits.

In the conceptual design, numbers of alternative wing configuration were considered. The wing design configurations were: high mono-wing, mid mono-wing, biplane, and tandem wing. The team also analyzed a mono-wing with flaps that could be either high, mid, or low. Each configuration is analyzed and rated aircraft const (RAC) is computed.

The tail alternative configuration considered were: T-tails, Y-tails, V-tails, Canards, conventional and no tail configurations. Every tail configuration is evaluated based on RAC, efficiency, drag profiles, and ease of building.

The team developed five different landing gear concepts for landing gear configuration. They were as follows: tricycle, conventional, wing- tip, sailplane gear, and quadracycle. Landing gear were rated based on drag profiles, weight, and constriction difficulty.

Based on the team discussions, four different fuselage concepts were considered. These four concepts were a conventional fuselage, a blended-wing-body, two fuselages with two engines, and one fuselage with two cargo/engine pods. The factors in determining the best solution for fuselage were RAC, drag profiles, and weight. The fuselage concepts were rated based on those three factors.

Propulsion system consisted of three major components: propeller, engine, and batteries. The propeller concepts considered for conceptual design were single-bladed propeller, two-bladed propeller, three-bladed propeller, four-bladed propeller, and pusher propellers. The engines considered in Conceptual Deign are all brushed direct/geared Astroflight motors. Also, eight different Ni-Cad cell manufacturers were considered. Engine selection was based on engine data that was provided by

manufacturer. Rated results are tabulated in FOMs. Propeller decisions were based on several characteristics such as ground clearance, performance, amiability, and weight. Ni-Cad cell was rated according to capacity over the weight ration.

1.1.1 Design Tools

Design tools used varied from hand calculations to computer program/codes and computer graphics. One very valuable tools used was MS Excel based code from previous year's teams. First, the code had to be adapted to fit this years mission profile and rules. This tool allowed easy manipulations of many aircraft design/configuration parameters. The code was to estimate RAC for each composite aircraft configurations. For propulsion system calculation by hand, test and MotoCalc program was used to evaluate and rate each configuration. MotoCalc was used mainly to verify and provide assistance to on decisions that were maid.

1.2 Preliminary Design

The Preliminary Design phase of the 2001/2002 VT DBF team was used to create basic dimensions of the Marooned aircraft. The team completed this design phase by splitting into separate teams, the Wing Team, the Fuselage Team, and the Propulsion Team. Splitting into the gropes improved the efficiency of the team and it was possible to hit several major areas of interest at the same time. In the Preliminary Design phase, the following subroutines were implemented: Wing, Tail, Landing Gear, Fuselage, and Propulsion.

In this part of the wing design phase geometry, configurations and structure were evaluated and rated. The wing's span, AR, and airfoil were all analyzed decided upon. The team also did some test building that provided better estimates of the wing weight required to meet the competition's and team's requirements. The fuselage team also analyzed fuselage length, ball layout, and general composition during this phase. Their own test constructions helped refine the fuselage structure.

Tail sizing and characteristic that had to be found were assigned to the same group of students that was working on wing. That was done because much of the tails sizing is dependent on aspects of the wing. The study of previous VT DBF team aircraft tails and model aircraft tail was done to better understand and find the way to relate to the aircraft that is being designed.

In Preliminary Design phase, after the FOM and hours of team discussion resolving landing gear issues the final verdict was reached. The team decided on conventional landing gear. Many factors supported the decision. Drag profile, difficulty of construction, and RAC played major role in deciding on landing gear design.

In the Preliminary Design propulsion phase three Astroflight engines, two-bladed propellers and two Ni-Cad cells were chosen for further analysis. In analysis of these components, many techniques were used: computer program calculation, wind tunnel tests, static-trust tests, independent calculation by hand, and VT DBF engine performance research and past experience. Engines were analyzed and

evaluated based on data taken during test runs and test data modifications. Final decision on Ni-Cad cell battery manufacture and cell were aided. For propeller it was determined that it is needed certain amount of thrust to fulfill aircraft requirements. This fact was used in further eliminating and making decisions.

1.2.1 Design Tools

The MS Excel code was used extensively in the preliminary design phase. Some independent calculations and computation were preformed to ensure accuracy of the code. At some instance more theoretical work was needed to achieve the short-term goals, so the works done by other engineers that were published in book were used to evaluate aircraft components. For the propulsions system the methods used the most were real life simulations in the Virginia Tech Open-Jet wind tunnel, static thrust in the VT DBF Lab, and past and new experience.

1.3 Detailed Design

The Detailed Design phase of the 2001/2002 VT DBF project required an optimization of the Preliminary Design first. Once this optimization was completed aspects of the design needed to be computed. These computations included weight and balancing of the aircraft, performance of the aircraft, handling qualities of the aircraft, and finally the RAC for the final design. These analysis and computations would provide the team the necessary information to build the Marooned aircraft.

Final fuselage and tail design are achieved which included longitudinal wing position with respect to fuselage, final fuselage dimensions, locations an characteristics of empennages.

In the Detailed Design phase a lot of time was spent on the propeller analysis and selection of the right propeller for the aircraft. Every aspect of the aircraft design was explored and discussed in detail. The aircraft's structural and flight performance characteristics were pushed to the limit, with a certain margin of safety and error incorporated. This was done in order to achieve the best possible results for the flight score. For the propeller selection, a number of static and wind tunnel tests were performed to optimize the propeller-engine-battery combination. At the end of the analysis, the two propellers that performed the best were chosen.

1.3.1 Design Tools

In Detailed Design phase most of the calculation was done by had and with help of computer programs such as MS Excel, MotoCacl and Mechanical Desktop 4.0. However, many references were used from the books to help determine the final configurations and specifications. Also numbers of codes were written in MS Excel to simplify Detailed Design computations.

2.0 Management Summary

Six members from last year's VTBDF team came together again, having the same goal of designing the aircraft that would successfully compete at the 2001-2002 Cessna/ONR Student Design/Build/Fly aircraft competition that is going to be held at the Cessna Aircraft Company in Wichita, Kansas.

The returning members of the team were mainly juniors and seniors. One of the primary goals was to recruit new people-- freshmen, sophomores, and juniors. This was done to keep Virginia Tech's Design/Build/Fly tradition alive in successive years. After two weeks of advertising in classrooms, hallways, and AIAA meetings to get people interested, the team size tripled. This initial success of gathering motivated students bust up the VT DBF teams attitude and opened their minds toward new frontiers that have not been yet explored.

2.1 VTDBF Architecture

On the top of the agenda for the fist meeting was to set goals and make a reasonable timeline to achieve those goals. The decisions were that VT DBF team would hold meetings twice a week, and contacts amongst the groups between meetings was maintained via email and web site. The Web site was primarily for advertisement, getting people interested in team and posting major success in teamwork and design phases. In the spring semester, the VT DBF team met three times a week for the specific purposes of designing, report writing, and building. This was necessary to complete everything in a timely manner.

The VTDBF Team Leader arranged meeting times and conducted meetings in an orderly fashion so all opinions could be expressed and discussed. The team was split up into the Fuselage Team, the Propulsion Team, and the Wing Team. Each team has an assigned team leader whose primary objective was to coordinate the individual efforts of the team members to meet the final goals of that team. Table 2.1.1 (Table 2.1.1 is shown on the next page) shows the summary of team organization.

Entire VT DBF team met once a week. On the meetings, each sub team reported their progress, ideas, and plans for improvements. After each sub team has presented their work and plans, problem-solving session starts. Here each team addresses an issue and problems what it might have with changes of other team. This problem solving sections lasted anywhere between a half an hour to several hours of discussions. This provided solid discussions of problems and ideas implemented during sub team presentations. Challenge is the only way the team can improve and each team member has learned that the dream work is built on solid foundations of teamwork.

VT DBF Team Leader: Dzelal Mujezinovic Chief Engineer: Tim Miller Faculty Advisor: Frederic Lutze		
Fuselage Team Leader: Mike Metheny	**Propulsion Team** Leader: Dzelal Mujezinovic	**Wing Team** Leader: Tim Miller
Charles Ashton Robert Sidell Jeremy Andrews Derek Jackson Mike Reilly Aaron Block	Rodney Bajnath Chase Ashton Lobus Brienda	Amy McIvor Ernie Keen Andrew Krohn Stephen Banas Steven Todtenhagen Eric Danner

Figure 2.1 VT DBF Team organization summary and assignments

2.2 Configuration and Schedule Control

As the 2001-2002 school year began and our VTDBF team formed, our team leader set out a plan for us to follow. This plan described our timetable of tasks to be performed and a means in which to accomplish them. Every VT DBF team member was assigned to a sub team. From there, sub team leaders assign task to each sub team member. Personal assignments by sub teams and design phases are presented in Table 2.1. Our team leader assigned dates to several milestones and made sure we stayed on task. The key factor to triumph is hard individual work, teamwork, and challenge. One of the biggest challenges faced was time line. It was intense, moving fast, and there were no breaks or time for relaxing. This could be see from Figure 2.2, the VT DBF team met almost all of the deadlines on time, which showed team members hard work and willingness to succeed.

As our knowledge of design and principle concepts strengthened during the early period of VT DBF team, our plan was slightly changed. Upon returning from Thanksgiving Break, our team leader, with help from the team, modified our timeline to fit our dynamic design process for goals already achieved and for those yet to be completed. The detailed milestone of VT DBF team with planned schedule and actual timings is shown in Figure 2.2.

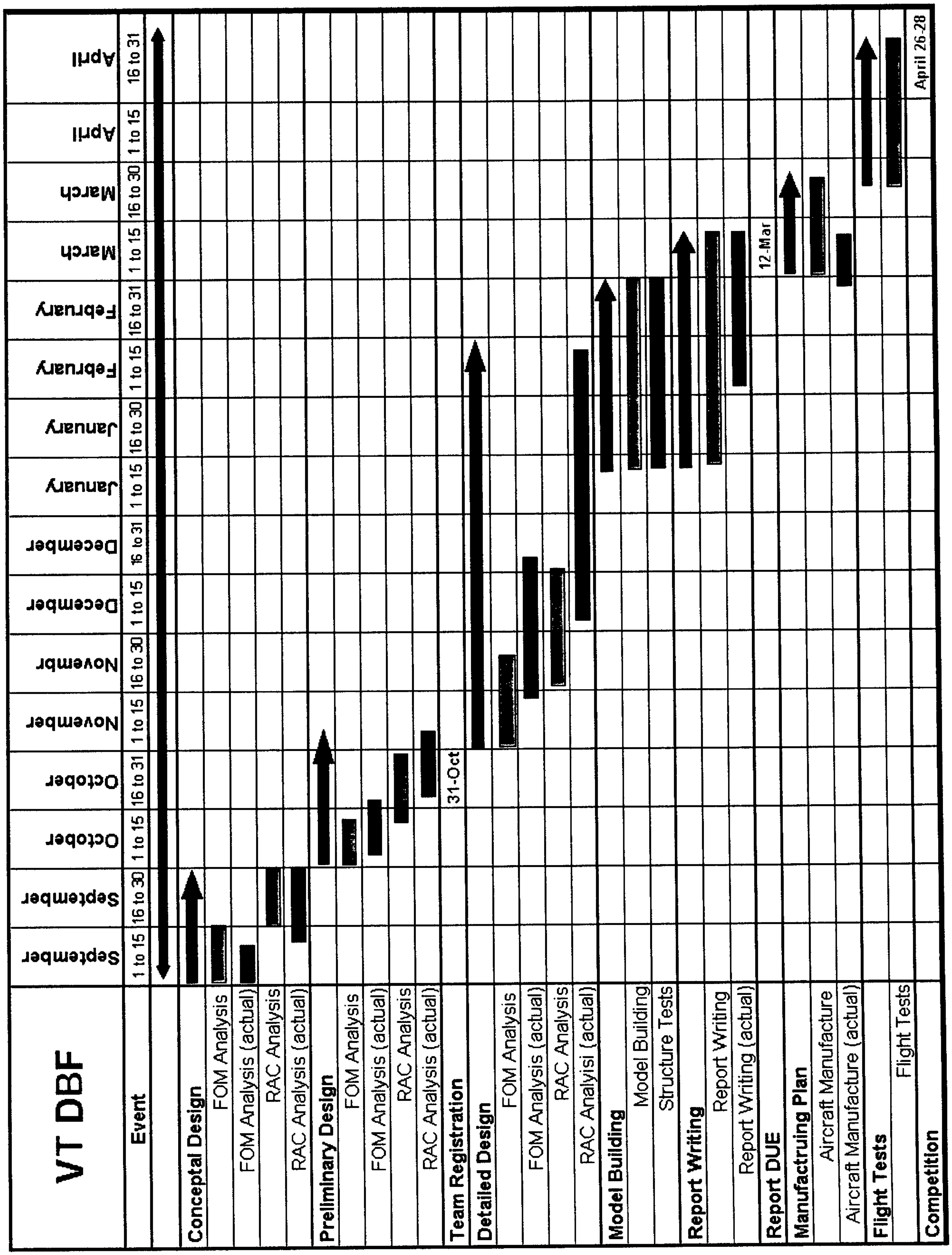


Figure 2.2 VT DBF Team Milestone Chart.

	Jeremy Andrews	Charles Ashton	Rodney Bajnath	Stephen Banas	Lobus Brieda	Eric Danner	Derek Jackson	Ernie Keen	Andrew Krohn	Amy McIvor	Mike Metheny	Tim Miller	Dzelal Mujezinovic	Mike Reilly	Robert Sidell	Steven Todtenhagen
A. Wing Team																
1. Conceptual Design																
Wing Concepts	2	1	0	5	0	4	0	4	3	5	0	5	0	0	0	3
Landing Gear Concepts	3	5	0	5	0	2	0	2	3	4	0	5	0	0	0	2
FOM/RAC	4	3	0	5	0	4	0	3	4	4	5	5	0	0	0	2
Wing Analysis	4	2	0	5	0	2	0	5	5	5	5	5	0	0	0	3
2. Preliminary Design																
Wing Analysis/Geometry	4	1	0	5	0	2	4	2	4	4	5	5	0	0	0	2
Wing Structure	4	1	0	5	0	4	4	5	4	4	5	5	0	0	0	2
Material Research	3	1	0	5	0	4	4	3	2	5	5	5	0	0	0	3
3. Detailed Design																
Wing Analysis/Structure	2	1	0	0	0	5	4	0	2	5	5	5	0	0	0	2
Material Decisions	2	1	0	0	0	4	2	2	5	4	4	5	0	0	0	2
Final Decisions	2	1	0	0	0	4	2	5	2	5	4	5	0	0	0	1
4. Manufacturing																
Wing Building	2	1	0	0	0	4	0	0	2	5	4	5	1	5	0	3
B. Propulsion Team																
1. Conceptual Design																
Engine Selection	2	5	5	0	0	0	0	0	0	0	3	1	5	0	0	0
Battery Selection	4	4	5	0	0	0	0	0	0	0	4	1	5	0	0	0
Propeller Selection	2	5	5	0	0	0	0	0	0	0	3	2	5	0	0	0
Calculations	5	4	5	0	0	0	0	0	0	0	3	1	5	0	0	0
2. Preliminary Design														0		
Engine Selection	3	4	5	0	0	0	0	0	0	0	3	4	5	0	0	0
Battery Selection	3	4	5	0	0	0	0	0	0	0	3	2	5	0	0	0
Propeller Selection	3	4	5	0	0	0	0	0	0	0	3	3	5	0	0	0
Wind Tunnel Tests	4	4	5	0	0	0	0	0	0	0	3	1	5	5	0	0
Static Thrust Tests	4	4	5	0	1	0	0	0	0	0	3	1	5	5	0	0
Analysis	2	4	5	0	2	0	0	0	0	0	3	4	5	0	0	0
3. Detailed Design																
Engine Selection	2	5	4	0	0	0	0	0	0	0	2	1	5	0	0	0
Propeller Selection	2	4	4	0	0	0	0	0	0	0	4	1	5	0	0	0
Wind Tunnel Tests	2	3	5	0	4	0	0	0	0	0	3	1	5	0	0	0
Static-Thrust Tests	2	4	2	0	2	0	0	0	0	0	3	1	5	5	0	0
Analysis/Final Decisions	1	5	2	0	4	0	0	0	0	0	3	1	5	0	0	0
C. Fuselage Team																
1. Conceptual Design																
Fuselage Concepts	2	5	0	5	0	0	5	5	3	0	5	3	3	0	4	4
Tail Concepts	3	2	0	5	4	0	5	5	4	1	5	3	3	0	4	4
FOM/RAC	3	3	0	5	0	0	5	5	2	1	5	3	2	0	4	3
2. Preliminary Design																
Fuselage Analysis	3	2	0	5	0	0	5	5	2	2	5	3	3	0	3	4
Tail Analysis	3	3	0	5	4	0	5	5	4	0	5	3	3	0	3	3
Material Research	3	4	0	5	0	0	5	5	5	4	5	3	4	0	3	2
Tail Structure Tests	3	5	0	5	5	0	5	5	4	5	5	5	5	5	3	3
3. Detailed Design																
Fuselage Analysis/Structure	3	4	0	0	0	0	5	4	4	2	5	3	2	0	0	2
Material Decisions	3	4	0	0	0	0	4	4	4	0	5	3	2	0	0	3
Final Decisions	1	2	0	0	0	0	2	2	4	1	5	3	2	0	0	3
4. Manufacturing																
Model Building/Testing	2	4	0	0	5	0	0	1	4	4	5	5	5	5	0	3
E. Report Writing																
Graphics	1	2	0	0	0	1	0	0	0	3	5	5	3	0	0	0
Writers	1	4	0	0	0	3	0	1	0	5	5	5	4	0	0	0
Proof Readers	1	4	0	0	0	1	0	1	0	3	5	5	2	0	0	0

Table 2.1 VT DBF Personal assignments and design phase summery.

On the scale from 0 to 5, 5 indicate maximum involvement.

3.0 Conceptual Design

The team considered many different design options during The Conceptual Design phase. The team decided which design variables were driving factors in the design. The team then considered many configurations that could be used to build the final design. These designs were evaluated with Figures of Merit (FOM), and numerical representation of their Rated Aircraft Cost (RAC). The results of this evaluation determined which configurations were studied further in the Preliminary Design phase.

3.1 Subsystem Conceptual Design

3.1.1 Wings

The most important aspect of an airplane's flight can be considered its wing. The team analyzed six different concepts for wing design. They were as follows: high mono-wing, mid mono-wing, low mono-wing, biplane, and tandem wing. The team also analyzed a mono-wing with flaps that could be either high, mid, or low. These wings were assumed to have a taper ratio of 1. This was set to reduce the difficulty of construction

The high mono-wing consists of one wing mounted atop the fuselage. This design is the most stable in flight and has a low RAC. However, the team felt that it would interfere with the loading and unloading of the payload.

The mid mono-wing consists of one wing mounted in the middle of the fuselage. This is the most aerodynamic design and has a low RAC. However, it can be less stable in the air, and the support structure would interfere with payload capacity.

The low mono-wing consists of one wing mounted at the bottom of the fuselage. This design does not interfere with payload loading and unloading or payload carrying capacity. It also has good ground effect, fair stability, and has a low RAC. Having the wing mounted low also allows the landing gear to be mounted to the wing.

The biplane wing design consists of two wings: one mounted atop the fuselage, and one mounted at the bottom, directly underneath the top wing. This design has excellent lift, stability, and handling characteristics. However, having two wings produces a lot of drag, which slows the aircraft down. In addition the two wings have a higher RAC than a mono-wing.

The tandem wing design is similar to the biplane. This design consists of two wings mounted on the same level, either low, mid, or high. One wing is placed in front of the other. This design also produces high lift. However, the second wing increases drag and handling characteristics are less.

The flapped wing concept is the same as the high, mid, and low mono-wing, with the addition of flaps on the trailing edge of the wing. The flaps allow the wing to reach higher maximum lift coefficients without an increase in wing area. This also allows for a wing with smaller wing area to land at a slower speed. The flapped wing has a few disadvantages, like a higher RAC and an increase in building complexity.

3.1.2 Tails

The aircraft's stability is an important consideration in the design and is dependent on the tail configuration. After the team's experiences at last year's competition, this fact was made alarmingly clear. Many different tail types were considered, from conventional to tailless designs.

The conventional tail arrangement consists of a horizontal stabilizer mounted to the fuselage, with a vertical tail mounted to the fuselage. This is a proven design that has flown on the majority of aircraft produced throughout the world. It is known for its good stall characteristics, ease of manufacture, and separation of roll and yaw modes during maneuvers. This tail design also facilitated burying a tail wheel in the rudder for a conventional gear design.

The T-tail is well known for its improved performance over the conventional tail design. This is evidenced in part by its wide use in small business jet design. This is because the horizontal tail has an endplate effect on the rudder, which improves the rudder's efficiency. This is usually realized in the form of reduced vertical tail area. Unfortunately, the mounting of the horizontal tail at the top of the vertical tail requires more strength of the vertical tail to take the loads created by the horizontal tail. This results in a disadvantageous increased tail weight.

The tailless aircraft is a very desirable configuration from an aerodynamic and structural standpoint. This design eliminates all friction drag from having a tail boom and tail surfaces. The tailless aircraft has many disadvantages. The skin-friction drag reduction is often more than compensated by an increase in trim drag due to the wing's required reflexed airfoil. Another major disadvantage is the reduction in static stability margins, as witnessed in the team's 2000/2001 tailless design, which had a full scale c.g. envelope less than ¼ of an in.

Two families of tail designs were also considered. One was the canard family; the other was the Y/V tail family. The canard family consisted of a canard with either 1 or 2 vertical surfaces. The 2-vertical-surface canard design is well known from Burt Rutan's Vary-EZ line of aircraft. The single-vertical-surface canard concept was an attempt by the team to improve the RAC of the canard concept by requiring only one vertical surface. The canard is a beneficial design because the canard is providing an upward lifting force during cruise, instead of the conventional tail's negative balance force. However, a disadvantage to this design is an increase in wing area required for takeoff. It is difficult to integrate flaps onto the wing of a canard aircraft because the canard must be able to compensate for the increase pitching moment produced by the deflected flap.

The Y/V tail family consists of four designs that are very similar. There are the Y-tail, the V-tail, the inverted Y-tail, and the inverted V-tail. The V-tail design is advantageous because in the new RAC rules it provides a price break. Also the V-tail can have less drag because it has fewer intersections than other tail designs. The V-tail's drawback comes from the coupling of yaw, pitch and roll responses into deflections of the tail surfaces. The inverted V-tail is an attempt to make some of these responses work to the benefit of the plane. For example, an inverted V-tail performing a left yaw would also induce a left bank. This design is limited by the tail clearance required to implement this design. Another attempt at

fixing the coupling associated with the V-tail is found in the Y-tail. This design is a V-tail with a small moveable vertical surface on the underside of the tail boom. This surface provides increase yaw response, requires less V-tail area, and can be used to hide a conventional gear's tail wheel. The inverted Y-tail attempts to provide the beneficial roll-yaw response of the inverted V-tail, with the improved yaw control of the Y-tail. This comes at the same expense of increased tail clearance. Both Y-tails also fail to gain the RAC benefit of the V-tail and could potentially have a cost higher than the conventional tail.

3.1.3 Landing Gear

Landing gear is essential to a successful flight; without it, the plane cannot takeoff or land without causing damage to itself or surroundings. It has to be strong enough to support the entire weight of the aircraft on the ground, and it has to have low drag and low weight to have a minimum influence on its flight characteristics. This means the landing gear must be light, strong, and aerodynamically streamlined. The team developed five different landing gear concepts for landing gear configuration. They were as follows: tricycle, conventional, wing- tip, sailplane gear, and quadracycle.

The tricycle landing gear consists of two main struts placed behind the aircraft's center of gravity, and a smaller strut mounted in the nose of the craft. With this design, the nose gear is used for steering, and this gives the best ground stability for a given weight. This design requires three struts, which adds weight and creates more drag. Also, incorporating mounting points for the fixed gear into the structure is more difficult. Finally, the team observed at the previous year's competition the difference in landing technique between RC and private pilots. RC pilots tend not to flair as much as private pilots and this tends to place the majority of the landing load on the nose gear. Thus, this design requires more strength of the nose gear to withstand these loads without breaking.

Conventional gear, also known as 'tail dragger,' consists of two main struts in front of the center of gravity and one small wheel attached to the underside of the tail. With this design, the tail gear can be incorporated with the rudder, which is used for steering on the ground and in the air. At rest, the plane slopes back with the nose up high and the tail close to the ground. When accelerated down the runway, the tail raises up and the plane rolls down the runway parallel to the ground. This requires only two main struts for support, which helps to reduce weight and drag.

The wingtip gear is essentially the same as the traditional conventional landing gear. In this design, the two main wheels are mounted out on the wingtips. The advantage to this design is that it would be more aerodynamic and cause less drag. The disadvantage is that it would put more stress on the wings, causing them to be reinforced, adding more weight.

The sail gear is described as having one large wheel mounted in the center of the plane, just in front of the center of gravity, a small tail wheel, and two small wheels out on the wingtips. This design has good aerodynamics, it is lightweight, and it has good ground control. The disadvantage to this design is that it cannot provide for the necessary propeller clearance.

Quadracycle landing gear is described as two wheels near the front of the aircraft, and two wheels in the back. The advantage to this is that the plane would be stable and easy to control on the ground. The disadvantages are that it would add excess weight, and the steering mechanism would be difficult to construct.

3.1.4 Fuselage

The fuselage is a key component of the aircraft because it is the hub for all other load-bearing structures. Therefore, it must be able to withstand all the stresses placed upon it by the engine, wing, tail, and payload. The fuselage must be aerodynamically sound to reduce drag. It must also be designed to have the maximum cargo-space-to-area ratio, and allow for easy loading and unloading of payload. Through team discussion, we considered four different fuselage concepts. These four concepts were a conventional fuselage, a blended-wing-body, two fuselages with two engines, and one fuselage with two cargo/engine pods.

A conventional fuselage is considered to be a long slender structure with a single motor. This design is advantageous because the cross sectional area is reduced due to the long structure, which can reduce drag. Another advantage is ease of construction, because only one fuselage is to be constructed. Other advantages are large payload capacity and relatively low weight.

The next concept was the blended-wing-body. In this design, the root of the wing would be as thick as the fuselage itself. The wing would gradually increase in thickness until it smoothly joined with the fuselage. This would cause a larger fuselage cross-sectional area, which would increase drag. This design would be easy to build, the RAC would be low, and the aerodynamic qualities around the wing-fuselage would be better. The wing would have to be mounted midway on the fuselage, which would interfere with payload, reducing capacity.

The two-fuselage design consists of two booms, each carrying payload, and each with an engine mounted in the front. The wing and tail would join the two booms together and provide structural support. Some advantages to this design are that the individual cross-sectional area would be reduced and the additional engine would provide greater thrust and speed. A disadvantage to this design includes a high RAC due to the combined length of both fuselages and the second engine. Finally the added structural support of second fuselage would weigh more.

The one fuselage, two-pod design consists of one fuselage similar to the conventional fuselage with two shorter fuselages, or pods, mounted a short way out on the wing. These two pods would house one engine each, and possibly some payload. Some advantages to this design include efficient payload area produced by placing payload in the two pods and increased thrust and speed due to the additional motor. Some disadvantages are the increased weight from the additional motor and pod structure, a larger cross-sectional area, a larger RAC. As well, loading and unloading of payload in pods may be more difficult.

3.1.5 Propulsion

The propulsion system consists of three major components and electronic sub systems. The three major parts of the propulsion system are the propeller, the engine, and the batteries. The electronic sub system is composed of the speed controller, receiver, transmitter, and wiring system.

The task for the propulsion team was to investigate and optimize the components and the configuration of the propulsion system to meet all required criteria while producing the highest possible flight score. The four criteria are to takeoff within 200 feet, fly all three missions within 10 minutes, have a battery flight pack weight less then 5lbs, and have a current draw less than 40 amperes. To meet these criteria, the propulsion team conducted research on engines, propellers, batteries, and different combinations of propellers and engines.

The engines considered in Conceptual Deign are all brushed direct/geared Astroflight motors. The propeller concepts considered for conceptual design were single-bladed propeller, two-bladed propeller, three-bladed propeller, four-bladed propeller, and pusher propellers. Research was performed to find the best Ni-Cad cell configuration that would suit the needs of the team best. Eight different Ni-Cad manufacturers, ranging from Ni-Cad-Toy cell manufacturers to Ni-Cad-aircraft/spacecraft cell manufactures were considered, and a detailed analysis of each was done.

3.2 Conceptual Design Decisions

With all the different design options available to the team, a method for choosing the best design was required. The design decisions were broken into two segments. These segments were wing / fuselage / tail / landing gear (body), and propulsion. The 'body' team decided on a two-iteration selection process that began with the FOM analysis. The second iteration would be done with an individual analysis of each element of the aircraft. The decision process would conclude with a combination of an FOM and RAC analysis.

3.2.1 Figures of Merit

For the task of choosing a particular concept, the team discussed each and rated them on a scale from one to five, where five was good or stable, and one was bad or unstable. Charts for each aspect of the design were created, and each concept was listed. Ratings were given in the following categories: Rated Aircraft Cost (RAC), Weight (W), Manufacturability and Payload (M/P), Stability and Control (S/C), and Lift to Drag (L/D). The RAC was based on the concept's ability to lessen (5) or increase (1) the RAC of the aircraft. M/P refers to how easily the structure can be built and/or how much payload it can carry, with easy access to payload. S/C refers to how stable the aircraft will be in flight and on the ground, and how easily it is controlled in these regimes. L/D refers to the lift produced by a structure compared to the amount of drag it produces.

	RAC	W	M/P	L-D	S/C
Landing Gear					
Tricycle	n/a	2	2	2	4
Conventional	n/a	4	5	4	3
Sail plane	n/a	5	4	5	1
Conventional @ Wingtips	n/a	2	3	5	1
Quad (4 Wheels)	n/a	1	2	1	4
Wings					
Mono: High	3	3	2	3	4
Mono: Mid	3	2	1	2	4
Mono: Low	3	3	4	4	3
Biplane	2	2	1	3	4
Mono - Flaps	3	3	3	5	4
Tandem Wing	2	2	2	3	3
Tail					
Conventional	2	3	4	3	4
T - Tail	2	2	3	3	3
V - Tail	3	4	4	4	2
Y - Tail	2	3	2	3	2
Inverted V - Tail	3	4	3	4	1
Inverted Y - Tail	2	3	2	3	3
Canard w/ 1 Vertical	2	3	4	3	4
Canard w/ 2 Vertical	1	2	2	4	4
None	5	5	5	5	-5
Fuselage					
Conventional	3	4	4	5	5
Blended-wing-body	4	4	5	4	3
2 Fuselages	2	3	3	3	2
1 Fuse w/ 2 Pods	1	2	2	3	1

Table 3.1 Individual Concept FOM

The FOM of the individual concepts allowed the team to focus on a selection of design concepts. The individual concepts were combined to form entire aircraft concepts, a process which eliminated several combinations. Other aircraft concepts were eliminated because of their low total FOM scores. In the end, twenty-two composite aircraft concepts were investigated further through RAC and FOM analyses. The RAC analysis was done using the Excel file "ConceptSheet", while the FOM analysis was done by multiplying the weighting factors for each concept's 4 design parameters, and finding the total FOM by adding these values together. The results of these analyses are found in Table 3.2. This table also includes the team's decision for concepts that warranted further investigation during the preliminary design phase. These included the low placed wing with and without flaps, a conventional tail, and sailplane or conventional gear. Despite its high total score, the V-tail was eliminated from further analysis

because of the team's fear of its roll and yaw coupling dynamics. This also allowed for simplification of designs to be considered during Preliminary Design.

Concept #	Wing	Tail	Gear	Fuselage	Engines	RAC	FOM	Total	P/-
1	Norm w/flaps	Conv.	Conv.	BWB	1	12.511	264	21.101	P
2	Norm w/flaps	Conv.	Conv.	Conv.	1	12.511	265	21.181	P
3	Norm w/flaps	Canard	Sail	Conv.	1	12.875	258	20.039	-
4	Norm w/flaps	V-tail	Sail	Conv.	1	12.411	256	20.627	P
5	Low	Conv.	Conv.	BWB	1	12.757	260	20.382	-
6	Low	Conv.	Conv.	Conv.	1	12.757	261	20.46	P
7	Low	Canard	Sail	Conv.	1	12.755	245	19.209	-
8	Low	V-tail	Sail	Conv.	1	12.657	259	20.463	P
9	High	Conv.	Conv.	BWB	1	12.757	251	19.676	-
10	High	Canard	Sail	Conv.	1	12.755	245	19.209	-
11	High	V-tail	Sail	Conv.	1	12.657	243	19.199	-
12	Norm w/flaps	Conv.	Conv.	BWB	2	14.845	264	17.784	-
13	Norm w/flaps	Conv.	Conv.	Conv.	2	14.845	265	17.851	-
14	Norm w/flaps	Canard	Sail	Conv.	2	15.208	258	16.964	-
15	Norm w/flaps	V-tail	Sail	Conv.	2	14.745	256	17.362	-
16	Low	Conv.	Conv.	BWB	2	15.09	260	17.23	-
17	Low	Conv.	Conv.	Conv.	2	15.09	261	17.296	-
18	Low	Canard	Sail	Conv.	2	15.088	245	16.238	-
19	Low	V-tail	Sail	Conv.	2	14.991	259	17.278	-
20	High	Conv.	Conv.	BWB	2	15.09	251	16.633	-
21	High	Canard	Sail	Conv.	2	15.088	245	16.238	-
22	High	V-tail	Sail	Conv.	2	14.991	243	16.21	-

Table 3.2 Composite Aircraft Analysis used to determine the designs warranting further design optimization. A "P" means "Passing to Preliminary Design," and "-" means no further investigation.

3.2.2 Conceptual Propulsion Analysis

The tools used to analyze propeller/engine/battery configuration were the computer program *MotoCalc*, independent calculations by hand, static-thrust tests, wind tunnel tests, and team member experiences and research.

3.2.2a Engine Analysis

Using *MotoCalc* and some independent calculations, each engine was analyzed and given a score. The engine with the highest score was taken for detailed analysis in the preliminary design. The factors used to eliminate the engines were voltage constant, torque constant, and weight. Desired factors are high volt and torque constants and low engine weight. Each factor was scored from 1 to 10, with 10 being the best. The motor scores and results are presented in Table 3.3

	Engine	Volt Constant	Torque Constant	Weight	Total Score
1	CO-035D	9	0.5	10	19.5
2	CO-035G	8	1	9	18
3	CO-05D	7.5	1	8.5	17
4	CO-05G	7	1.5	8.5	17
5	**CO-15D**	6.5	1.5	8	16
6	**CO-15G**	6	2	8.5	16.5
7	**CO-25D**	6	2	7.5	15.5
8	**CO-25G**	5	3.5	7.5	16
9	**CO-40D**	4.5	4	7	15.5
10	CO-40G	4	7	6.5	17.5
11	CO-40S	4	8	6.5	18.5
12	CO-61D	4	9	5	18
13	CO-61G	3.5	9.5	5	18
14	CO-90D	3	10	4.5	17.5
15	CO-90G	2.5	10	4.5	17

Table 3.3 Conceptual design engine selections.

According to the Table, ten of the fifteen Astroflight engines are good candidates for analysis. The goal for the conceptual design was to narrow the engine selection down to three. Batteries and propellers were included in engine selection in order to achieve this elimination. Using the *MotoCalc* program and the assumption that 7lbs of static thrust is required for takeoff eliminated four engines, including the two-engine/two-propeller combination. Another reason for the twin-engine elimination is its significantly higher RAC with little gain in performance.

3.2.2b Propeller Analysis

The propeller selection was based on independent calculations, plus previous experience and studies done by VT DBF team members. Factors in propeller selection were availability, ground clearance, performance, and weight. Each factor was scored from 1 to 10, with 10 being the best. Table 3.4 shows propeller selection and results.

Propeller	Ground Clearance	Performance	Availability	Weight	Total
Single Bladed Propellers	1	7	1	10	19
Two Bladed Propellers	4	9	10	7.5	30.5
Three Bladed Propellers	7	7	6	5	25
Four Bladed Propellers	9	3	3	2.5	17.5

Table 3.4 Propeller Selection table.

The two-bladed propeller concept scored significantly higher than all other propeller concepts. The pusher-propeller concept was eliminated because of aircraft design and configuration. One key factor that eliminated three- and four-bladed propellers were the blade separation angles of 120 and 90 degrees, which can cause wakes, vortices, and air density variations that might create other problems. Single-bladed propellers are eliminated because they are not readily available commercially.

3.2.2c Ni-Cad Cell Analysis

Battery selection was based on current team research. The primary goal of the research was to find the best cell arrangement to fit the design parameters. Out of eight different Ni-Cad cell manufacturers, two were found to fit the requirements best. One cell with highest capacity-to-weight ratio was chosen from each manufacturer.

3.3 Analysis Tools

Analysis of so many concepts would be arduous without the aid of computer computation. Fortunately the team had access to a computer code developed by the previous year's team. This was an Excel-based code, which allowed easy manipulation of many aircraft design parameters like number of fuselages, wings, tails, and the size of each item. One team member familiar with the code was assigned to update it to reflect changes in the rules for this year's competition. This also provided a chance to correct some overly optimistic assumptions made in the first code. During the Conceptual Design phase the code's use was limited to calculating RAC numbers for each composite aircraft design. Further explanation of the code will be provided in the Preliminary Design section, where the Excel code was utilized more.

Another analysis tool already mentioned was MotoCalc. The team used a free 90-day trial version of this software to do some calculations. This program was used to find several different propulsion values for many different engine/battery/propeller combinations. This programmed simplified conceptual analysis of propulsion systems, and gave theoretical values that experimental data could be checked against.

3.4 Conceptual Design Summary

The Conceptual Design phase of this year's VTDBF team provided the team a chance to analyze many different design concepts. This analysis allowed the team to select several designs that provided the best all-around aircraft that could be optimized during the Preliminary and Detailed Design phases. The team went through two iterations of analysis to come to the decision it reached. The team selected two wing types, and two landing gear types to further investigate during Preliminary Design in order to find an aircraft design with optimum efficiency in performance and construction.

In conclusion of this conceptual propulsion deign, three engines, the two-bladed propellers, and two Ni-Cad cells were selected for further detailed analysis and study. The three Astroflight electric

engines selected for further study were the CO-40, CO-60, and CO-90. The two Ni-Cad cells were chosen for further analysis. The two bladed propellers were selected for its excellent performance characteristics and lightweight.

4.0 Preliminary Design

The Preliminary Design phase of the 2001/2002 VT DBF team was used to create basic dimensions of the Marooned aircraft. The team completed this design phase by splitting into separate teams, the Wing Team, the Fuselage Team, and the Propulsion Team. This allowed for more design work to be accomplished, and allowed for efficient use of the team's time. Each team was free to determine their course for the Preliminary Design phase, although most of the design phase fell along the lines of empirical and computational analysis. Other analyses like the collection of data, and research of methods to find basic dimensions was completed for each subsystem.

4.1 Subsystem Design

Each subsystem of the design completed research and general analysis of their designs. This allowed them to determine what aspects of the competition were driving factors in their designs. It also allowed the teams a chance to define the best way to analyze their design empirically, computationally, or both. This process helped the team create the optimum plane for the provided specifications.

4.1.1 Wing

The Wing Team began the Preliminary Design phase with a low wing design. The team was required to devise everything else. The team planned to provide a selection of airfoils for the wing, an aspect ratio range, wing span length, and ideas for construction methods. With these areas of interest specified, the team split up in order to conducted their analyses.

The underclassmen on the team were assigned to find airfoils that fit the parameters set by the Wing Team Leader. The driving factor in the initial search was a C_{lmax} of 2.2. Clmax was chosen as a driving factor because it heavily influenced takeoff performance. The underclassmen utilized the online airfoil data collected by Professor Michael Selig (Ref. 1). The upperclassmen were assigned to the evaluation of the wingspan, chord, and AR. This was done because the platform configuration of the design required a moderate understanding of aerodynamics. Most of the underclassmen did not have experience calculating aerodynamic properties. Thus the team leader decided that the best use of their energy would be through the more mundane collection of airfoil data. This exercise required no understanding of aerodynamics and provided an opportunity for them to learn about aerodynamics. The team leader coordinated the underclassmen's research so that a person with experience in aerodynamics could answer their questions and teach them the rudimentaries. With their assignments defined the team began the Preliminary Design of the wing.

4.1.1a Wing Geometry Configuration

The Wing Team began the Preliminary Design phase by splitting into two groups as previously specified. The upperclassmen initiated their configuration design with an analysis using the PrelimSheet code. This and further analyses are covered in the Computational Analysis section (4.3.2). The first analysis with the PrelimSheet showed that research into ARs would help the determination of the wing geometry. During this search an article (Ref. 2) was found showing that the best AR to use for Formula 1 race planes was 8.25. This AR was determined through the Formula 1 racecourse that has a significant portion of climbing and turning flight. Since the AIAA D/B/F course is similar in this respect to the Formula 1 course, this AR provided a good point to design the aircraft around. Thus, an upper limit of AR=9 and a lower limit of AR=8 were selected for the design

The first search of airfoils by the underclassmen only yielded two airfoils, the S1223 and S1210. Because of the slim selection and the high drag of these airfoils the, Wing Team Leader decided to relax the specifications to a C_{lmax} of 1.6. The underclassmen conducted another search through the low Reynolds number data (Ref. 1). This was complemented by a further search by the Wing Team Leader of another online resource (Ref. 1). These searches provided a number of new airfoils that fit within the specification. The team leader decided that the airfoils would be evaluated based on C_{lmax}, C_{dmin}, and the range of AoA that the C_d is within 10% of C_{dmin} . These airfoils and their evaluation variations are presented in Figure 4.1.1.a.1. The maximum of C_{lmax} and the range of Cls, and the minimum C_{dmin} were the criteria determined by the team. These parameters were chosen because of their effect on the flight performance of the aircraft. A high Clmax will require a smaller wing, with a lower weight and fulfill the 200ft takeoff requirement. A low Cdmin will reduce the airplane total CD, which will allow the aircraft to fly at a higher cruise speed, or at a lower engine-power consumption. The range of Cl's for which the airfoil Cd is 10% of Cdmin is a way to quantify the airfoil's overall performance. Thus, an airfoil with a larger range will be able to maintain a lower Cd through various aspects of the flight such as climbing and turning. This means the airplane will be efficient during all phases of the flight, not just straight and level cruise. The team viewed this parameter as the most important, because a large part of the course would be in climbing or turning flight. A further investigation of this selection is provided in the section 4.4.1 FOM Analysis.

4.1.1b Structure

Several methods of construction were presented as options for the design. The team looked into composite covered blue-foam, and wooden spar and ribs with Monokote™ covering. A lot of experience had been gained from the previous year's carbon fiber skinned, blue-foam cored wing. The lessons were carbon fiber's inability to resist shear stress, its heat absorption, and its high expense for construction. The previous year's team benefited from an excess of carbon fiber in its possession. These factors eliminated the use of carbon fiber as the major wing composition. Kevlar was considered because it has

better structural and heat properties than carbon fiber; however, its expense precluded its use. Fiberglass was also considered but rejected because the team felt that it would have a heavier weight for the strength required. Thus, the team settled on wooden construction with Monokote™ covering. The motivating factors for the use of wood and Monokote™ were their lower price, their ease of construction, and their superior finish. Almost all the team members had had experience building with wood, and wood is easily shaped. In addition, wood is very inexpensive compared to other load-bearing materials. Finally, the Monokote™ can provide an exceptional finish without the weight and construction complication of painting and sanding.

With a decision to use wood as a construction material, the team set about on a load structure and material selection. The team looked into two spar structures made of wood. The first was a box-spar design and the second was a two-spar design. The box spar would give the wing good bending moment resistance and torsional rigidity by placing wood in the areas that would make it the most effective, along the top and bottom of the airfoil connected by the shear webs. The box spar would be made to conform to the outline of the airfoil chosen, simplifying construction. The two-spar design would be built with a main spar at the wing quarter chord that would handle mostly the bending moments. There would be a second spar placed at the wing three-quarter chord to take up wing-twisting moments. Thin wood ribs would separate the two spars and define the airfoil shape along the wingspan.

Upon further research, it was found that the thickness of the airfoil would be too small to take advantage of the box-spar design. This is because the two bending-resistant parts of the box spar would not be able to be placed far enough away from each other to resist bending. This left the team with the two-spar design. The team was concerned with the airfoil's continuity across the span with this design. If the ribs were solely covered in Monokote™, there would be sagging of the covering between the ribs, which would reduce the effectiveness of the wing. The team's solution was to cover the wing in balsa and then cover the balsa with Monokote™. A final improvement on the design was to replace the front ribs with a hotwired blue-foam rib affixed to the front of the main spar. This would significantly reduce building time and complexity, but would increase the weight of the wing. The team decided that this increase in weight would be more than compensated for by the ease of construction.

With the two-spar design settled on the Wing Team did a basic moment analysis of the wing to find the required spar thickness to resist bending moments. One of the team members did an analysis of the bending moment on the semi-span. This analysis was done with an assumption that 90% of the semi-span would be able to produce a constant amount of lift, while the outer 10% of the wing would have a linear loss of lift from the constant value to no lift at the tip. This was done to make a simple approximation of the lift across the wingspan while including the 3D lift degradation experienced by wings of finite span. This lift could be easily written as a function of the length along the semi-span, which also meant that it could be easily converted to an equation of the bending moment at any point along the semi-span. This equation for the bending moment was used to calculate the bending moment every 6in of the span during a 6g, 90° bank angle turn. Using the bending moments calculated, the ultimate tensile

strength properties of basswood, and the known thickness of the airfoil at the quarter-chord allowed for a solution of the chord length required of the spar. This resulted in a spar with chordwise lengths as listed in Table 4.1.

Semi-Span Position	Chordwise Length
48	1 3/8
42	1 1/8
36	1
30	7/8
24	3/4
18	5/8
12	3/8
6	1/8
0	1/8

Table 4.1 Initial Size of Wing Spars

4.1.2 Tail

The tail design was assigned to the same group that was working on the wing. This was done because much of the tail sizing is dependent on aspects of the wing. The mission of the Wing Team in this portion was to determine the basic sizing required for the conventional tail. The team started by doing research on common dimensions of conventional tail designs and consulted Raymer's (Ref) suggestions on tail sizing and shaping. These were then combined to find the initial sizing of the tail.

The first assignment for the tail design was to come up with a tail length. This would be the distance from the quarter chord of the wing to the quarter chord of the mean aerodynamic chord of the horizontal tail. A margin of 3.2 was settled on by the team after a review of notes taken in a Stability and Controls course the previous year. This margin was then multiplied by the root chord of the wing to find the length of the tail. This produced a length of 35.2 inches, which was lengthened to 36 inches for maximum safety margin and ease of calculation and measurement.

Next a study of previously designed conventional model aircraft tails was done. This study included the tail from the 97-98 entry of the VT DBF team, which was unfortunately all that was left of this aircraft. Other tails studied were from personal aircraft of the current year's team, as well as figures calculated from model brochures. These studies produced the mean line curve in Table 4.2. A further study was done to find resources that provided guidelines for tail sizing, which resulted in Reference 3. This design was combined with the data in Table 4.2 to produce a sizing of the horizontal and vertical tails. The aspects of these tails are defined in Table 4.2, which provides the AR, root chord (c), and taper ratio (λ). The areas of this design can be found with the data collected in Table 4.2. An initial calculation of the load required on the horizontal tail in cruise was done to find its C_l. The cruise condition was defined as a speed of 60 mph, which produces a tail lift coefficient C_{lt} of –0.332.

	Horizontal Tail	Vertical Tail
AR	5	2
c	0.5	0.04
λ	0.5	0.5

Table 4.2 Definition of tail coefficient values

4.1.3 Fuselage Design

Preliminary fuselage design started with choosing an effective cargo layout. For overall stability of the aircraft, the quarter chord of the wing was decided for the cg (center of gravity) location. To keep flight characteristics as consistent as possible under both laden and unladed configurations, a symmetric ball layout that would be placed in the fuselage such that the cg location did not move was desirable. For bracing, structural stability, and the goal of a fairly conventional aircraft design, it was decided that the internal width of the fuselage should be approximately a foot at its largest section. This limited the ball layout to three softballs in width. Thus, six rows of three were chosen for the main fuselage section. The remaining softballs were placed in rows of two along the longitudinal axis of symmetry. This allowed tapering of the fuselage at the front towards the nose, and tapering of the fuselage section towards the tail earlier than would be possible with say, eight rows of three.

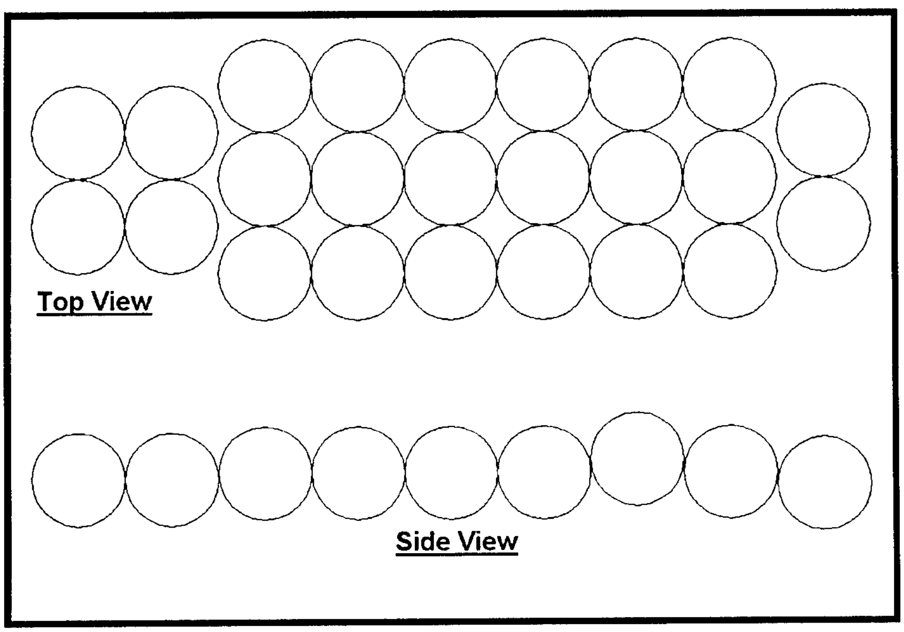


Figure 4.1 Ball Layout

The next step in the design process of the fuselage was to determine bulkhead structure and spacing. For ease of manufacture considerations, a rectangular cross-section for all bulkheads was

chosen, with 1" outer radius fillets. Main bulkheads located at the quarter chord and three-quarter chord were chosen to coincide with the box spar of the wing. This is necessary for mounting of the wing inside the fuselage, and for load transference. The distance between these two bulkheads is 5.5 inches. Similarly, spacing between the other bulkheads was initially estimated at 6 inches. This number was arrived at through consideration of loads, ease of loading and unloading cargo, weight, and finally, the shaping of the aircraft.

General battery placement was next determined. The battery's weight determined that they would have to be placed close to the wing so that they would not affect the CG too much. Thus three placements were considered, directly in front of the wing, between the wing spars, and directly behind the wing. Final decision on the battery placement would be determined during detailed design, when a weights and balances analysis could be better calculated.

Another accomplishment of the Fuselage Team during the Preliminary Design phase was the completion of two tail sections. These tail sections represented two competing structural designs for the fuselage. One design had bulkheads connected by stringers placed in the corners of the bulkhead. The second design was built with bulkheads connected by 4 panels running the length of the tail, with holes cut in the panels to reduce weight. Both designs were covered in Monokote™ because it's easy of use, and superior finish. The tails were tested in bending and torsional stiffness. Few conclusions could be deduced from this data though, since the rigidity of the tails to the test stand was questionable. However the experience provided several valuable lessons. The first was an introduction to building to many of the underclassmen. Secondly, the team reached a conclusion about the structural composition of the fuselage. The team decided a compromise was the best solution, and provided the best weight savings for a given strength. This compromise was the use of keels in the bottom of the bulkheads, along with side panels. This gave the bending strength of the stringers, and the torsional rigidity of the side panels. This compromise marked the conclusion of the fuselage team's preliminary design.

4.1.4 Propulsion

In preliminary propulsion phase three Astroflight engines, two-bladed propellers and two Ni-Cad cells were chosen for further analysis. In analysis of these components, many techniques were used: computer program calculation, wind tunnel tests, static-trust tests, in depended calculation by hand, and VT DBF engine performance research and past experience. The methods used the most were real life simulations in the Virginia Tech Open-Jet wind tunnel, static thrust in the VT DBF Lab, and past and new experience.

4.1.5 Landing Gear

After evaluating the FOM charts, the team decided on conventional landing gear. The team decided that the main gear would be mounted under the wings and the tail wheel would be embedded in the rudder. Mounting the tail wheel in the rudder would reduce drag as well as the amount of linkage

required to operate it. After making this decision, the team had to further narrow the choices to fixed gear or retractable gear. Fixed gear would have a low RAC and would be easy to construct; however, it would also increase drag. Retractable landing gear would have higher RAC and it would be more difficult to incorporate, but it would greatly reduce in flight drag.

4.2 Empirical Analysis

A limited amount of empirical analysis was completed during the Preliminary Design phase. This analysis was used in two areas of design, the Propulsion Team and Fuselage Team. The Propulsion team used empirical analysis to provide better data on propeller performance. The fuselage team used this process to compare two designs for fuselage construction. The empirical design phase helped team members determine real-world performance of their designs, as well as give them a taste of engineering research methods.

4.2.1 Wing Empirical Analysis

As part of the Preliminary Design phase two test-spars were made using the thickness specified in Table 4.1. The first test model was made of laminates of 1/8 in basswood that were 2 feet long. This wood was used because it was the only basswood that could be purchased in a short time frame. This test model was intended for testing the load resistance of the spar design. However, if a load were imposed on the spar, it would respond with a clockwise twist. In the final design this twisting would be counteracted by the ribs and three-quarter spar. For this reason the twist was not viewed as detrimental to the structural concept, but did preclude it's load testing. Good data was gleaned from this spar despite not testing its structural limit. Building the spar gave the team experience with a potential method for building the spar, as well as a better estimate for what the spar would weigh in the Marooned Aircraft. The

The second test-spar was an attempt at the team to decrease the weight of the spar, and try other building materials. The first spar had several limiting factors. The first was that the basswood that fit our specifications could only be readily found in 1in x 1/8in x 2ft pieces, instead of the desired size of 1in x 1/8in x 8ft. The other reason was that the team felt that a lot of the 1lb weight of the spar was due to the lamination process and weight of epoxy. Thus the team tried spruce wood, which is available at most hardware stores. The team found spruce wood in the dimensions of 1½in x ½in x 8ft at one of the local hardware stores. The team then lay out a 1/8in thickness at each tip of the spar and a thickness of 1.375 in at the centerline of the spar. These thicknesses were connected by a linear spar-chord-length taper. The team was then faced with the challenge of ripping off ½in from the height of the spar, and cutting the angled spar taper, all with a band saw since that, and a scroll saw were the only power saws available to the team at the time. From this process three things were gleaned. That the team's band saw, no matter the patience, doesn't make straight cuts longer than 6in. A linearly tapered spar shouldn't be considered. Finally, access to a table saw had to found before final construction was initiated. At the end of

constructing the spruce spar, it was found to weigh less than the laminate spar at 0.5lbs for the a half-span spar. Despite the weight advantage of the linear tapered spar, the laminated spar has much easier building requirements, and lends itself to vertical alignment of the ribs.

4.2.2 Propeller Analysis

Numerous propeller analyses and static thrust tests were performed to gather data and make propeller selection. Six different propellers were tested in the wind tunnel. The data is shown in Table 4.3.

Prop	Kv	Kt	Rm	Io	I in Current, I	Vin Voltage, V	I max eff Current	Speed RPM	Torque Torque	Trust
D X P	(rpm/V)	(torque/I)	(ohms)	(amps)	(amps)	(volts)	(amps)	(rev/min)	(in-oz)	(lbs)
22 x 12	347.0	3.610	0.103	2.5	22.47	31	27.43	3910	81.1	7.6
20 x 8-14	347.0	3.610	0.103	2.5	20.48	31.5	27.65	3973	73.9	5.3
16 x 6	347.0	3.610	0.103	2.5	9.74	36.3	29.68	4579	35.2	2.6
15 x 8	347.0	3.610	0.103	2.5	9.94	35.5	29.35	4478	35.9	2.8
11 x 5	347.0	3.610	0.103	2.5	6.9	37.3	30.09	4705	24.9	1.3

Table 4.3 Propeller Test Data

The only propeller that was able to give required 7.5 lbs of thrust was 22x12. The rest of the propellers were able to give desired thrust, but more voltage is required for the engine to rotate at a higher speed and that would increase the battery cell count past 32 cells. Increasing propeller rotation speed will increase torque and torque is directly proportional to battery cell capacity. Any propeller diameter larger than 22 in at a pitch higher than 2 degrees is going to give needed thrust at relative moderate propeller speed.

4.3 Computational Analysis

Computational analysis was required to investigate the designs passed to the Preliminary Design phase. This analysis was done for several aspects of the aircraft including the engine, propeller, and wing. Engine and battery performance was modeled by theoretical computations using Excel. The upperclassmen in the Wing Team utilized two team-developed Excel codes to accomplish this analysis. These Excel codes were named "PrelimSheet" and "TakeOff." The use of all three of these codes provided significant help in the defining of the Preliminary Design of the Team Marooned Aircraft.

4.3.1 Engine Analysis

The three engines that were selected in conceptual design were Astro-Flight's CO-45, CO-61 and CO-90. The direct drive (D), the geared drive (G), and the super box drive (S) were all considered as modifications of each motor. The direct drive system has no gearbox and the propeller is attached directly to the engine shaft. The geared drive has a 2.75 to 1 gear ratio between the engine shaft and the

propeller. The super box has a 3.3 to one gear ratio between the engine shaft and the propeller. One of the major problems that were faced was the cost and time to get the engines. The engine used last year was Astroflight CO-61. The other two engines, CO-45 and CO-90, were not available to the team for test and analysis. To cut down on cost and save time specific strategy was developed based on numerous static-thrust tests, real life simulations and engine specification data, such as voltage/current constants, resistance, and weight to indirectly analyze the engines that were missing. The following strategy was developed.

Based on the 200 foot required takeoff distance, it was determine that 7.5 lbs of thrust was necessary to accomplish a successful takeoff with a heavy payload. This calculation of needed thrust of 7.5lbs with other parameters and tests helped to narrow engine selection.

Real-life simulations in the wind tunnel and static-thrust tests were performed with the CO-61 engine powered with last year's battery pack that had 32 2300mAh Ni-Cad cells connected in series with a nominal voltage of 38.4 volts. Data from these tests was taken and adjusted to the other engines. One set of the data taken during the test is shown in Table 4.5.1.

							Fixed		Fixed			
Engine					Gear	RPMsh	**RPMp**	Qsh	**Qp**	I in	Vin	**Thrust**
	Kv	Kt	Rm	Io	Ratio			Torque	**Torque**	Current	Voltage	
Units=	(rpm/V)	(torque/A)	(ohms)	(amps)		(rev/min)	**(rev/min)**	(in-oz)	**(in-oz)**	(amps)	(volts)	**(lbs)**
CO-61G	347.0	3.610	0.103	2.5	2.75	10757	**3912**	81.12	**223.2**	22.47	31.00	**7.70**
CO-61D	347.0	3.610	0.103	2.5	1.00	3912	**3912**	223.20	**223.2**	61.83	11.27	**7.70**
CO-40D	682.0	1.987	0.121	2.0	1.00	3912	**3912**	223.20	**223.2**	112.33	5.74	**7.70**
CO-40G	682.0	1.987	0.121	2.0	1.68	6572	**3912**	132.86	**223.2**	66.86	16.19	**7.70**
CO-40S	682.0	1.987	0.121	2.0	3.30	12910	**3912**	67.64	**223.2**	34.04	62.47	**7.70**
CO-90D	230.0	5.890	0.155	2.5	1.00	3912	**3912**	223.20	**223.2**	37.89	17.01	**7.70**
CO-90G	230.0	5.890	0.155	2.5	2.75	10758	**3912**	81.16	**223.2**	13.78	128.63	**7.70**

Table 4.4 Data from engine test and adjusted data for find other two engines

To have 7.5 lbs of thrust with a certain propeller (22x12), the propeller's revolutions per minute (RPM) had to be 3912 RPM and the torque required to rotate the same propeller at that speed was 223.2 in-oz. This valuable information helped narrow engine selection. As shown in Table 4.4, engines CO-61D, CO-40D, and CO-40G were eliminated by current requirements of 40 amps. The thrust requirement was increased to 7.7 lbs to lower the measurement and human error that might occur.

4.3.2 Battery Analysis

Two Ni-Cad battery cells were considered for further analysis. The cells came from different manufacturers. The capacity of both cells is 2400 mAh and dimensions were the same. The only difference was the weight; one cell was slightly lighter than the other. The difference in weight would add one extra cell. To solve the problem, the manufacturer's history was taken into account. The final Ni-Cad

battery selection was the 2300-2400 mAh Sanyo cell with a nominal voltage of 1.2 volts, weight of 58 grams (2.05 oz) and a size 4/5 that of a C-size cell.

With this battery specification, the max cell count in 5 lbs is 39. This cell configuration does not take in to consideration the material needed to put battery cells together and shrink-wrap. Approximating with Ni-Cad Battery packs that are available an adjusted single cell weight is 2.14 oz. This adjusted weight was used to determine a new cell count of 37. With this cell count, the battery pack would weigh 4.95 lbs. To take in to account that the adjusted cell weight is off and that every scale will read a different weight the final cell count was dropped to 36 cells. This final battery pack specifications are 36 cells connected in series with a voltage of 43.2 volts, and a battery pack weight of 4.815 lbs.

4.3 Computational Analysis

The spreadsheet PrelimSheet was based on an Excel code written the previous year. It was modified to fit this year's competition rules and to provide more conservative estimates of performance and cost. The spreadsheet accepts as inputs the configuration geometry, the mission specifications, and the Oswald Efficiency Factor "E." The total drag is then calculated by assuming a parabolic drag polar. The drag polar data is utilized with a specified payload weight to find the thrust required for steady straight and level flight. With the flight speed given in the mission specifications, the power required was also computed from the total drag.

The battery energy required for a single lap was found by using the drag in cruise conditions and multiplying it by the distance flown for a single lap, divided by the efficiencies in the propulsion system. The distance was estimated for a constant radius turn, and the efficiencies were constants entered by the programmer. A cycle was defined as two laps with payload and four laps without payload. This cycle was used to simplify the programming of the file and to overestimate the performance required by including the 360º opposite turn on the final two laps. The energy required per cycle is found by summing the energies over the parts of the cycle, with passengers and without passengers. The total battery energy was computed from the cell voltage and from the number of cells in the flight pack. With the total battery energy and the energy required for each cycle, the range in cycles was found by dividing the former by the latter. The range was helpful in defining the efficiency of each design.

The first consideration when using the code was to understand its limitations and estimations. The first of these was its inability to predict stability, a major reason in utilizing it more in Preliminary Design, compared to the previous year's use of the sheet in Conceptual Design. The code was written to be "a quick means of evaluating concepts, "(Ref. 4)" which meant it was highly idealized. The idealization had a detrimental effect on the previous year's design. So certain inputs were modified to reflect a more "real world" case. This modification included altering the planform specific weights of the fuselage, wing, and tail. The numbers used were taken from the previous year's design, which was acknowledged by the team to be excessively heavy.

Further understanding of the code's limitations was observed during the analysis of AR. The PrelimSheet code represented linear aerodynamic modeling, which states that an infinite AR provides the best efficiency. This condition is due to a simplification of straight and level flight like that of a glider. However it does not include climbing and turning flight. A look at the course, based on a constant 50 ft turning radius, showed that 33.42% of a two lap portion of the mission would be climbing or turning flight. This is a significant portion of the mission comprising climbing and turning flight. This analysis of the course and the limitations of the PrelimSheet code lead to the previously mentioned research into AR.

The team used the defined range of AR in this analysis served as a constraint for this analysis, and the suggested AR of 8.25 was used for any preliminary calculations. With this AR and an estimated static thrust of 7.5 hp the team used the PrelimSheet code to find a worse case scenario Gross Take Off Weight (GTOW) for the aircraft. In the first analysis the GTOW and static thrust were entered into the "TakeOff" code as a Thrust to Weight (T/W) ratio. The code provided a wing loading (W/S) necessary to fulfill the specified 200 ft. takeoff requirement based on a curve of the T/S required to takeoff. The 8.25 AR was plugged into the TakeOff code with a C_{lmax} of 2.2, and a takeoff weight of 27 lbs. This flight condition related to takeoff with payload and is the GTOW constraint. This corresponded to a T/W of 0.2778. This initial analysis was completed after the initial survey of airfoils that determined that no suitable airfoils were available with a C_{lmax} of 2.2. The TakeOff code was run for the same T/W with the new relaxed to a C_{lmax} of 1.6. This corresponded to a W/S of 4. Using the Formula 1 specified AR of 8.25 a wingspan of 7.75 ft and a wing chord of 10.9 in was found.

Another feature of the TakeOff code was it ability to take the inputs of Clmax, AR, b, and GTOW and calculate the distance to takeoff. This analysis showed that the dimensions provided by the initial analysis provided a small margin between the takeoff distance and the 200ft constraint. In addition the team was concerned about building the odd dimensions derived in the first analysis. The team decided to use the second feature of the TakeOff to refine the dimensions of the wing. This resulted in a span of 8ft and a chord of11in. This accomplished the team's goals to simplify construction and increase the margin for taking off in 200 ft. After this second analysis the new AR was calculated and found to be 8.727, fitting within the predefined limits of AR. The TakeOff code used the assumption that the 2D C_l of the airfoil could be produced throughout the wingspan. Because of this the team decided to leave the decision whether to include flaps or not until the detailed design.

4.4 RAC and Figures of Merit

During the Design phase FOMs were used to choose an airfoil, while is a method to convert qualitative values into quantitative values, used in this case to choose a wing airfoil. In addition FOMs were used to evaluate the Preliminary Design and help solidify the final configuration based on design features and mission requirements. Finally the RAC of the aircraft at the end of the Preliminary Design phase was compared to the RAC from the Conceptual phase to show the progress of the team.

4.4.1 Airfoil Figures of Merit

The airfoils were compared with respect to three FOMs, C_{lmax} at Re 150,000, C_{dmin} at Re 300,000, and C_l range at Re 300,000. These merits were chosen because of their effect on the overall aircraft performance. A high C_{lmax} at Re 150,000 would give a shorter takeoff distance, and this was an approximate Re number the wing would be experiencing at takeoff. The C_{dmin} was needed to be a minimum so that during cruise portions of the flight the aircraft would have a low drag. The C_l range that the C_d was within 10% of C_{dmin} was a measure of the airfoil's overall performance. A further explanation of this is that in the RC world it can be difficult to accurately measure the aircraft's angle of attack, and the aircraft would be in climbing or turning flight a significant portion of the competition. Therefore, airfoils with a small band of low C_d would hurt the aircraft's performance in anything but ideal conditions. The weighting of these factors can be seen in Table 4.5. The C_{lmax} weighting factor was is small because advances from the Propulsion Team were lessening the requirements of an extremely high C_{lmax}. As can be seen from Table 4.5, this led to the selection of the SG 6043 airfoil. This airfoil has a C_{lmax} of 1.6, a C_{dmin} of 0.0109 in cruise, and a C_l range of 0.84, this computes to the highest FOM score of these airfoils.

Airfoils				
Multipliers	1	2	5	
	CL max	CD min	Range	Total FOM
S1210	1.816	0.0116	0.4	3.8392
SG6043	1.538	0.0106	0.84	5.7592
FX 63-137	1.738	0.0148	0.5	4.2676
S1223	1.97	0.015	0.357	3.785
FX76MP140	1.725	0.21	0.4	4.145
FX74CL5140MOD	1.85	0.014	0.35	3.628
FX76MP160SM	1.6	0.014	0.35	3.378
S2091	1.4	0.014	0.45	3.678

Table 4.5 Airfoil FOMs

4.4.2 Design Parameter FOMs

Several design parameters were investigated during the detailed design phase including L/D, T/O, P/W, construction ability, and RAC. The parameter L/D influenced two design features the airfoil selection, and the fuselage nose and tail design. It was important to obtain a high L/D with a small drag is that the resulting aircraft would quickly complete the mission. The parameter T/O was the takeoff length required of the aircraft and primarily influenced the length of the wing. This parameter provided the maximum required W/S of 3.6818 lb/ft^2 to complete the mission. The motor selection was driven by the P/W, power-to-weight ratio, design parameter. During this phase it was found that a minimum power

loading (P/W) of 0.2593 was required to complete the mission successfully. The construction ability of any piece factored heavily into all the designs, since the team's facilities were limited. Thus the easiest and fastest built components were the highest scored. Finally the RAC parameter affected all the design features the same. All the design parameters had small influences on all the design features. The values of W/S and P/W required were based on the assumption that the aircraft would weigh a maximum of 27lbs at GTOW, that 7lbs of thrust could be produced at the second takeoff, and that the constraint diagram provided the correct W/S. The maximum GTOW of 27 is valid because of the heavy area-specific weights used for this analysis. The 7lbs of thrust available was justified by tests conducted by the propeller team during the Preliminary Design phase. The TakeOff code is validated based on it's use in the previous year, where the aircraft did takeoff within the require field length. These assumptions and FOMs helped to define the design produced during the Preliminary Design phase.

The use of these FOMs throughout the preliminary design resulted in a final Preliminary Design. This design is distinguished from the Conceptual Design by several features. The Preliminary Design has a wing span of 8ft with an AR of 8.7273 and flaps. This design has taildragger gear and a conventional tail. The fuselage has a maximum ball abreast of 3, and isn't blended into the wing. The ball layout is determined so that the CG doesn't shift when all the passengers are added or removed. A single engine and single propeller propulsion system also sets this design apart from the Conceptual Design.

4.4.3 Preliminary Design RAC comparison

The RAC for the Preliminary Design was computed to compare it with that of the Conceptual Design RAC. In doing this a flaw in the PrelimSheet code was found involving the 10hr/ft cost of the fuselage, which was incorrectly listed as 5hr/ft. Several assumptions were used in calculating this RAC, besides the costs outlined by the rules. This included the use of flaps, with 1 servo for both flaps, and 1 servo for both ailerons, as well as 1 servo a piece for the horizontal and vertical tails. Other assumptions were the Wing Area Specific Weight, and Fuselage Wetted-Area Specific Weight (Table 4.6). The same assumed Tail Area Specific Weight was used for both Conceptual and Preliminary Designs. With the PrelimSheet code fixed, the RAC at the end of the Preliminary Design phase was calculated based on the parameters specified in Table 4.6. This RAC as can be seen from the table was found to be 12.82076, compared to the 12.511 RAC from the Preliminary Design. This is an unfair comparison since the Conceptual Design RAC was calculated incorrectly, and would actually be 13.3869. Thus the Preliminary Design provided a lot of optimization of the Marooned Aircraft Design.

Preliminary Design RAC	
Wing	
Span	8ft
AR	8.7273
Area Specific Weight	0.4lb/ft2
Number of Servos	6
Control Surfaces	4
Fuselage	
total length	6.40833ft
Wetted-area Specific Weight	0.0032
Horizontal Tail	
Span	28in
Area	1.1458ft2
Vertical Tail	
Span	15.1789in
Area	0.8ft2
Engine	CO-60G
Weight of Motor	1.4375
Battery Weight	4.815
RAC	12.82076

Table 4.6 Preliminary Aircraft RAC

4.5 Summary

During the Preliminary Design phase several things were accomplished. The VT DBF team found a maximum W/S, and minimum P/W required to complete the mission. This was converted into several design studies, including the wingspan. The Wing Team did further analyses that produced the best airfoil to use in their 8ft wing. The fuselage team decided on a ball configuration and general layout of the aircraft. The fuselage team also built several test sections that provided insight into the best way to build the fuselage structure, as well as the fuselage's wetted-area-specific weights. The propulsion team used the Preliminary Design phase to investigate all aspects of their design.

Propeller selection was limited to propellers with a higher diameter than 22 in. and a larger pitch than 12 degrees. With a battery specification of 43.2 volts, all engines were eliminated but the CO-61G and CO-90D engines. At this point three out of four requirements from the conceptual design phase were met. Those two engines with different propellers were further tested, compared, and analyzed in the Detailed Design phase to achieve the best possible propulsion configurations and to meet the last requirement of ten minutes or less for total mission time.

5.0 Detailed Design

The Detailed Design phase of the 2001/2002 VT DBF project required an optimization of the Preliminary Design first. Once this optimization was completed aspects of the design needed to be computed. These computations included weight and balancing of the aircraft, performance of the aircraft, handling qualities of the aircraft, and finally the RAC for the final design. These analysis and computations would provide the team the necessary information to build the Marooned aircraft.

5.1 Subsystem Design

Each of the Marooned aircraft's subsystems had to be investigated in further detail by the VT DBF team to produce an optimally designed aircraft. This optimization would allow for an accurate calculation of the plane's weight and balance, performance, and handling characteristics.

5.1.1 Detailed Wing Design

During the Preliminary Design phase the Propulsion Team had selected a battery and propeller combination that provided more thrust than that originally estimated by the teams in the Conceptual and beginning of the Preliminary phases. This increase in thrust allowed the Wing Team to eliminate the flap surfaces on the wing. This reduced the cost of the aircraft by the reduction of the number of control surfaces and weight of servos, and simplified the construction of the wing. The wingspan was kept the same length in order to retain some of its advantages. The longer wing allow for a wide margin, so that if the plane turns out to be heavier than expected, it will still be able to takeoff in the field length provided. Another added benefit was found using the PrelimFile code, that showed that the 8ft span, 11in chord wing had very similar L/D's for both passenger and ferry flight regimes. This was viewed as very favorable since this would provide the pilot that it flew similarly in both conditions, requiring less training and adapting. In an effort to improve the efficiency of this wing the team decided to employ the use of Hoerner wing tips. These tips are a 30deg upward thickness taper at each wing tip. This wing tip would help recover some of the 3D losses of a wing without incurring a vertical surface cost like a winglet would.

One aspect of the tail that hadn't been researched was airfoil selection. The team considered two tail airfoil concepts. The first was to use a symmetric airfoil, the second a cambered airfoil inverted to provide a downward force at the tail. The SG 6043 airfoil has a nose-up pitching moment, which in cruise would require the symmetric airfoil to fly with a deflected elevator to provide the necessary download.. Minimum drag was one of the Wing Team leader's main concerns, so he instructed the underclassmen to re-search Reference 1 to find the airfoil with least amount of drag at a Re of 300,000, and C_l of 0.331. The search produced the RG-14 airfoil, which at the specified conditions has a C_d of 0.0067. Analysis was then completed for the pitch stability of this configuration. The team found that computing the stability derivatives were very difficult because of keeping the sign conventions correct. An additional concern was that the horizontal tail would stall in nose down maneuvers, and that at high angles of attack the tail would not produce enough force to maintain the required angle of attack. While, with the negative

pitching moment of the wing it would produce a stable aircraft, the team decided it was not an optimal solution. It was also found from interpolating data from Reference 1 and Reference 5 that the drag produced by a NACA 0012 could be the same at a -3deg angle of incidence would be about the same as the drag from the RG-14 airfoil. Thus, the team decided that for the ease of stability calculation, and no drag penalty that the NACA 0012 airfoil would be used for the horizontal tail airfoil.

The vertical tail provided considerably more investigation due to its integration with the tail. The Wing Team Leader wanted to have a smooth transition from the sides of the fuselage tail into the vertical tail and rudder. This would minimize intersections, which would reduce the complexity of construction and the profile drag related to interference drag. The interference drag is of considerable importance since it rises by the square of C_l (Ref 2), just as induced drag does. The team knew that the airfoil would be a symmetric NACA airfoil, but the thickness determination was hampered by the fuselage width at the tail. In order to maintain a 10° slope along the sides of the retracting fuselage, the fuselage would come to a 1.25in width at the rudder. This would require a NACA 0033 airfoil of the vertical tail to provide the desired fuselage/tail integration. This was not good alternative, since the extra thickness of the vertical tail would probably more than compensate for the interference drag of the discontinuous fuselage/tail intersection. A compromise was reached by having a NACA 0012 airfoil for the stabilizer portion of the vertical tail, and allowing the fuselage to maintain its 10° slope. The Rudder would attempt to smooth the flow by having a thickness of 33% behind the fuselage, tapering to a thickness of 13% at the tip of the vertical tail.

c/4 spar chord lengths		
Span	Wood	CF
48	3/8	3
36	3/8	2
30	3/8	2
24	3/8	1
18	1/4	1
12	1/8	1
06	1/8	1
3c/4 Spar chord lengths		
Span	Wood	FG
48	3/8	3
36	3/8	2
30	3/8	2
24	3/8	1
18	1/4	1
12	1/8	1
06	1/8	1

Table 5.1 Final Spar Configurations

The last thing for the Wing Team to accomplish was a reanalysis of the wing spar width. Using the bending analysis of a beam of multiple materials (Ref. 7) helped the team write an Excel code named SparCalc that found the optimum thickness of the spruce spar, and optimal placement of a composite strengthener every 6in along the wingspan. The code used an assumption of 1/32in thickness for each layer of composite lay-ups. This analysis resulted in a spar that weighed 0.546 lbs without the weight reduction lightening holes would provide. This is less than the 1.0lbs and 0.9lbs for the two spars built during the preliminary design. A second beam analysis was done with SparCalc for the aft wing spar assuming that it would carry 1/4 of the bending loads of the wing. Fiberglass was used in this analysis because of its resistance to more than just tensile loads. This analysis resulted in the spar designs represented in Table 5.1. The spars outlined in this figure can handle a 7g turn without exceeding the maximum stress of the spar materials. A minimum safety factor of 1.053 is observed in these designs, at a point outside where the landing gear will be mounted. Thus, if the wing should fail the plane could still land. The team also knew that the balsa and Monokote™ sheeting would provide an additional safety factor to the design, which was adequate to the team. This if further bolstered by the fact that the team does not plan on flying in competition above 4gs. The overbuilt nature of the design is to allow for unexpected flight conditions that the Marooned aircraft could experience. Using a an experimentally calculated value of the area-specific weight of Monokote™ (1.1489*10-4 lb/in2) and a balsa density of 0.00081 lb/in2 a full wing weight was calculated to be 1.318lbs. This produces a planform area-specific weight of 0.1797 lb/ft2 in comparison to the previous year's planform area-specific weight of 0.464 lb/ft2.

5.1.2 Detailed Fuselage Design

Next, specific placement of the batteries was determined. With the engine placement determined from smooth tapering of nose bulkheads, rough moment calculations were made to determine specific battery placement, as the battery pack is the heaviest component of the aircraft. The following components were considered in this calculation: engine and gearbox, propeller, nose section, wing, batteries, and tail section. The weight of the main fuselage section was not considered, as its relative CG location was calculated to align with the quarter chord. Weight of servos and landing gear were not considered. Pitching moments about the quarter chord were taken as follows.

Forward of the quarter chord, the propeller, engine, and nose section contribute to the moment. The propeller weighs 0.125 lbs, and is 24.0" forward from the quarter chord. The engine/gearbox combo weighs 2 lbs and its relative CG is 22.2" from the quarter chord. The nose section is estimated as weighing 0.5 lbs, and it's estimated relative CG is 16.0" from the quarter chord. This gives a nose-down pitching moment of 55.4 lb-in.

Aft of the quarter chord, the wing, batteries, and tail section contribute to the moment. The wing is estimated to weigh 3 lbs, and its relative CG location is estimated to be 1.1" aft of the quarter chord. The batteries weigh 5 lbs, and for this calculation, were placed 7.3" from the quarter chord. The tail

section is estimated to weigh 0.9 lbs, and its relative CG is estimated to be 21.4" from the quarter chord. This yields a nose-up pitching moment of 59.0 lb-in.

The main uncertainties in the above calculations are in the weight of the wing, nose section, tail section, as well as their estimated CG locations. The error in the weight of these components is estimated to be within 10%. The error in CG locations of these components is estimated to be within 20%.

To allow for placement of two longitudinal keels and to fit the battery pack between two bulkheads, the 36 batteries were arranged in a 9 x 4 rectangular configuration. In this configuration, the battery pack can be shifted approximately 1" between bulkheads to adjust CG location.

Four of the bulkheads for the main fuselage section are identical in shape, and allow for three softballs to fit between the cutouts. As the bulkheads are 6" high, with a 1" fillet on the sides, side panels for the main fuselage section are 4" thick, to allow good epoxy bonding with the bulkheads. Holes are cut in the side panels between bulkheads to decrease weight.

Two longitudinal keels are placed outside the battery pack, on the underside of the bulkheads, which are notched to allow the keels to be mounted flush. The keels are not continuous – there is a break where the wing fits into the fuselage. The wing will be removable, and have a box-spar that runs the span of the wing for structural and stability reasons. The middle section of the wing, which will fit flush to the fuselage, will supply all of the support continuous keels would provide, if not more.

The top of the fuselage is covered with a very thin layer of Monokote™ covered balsa, with Velcro strips running underneath along the edges. This is easy to build, makes the cargo bay easily accessible, and will securely hold to the fuselage when covered.

The nose section also has side panels that are bent along the curvature of the nose bulkheads. Concerns about the amount of torque on the nose structure by the engine were alleviated by adding X-bracing to the top and bottom of the nose section, using thin strips of carbon-fiber reinforced spruce ply. Slits and holes were added near the tip of the nose to provide airflow to the engine, allowing it to run cooler.

The tail section of the fuselage is designed with a 10° taper on the sides, as it was desired to maintain no higher than a 10° taper angle on the tail side taper and top taper, to keep flow attached to the body. This was not implemented in the nose section, as propeller wake would be the dominant flow feature in this area. As in the nose section, side panels and reinforced X-bracing is used on the tail section to give longitudinal and torsional strength and stiffness.

To assist in visualization and detailed design, AutoCAD Mechanical Desktop 6 (Educational version) was used. The majority of the structural components were drafted to precision to ensure that placement of various components would not pose a problem. In addition, Matlab was used extensively as a computational tool for automatically generating bulkhead sizes, tapering angles, optimum location for lightening holes, and other mathematically intense calculations that would need to be done repeatedly as the plane configuration evolved over time.

5.1.4 Propulsion

In Detailed Design final decision are made on engine and propeller selection. Two competing engines, Astroflight CO-61G and CO90D, were analyzed in the final detailed design for the best results. Propeller selection was much harder to do. A lack of desired propellers to test and we were not able to experimentally compare the propellers; therefore all of the decisions had to be based on mathematical modeling, independent calculation, and MotoCalc Program. The objective of the Detailed Design Phase was to determine a configuration that would give the best flight time. At this point in the design, a total flight time of 10 minutes was not acceptable. As aircraft and propulsion design evolve, it was found that total flight time could be cut to 5 min or less.

In order to compare the propellers, a 22x12 propeller was chosen as the lower limit when looking at the thrust. This propeller was tested in the wind tunnel. The total drag was estimated for the aircraft at airspeed of 50 mph was 1.5lbs. The total flight path is a function of the radius. If a 75-foot radius is assumed, the total flight path would be 3.165 miles. This data was used to simulate real flight in Virginia Tech's Open-Jet Wind Tunnel using CO-61 motor and a 32-cell battery pack. The following table and graphs represent the data taken during the wind tunnel test.

Extra Thrust, T	Air Speed	Path Flown	Path Flown	Time
(lb)	mph	Feet	miles	(min)
1.5	47.17	2075.48	0.39	0.5
1.5	47.17	4150.96	0.79	1
1.5	47.17	6226.44	1.18	1.5
1.5	47.17	8301.92	1.57	2
1.5	47.17	10377.4	1.97	2.5
1.5	47.17	12452.88	2.36	3
1.3	47.17	14528.36	2.75	3.5
1.2	47.17	16603.84	3.14	4
1.2	47.17	18679.32	3.54	4.5
1.1	47.17	20754.8	3.93	5
0.9	47.17	22830.28	4.32	5.5
-0.1	47.17	24905.76	4.72	6
0.1	47.17	26981.24	5.11	6.5

Table 5.2 Wind Tunnel Data at 47.74 mph

According to the Table 5.2 flying with 22x12 propeller and averaging air speed of 47.17 mph, with 50 foot turn radius total engine run time is about 3.5 min, 75 foot turn radius engine total run time is 4.0 min and with 100 foot radius total engine run time is 4.5 min. Increasing airspeed and decreasing turn radius

lowers the flight time, which is desired. In the worse case scenario with 100 foot radios there would be about 2 minutes extra time left after the mission is over. To get best results propulsion system had to be pushed to its limits.

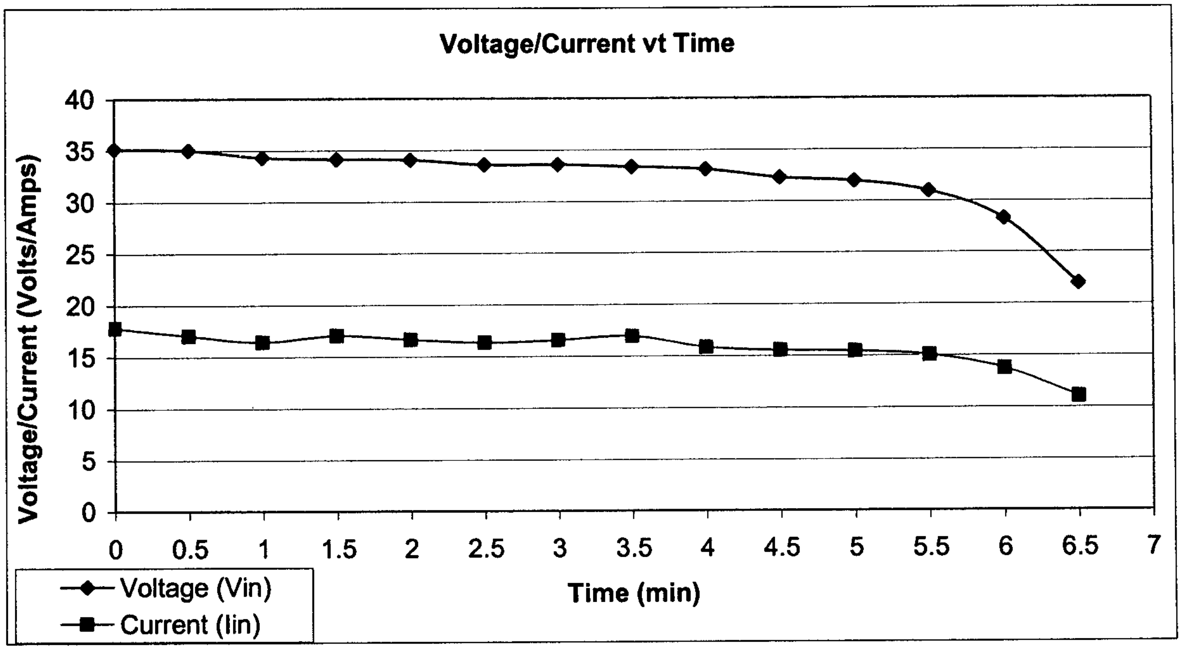


Figure 5.1 Voltage/Current vs. Time History of wind tunnel test

When voltage and current is tanked into account it was determined that slightly bigger propeller was be optimum solution. At this pint diameter of propeller was set to 22 inches. The reason was that this propeller is capable of producing more then enough of thrust that is required for 200-foot take-off distance. Increasing diameter of the propeller increases thrust and drug beyond the point without any benefit for performance based on final and optimum battery pack. Increasing diameters pitch increases drug but also increased air speed. Increasing the pitch requires higher torque and that will be draw higher currents

When the data taken with 32-cell battery pack is adjusted to fit 36-cell pack significant theoretical increase in thrust is notices. The propeller RPM increased by 606 PRMs and that constituted of 36% of increase of thrust compare to 32-cell pack. The initial trust was 7.7lbs and new battery pack adjusted thrust is 10.5 lbs. The static thrust-test with new battery was performed to confirm the theoretical prediction.

As expected from the theoretical conclusion, thrust increased significantly when the static-thrust test was performed with new battery pack and 22x12 propeller. The max thrust that could be produced with old battery pack was 9lb. This is two pounds more then required for aircraft to take off with a heavy payload within 200 foot take off distance. With new battery pack, as shown in Figures 5.2 and 5.3, the max thrust is about 12lb.

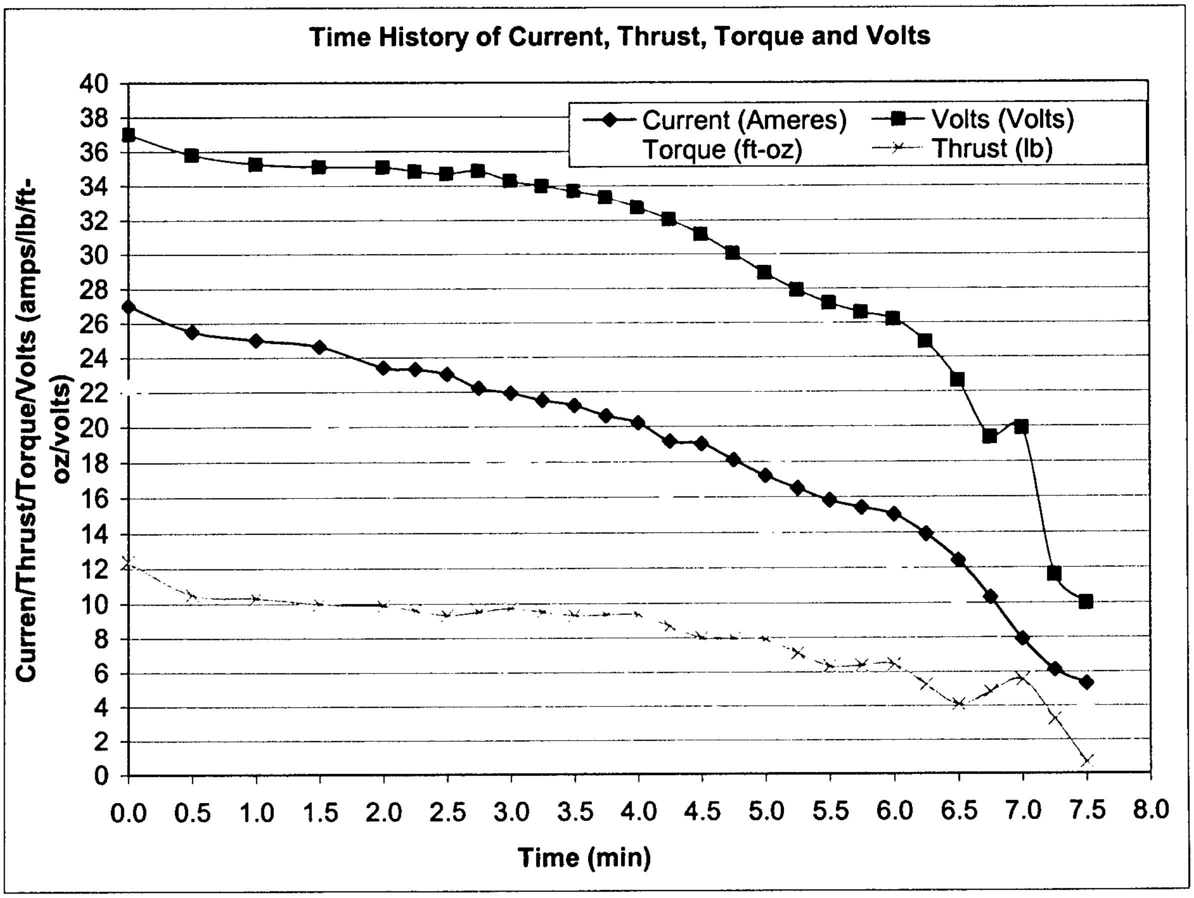


Figure 5.2 Time history of current, thrust, torque and volts

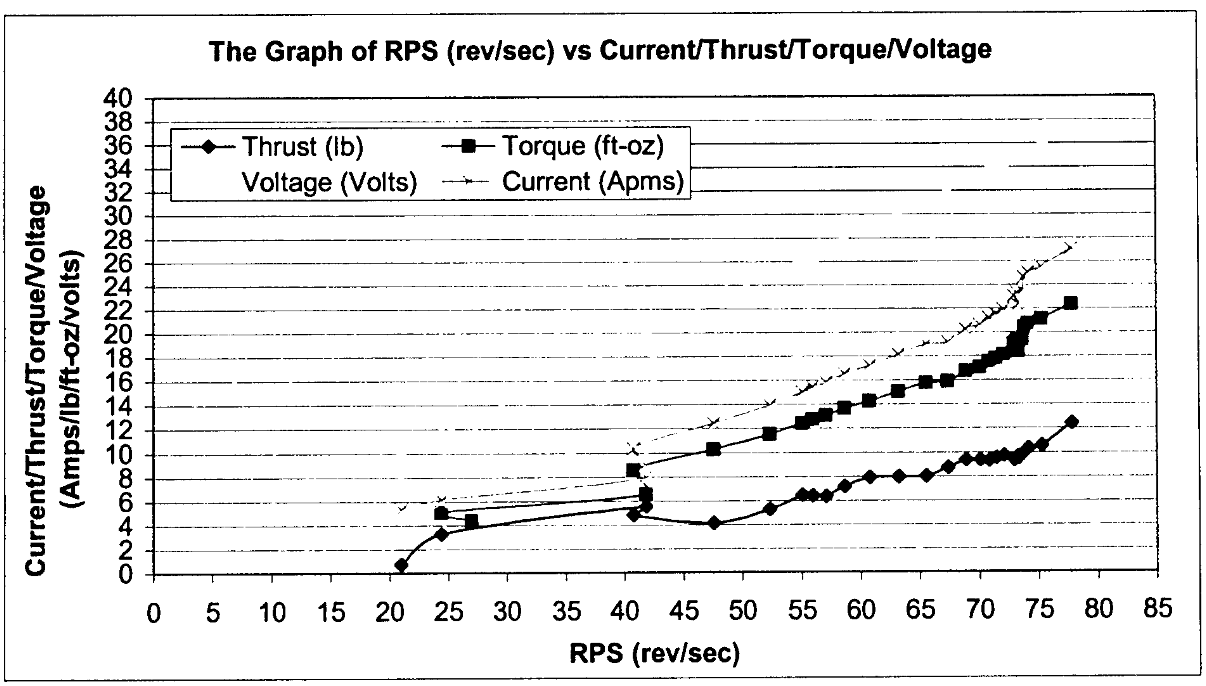


Figure 5.3 The graph of PRS (rev/sec) vs current/thrust/torque/voltage

This significant increase in thrust requires higher battery currents because thrust is directly concocted with RPM, voltage and torque. The technique used to eliminate this CO-90D is the same as strategy that was implemented in Preliminary Design where RPM and torque is kept constant to see how many amps is required for CO-90D to achieve propulsive results. The data is shown in Table 5.2.1. This change in amps draw eliminates CO-90D engine, the current would go over 40 amps limit.

							Fixed		Fixed		
Prop DXP	Kv	Kt	Rm	Io	Gear Ratio	RPMsh	RPMp	Qsh Torque	Qp Torque	I in Current, I	Vin Voltage, V
22 x 12	rpm/V	torque/I	ohms	amps		rev/min	rev/min	in-oz	in-oz	amps	volts
CO-60G	347.0	3.610	0.103	2.5	2.75	10757	4446	81.12	268.0	22.47	31.00
CO-90D	230.0	5.890	0.155	2.5	1.00	4446	4446	268.00	268.0	45.50	19.33

Table 5.3 Final motor selection

The propeller pitch selections needed some more research and calculations. The MotoCalc program was used to determine the propeller's pitch. The pitch was varied from 12 inches to 24 inches but diameter of 22 inches was kept constant. The data computed for a 22x12 propeller was compared with experiment data from wind tunnel tests to see how accrete the MotoCalc program was. MotoCalc data and experimental data seemed to be in agreement less then 5% of error. The thrust and flight time was found to be almost exactly the same as experiment values. This concluded that the MotoCalc program is a good way to finalize propeller selection.

Time is critical factor in this year's design. In the Preliminary designed it was concluded that total engine run time should be equal or less then five minutes. In this selection the top velocity that could be achieved with propeller was important factor in determining the pitch. Pitch is varied until torque required turning propeller at certain RPM was high enough to drain batteries in about five minutes.

The best two solutions that were computed with MotoCalc were: 22x16 propeller with CO-61G motor and 36 Ni-Cad cells, and 22x18 propeller with CO-61G motor and 36 Ni-Cad cells. The graph and characteristics propeller/engine/battery optimization are shown in Figures 5.2 and 5.3.

These propeller combinations resulted the best flight score possible for. The engine run time is decreased to its minimum with approximately a minute extra time to fly. This will allow the best flight score possible. In addition, propulsions system was set up so that a pilot has total control of how fast batty are drained. It is known that flying full throttle one will have enough power to finish the mission and that fliyng with ¾ of throttle will give enough power to fly decent at air speed. This allows pilot to use that full throttle when necessary to maneuvers and short take-offs.

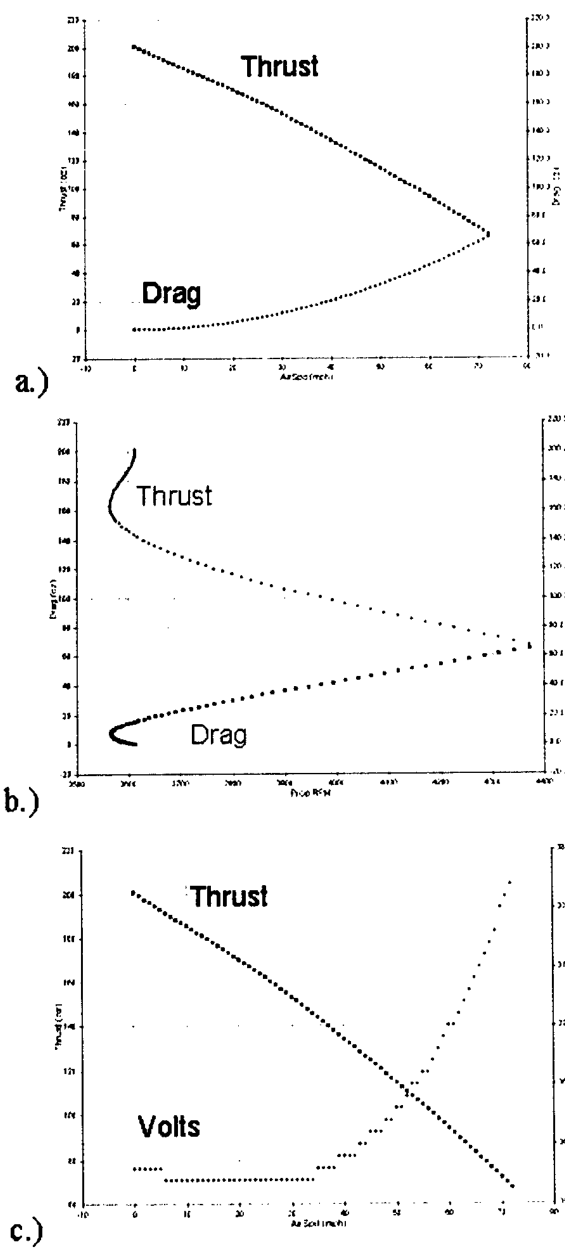


Figure 5.4a

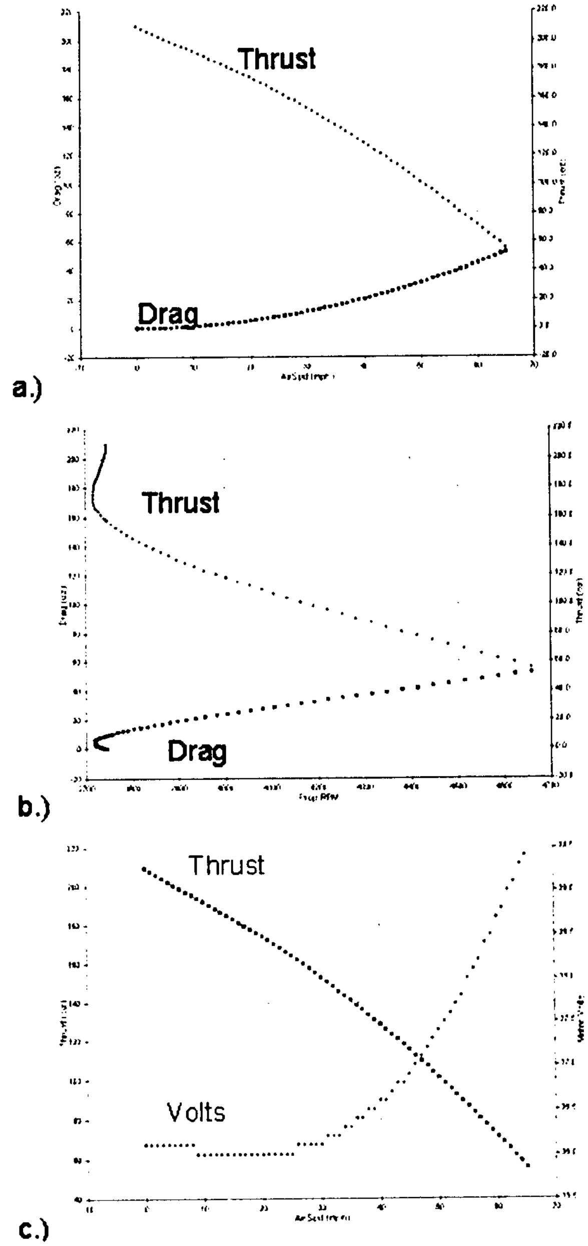


Figure 5.4b

The final propeller selections give a top speed of 73 mph for 22x18 and 65mph for 22x16 propeller as shown in Graph a.) of Figures 5.4a and 5.4b. The static thrust provided by the two propellers is approximately the same. A slightly decrease in thrust from the 22x18 is due to the very high pitch of the propeller. There is a point were increasing the pitch would no longer have positive affects on the thrust, and the thrust eventually starts to decrease. For the 22-inch diameter propeller, this point happens

at pitch values between 16 and 18 inches. Further increase of the pitch would decrease the thrust and the only way to increase the thrust is to increase the propeller diameter.

Changing the diameter of the propeller dramatically increases the torque required to turn the propeller at given speed and this has an unsatisfactory contribution of shorter engine-run times. Graph b.) in Figures 5.4a and 5.4b shows the propellers RPM required to produce need thrust for optimum flight score. Higher RPM is required for 22x16 then for 22x18 propeller to have the same thrust. In Figures 5.4a and 5.4b Graph c.), the thrust and voltage curves are plotted vs. airspeed. Higher pitched propellers have a requirement of higher currents needed. One of Ni-Cad's natures is that when high currents are drawn, the terminal voltage drops and that represents a lower terminal voltage for battery pack. This voltage drop can be seen on the c.) graphs (Figure 5.4 a&b) where thrust and voltage curves are plotted vs. airspeed.

5.1.5 Landing Gear

Due to difficulty of construction and incompatibility with the structural mainframe, the team decided to build a fixed landing gear. The landing gear has a mount 10in. from centerline on the wing. The wheels are 3in in diameter and 0.125in thick. They are made of aluminum. The two main struts are made of laminated 0.5in x 0.125in carbon fiber bars.

The total length of the landing gear struts is 9.5 in. The struts are mounted to the main wing spar at a 20° angle from vertical; this places the axel of the wheels 3.9 in. in front of the CG. Mounting the landing gear in this manner helps prevent the plane from "nosing over" on the takeoff roll, or when taxiing. The combined height of the struts and the wheels gives the plane a ground clearance of 10 in.

Fairings were added around the struts and wheels to reduce drag. The fairing has a 3.75in chord and a thickness varying from 0.75in to 1in. The length of strut from the bottom of the wing to the wheel has a thickness of 0.75 in. The remaining part of the fairing covers the wheel and is 1.0 in. thick. This portion is sheeted with balsa and covered in Monokote™. The upper portion of the fairing is formed by ribs attached directly to the carbon fiber strut and are covered in Monokote™. The ribs within the upper portion of the fairing are not sheeted because the balsa is inflexible and could break apart after any hard landing or major flexing of the struts.

The tail wheel is considerably smaller than the front two wheels and is mounted inside the rudder. The bottom of the rudder at its hinge point is 0.5 in. thick. This allows us to mount the wheel within. The primary reasoning for this is to reduce profile drag. A secondary reason is for consolidation of resources. Mounting the wheel in the rudder reduces the number of servos needed, and reduces the complexity of linkages needed to connect an external wheel to the rudder system.

5.2 Handling Characteristics

The Handling characteristics of the Marooned aircraft provided a chance for the team to refine the design. The majority of this refinement was accomplished in the configuration of the horizontal tail. The design guidelines used during the Preliminary Design phase were found to be inadequate for the pitching moment of the airfoil used. In addition the team questioned the Wing Team Leader's assumption that a 3.2 factor would produce a stable aircraft. The team decided to stick with the tail length to simplify the analysis, and because it provided a shorter aircraft that produced a smaller RAC.

Horizontal Tail Configuration	
Span b	33.1662 in
Chord c	6.63333 in
Airfoil	NACA 0012
Angle of Incidence i_w	3^o
Aspest Ratio AR	5
Taper Ratio λ	1
Elevator Chord	1.6583 in
Elevator Deflection δemax	60^o

Table 5.4 Final Horizontal Tail Configuration

The four main stability derivatives of concern to the team were $C_{L\alpha}$, $C_{m\alpha}$, $C_{m\delta e}$, and $C_{L\delta e}$. These were concerns to the team because of the negative pitching moment coefficient of the wing, and the experience last year with an aircraft sensitive in pitch. In addition the team did not have any numerical tools at their disposal, which could calculate lateral derivatives. Using Ref 8 the team did an initial analysis of the longitudinal stability of the Marooned aircraft. This analysis used the assumptions that the C_{mac} was located that the wing quarter-chord, the tail efficiency was Gamma = 0.9, the CG was at the quarter-chord, a 30deg maximum elevator deflection, and that the elevator was 30% of the horizontal tail. This produced a rear cg limit of 67.446% chord length, and a forward limit of 83.28% chord length, showing the plane couldn't fly. Several culprits were identified including an erroneous airfoil pitching moment value. A series of computations were conducted to find the final configuration of the horizontal tail as shown in Table 5.5. With this configuration the aircraft had a forward CG limit of 16.765% of the chord, and an aft CG limit of 56.16% of the chord. The produced a 4.333" inch CG travel range on the full-size aircraft, up from the 0.25" inch travel on the previous year's design. The stability derivatives calculated for this final configuration of the aircraft are available in Table 5.5. The absence of some of the lateral directional derivatives is due in part to the team's design. The ailerons were sized an extra 2in after the elimination of the flaps, and the rudder is larger than necessary too. The team's philosophy behind this is that the handling characteristics can be dialed into the aircraft with servo deflection-rate adjustments.

Lateral Stability	
$C_{L\alpha}$	7.7452
$C_{m\alpha}$	-4.3499
$C_{L\delta e}$	0.6341
$C_{m\delta e}$	-2.071

Table 5.5a

Longitudinal Stability	
$C_{n\beta}$	0.04091
$C_{n\delta r}$	-0.0194
$C_{L\beta}$	0.0385

Table 5.5b

5.3 Mission Performance

A performance analysis of the aircraft provided the team a validation of their design. The first part of the mission performance analysis was the takeoff distances for the aircraft for each of the three takeoff conditions. These takeoffs were computed using an estimated aircraft empty weight of 14lbs, rounded from a computation of the aircraft weight using the PrelimSheet code with updated area-specific weights for the wing and fuselage. Three different static thrusts of 9, 9.5, and 7lbs were used for these analyses. The first static thrust of 9 was used is a 3/4 throttle setting, which is being used to conserve battery energy for the GTOW load on the second takeoff. The 9.5lb static thrust is derived from tests of the engine that show that during a static test after the time period required to fly powered in the first flight segment the motor could guarantee at least that much thrust for the second takeoff. A similar procedure was used to find the 7lb thrust guaranteed available for the final takeoff. This provided the takeoff distances provided in the performance data found in Figure 5.2.

Lap	1	2	3
Take Off (feet)	36.79	105.96	48.03
Speed (mph)	60	60	60
Lift / Drag	15.8661	16.1612	15.8661
Turn Radius (feet)	48.286	62.096	48.286
Turn Loading (g)	4	4	4
Time (sec)	61.05	63.18	61.05
	Total Time (sec)	305.28	

Table 5.6 Competition Performance Chart

During the mission three performance figures were of main concern, the L/D for cruise condition, the turn "g", and the turn distance. The mission analysis was computed using the assumptions of a 60-mph cruise speed, and a 60-second landing time. The 60-mph cruise speed was used because it is higher than the most efficient speed of the motor, but lower than the maximum speed of the propellers,

providing a feeling of the performance produced from a ¾ throttle setting. The 60-second landing time was used to pick up any time lost from not flying at 60mph to landing or at takeoff, as well as to account for the loading time. The team had not had a chance to test the amount of time it would take to load and unload during the competition. The g varies so that a minimum amount of strain can be placed on the aircraft, while providing the smallest course length practical. It was in part derived from the cornering velocities of the aircraft in both unloaded/loaded conditions, which are 52.9185, and 69.2866 mph. The corner speed was kept at 53 for the unloaded flight, while the loaded flight corner speed was lowered to 60mph because it was known that the aircraft could achieve 60mph and fulfill the mission requirements. Using a 4g turn shortens the course, but also puts the aircraft to a safety factor of 1.843 from ultimate stress at a minimum. A final assumption to note in the data of Table 5.5 is that it assumes instant velocity changes, which wouldn't happen in competition, and possibly underestimating the time. However, it is still a good bar to measure the aircraft's performance at competition with.

5.4 RAC Analysis and Comparison

A final analysis of the design after the Detailed Design phase would provide the best qualification of how far the VT DBF team had come in the design of the Marooned aircraft. Some of the assumptions changed for this final aircraft RAC compared to the previous assumptions. The assumptions that produced the most change were the area specific weights, which significantly reduced the weight of the aircraft. This was justified by improved modeling of the aircraft's structures, load limits, and material composition over the course of the design, and compared to the previous year's design. Also, the elimination of the flaps allowed the assumption of 3 servos for the aircraft instead of the 4 required in the Preliminary Design. Another assumption that changed was that the horizontal-tail-span was less than 25% of the main wingspan. This rule was overlooked through the Preliminary Design, and it wasn't discovered until the end of the Detailed Design that it was realized the horizontal tail would be counted as a second wing. Upon analysis of this the team noticed that using the tail-span as a control seemed a poor choice for delineation of a second wing. The team humbly submits that the horizontal tail area would be a better control since most pitching-moment stability derivatives are based on the horizontal tail volume, which has the horizontal tail area, and not the horizontal tail span. A few assumptions that didn't change throughout the process were the landing gear C_{do}, and tail-planform specific weights. The C_{do} could be left the same because it did not affect the outcome of the RAC, and had a small effect on the overall C_{Do} of the aircraft. The tail-planform specific weights were kept heavy throughout the design because it allowed for some under prediction of performance due to weight, and because very little modeling of the tail structure was conducted.

Final Design RAC	
Wing	
Span	8ft
AR	8.7273
Area Specific Weight	0.2lb/ft2
Control Surfaces	4
Incidence Angle	0°
Fuselage	
Total Length	6.10533ft
Wetted-area Specific Weight	0.0025
Balls Carried	24
Horizontal Tail	
Span	33.1662in
Area	1.5278ft2
Airfoil	NACA 0012
Incidence Angle	-3°
Vertical Tail	
Span	15.1789in
Area	0.8ft2
Engine	CO-60G
Weight	1.4375
Battery Weight	4.7
Propeller	22x16
	22x18
RAC	12.0269

Table 5.6 Final Aircraft RAC

The RAC (Table 5.6) that came from this Analysis compares favorably with the RACs computed during previous phases of the design. The majority of this improvement is due to the decrease in weight realized by the team because of test building certain aircraft components, and making better analyses of the structures required to efficiently accomplish the 7g load case imposed on the Marooned aircraft. The 12.0269 RAC for the final design is lower than both RAC's despite the two-wing troubles encountered during the Detailed Design phase. A reduction of 6.19% was realized over the Preliminary Design, and a reduction over the erroneous Conceptual Design was 3.869%. Using the recalculated RAC value of 13.3869 for the Conceptual Design RAC revealed a 10.159% reduction in RAC over the course of the design process. This is evidence of the VT DBF team's hard work to produce an optimum aircraft that fit their building skills, budget, and the competition requirements.

6.0 Manufacturing Plan

In order for the team to implement the goal of winning this year's competition, an efficient manufacturing plan was required. Continuous building and rebuilding was done to make the structure better, lighter and stronger. This allowed the team to test the prototypes and gain experience in building techniques. The detailed manufacturing plan is presented in Figure 6.1. The actual manufacturing plan is shown in Figure 6.2.

6.1 Available & Required Skills

The successful building skills required to shape and design the wood frames structures for raw material are listed below. The summary of all skills needed and required to follow manufacturing plan efficiently are shown in Table 6.1. The ability to perform precise and aesthetic work is also important to achieve the goal.

Skills Available & Required	Number of Personnel with the skills
RC Modeling	2
Monokote™ covering	2
Fiberglass	5
Wood Design Skills	1
Wood Frame Building Skills	1
Foam Cutting Skills	7
Gluing Skills	17
Wood Cutting Skills	12
Soft Wood Skills	1
Precision Work Skills	9
Vacuum Molding Skills	5
Carbon Fiber Molding Skills	8

Table 6.1 Skills List

6.2 Material Selection

Many hours of research were put into material selections. The goals were to find the strong, light and easy to work with material. All metal sere eliminated because of their weight and difficulties of working with the metals. Another factors the eliminated the metal was that not many students on the team have experience working with such a material. Material selections leaned more towards the wood, composites and carbon fiber. All of these materials have excellent properties such as weight to strength ration, ease of building, and the materials are relatively affordable.

Final decisions for materials were: balsa wood, basswood and carbon fiber. Those three materials are light, strong and could handle the loads for the factor of safely of 2. The Table 6.2 shows material selection process.

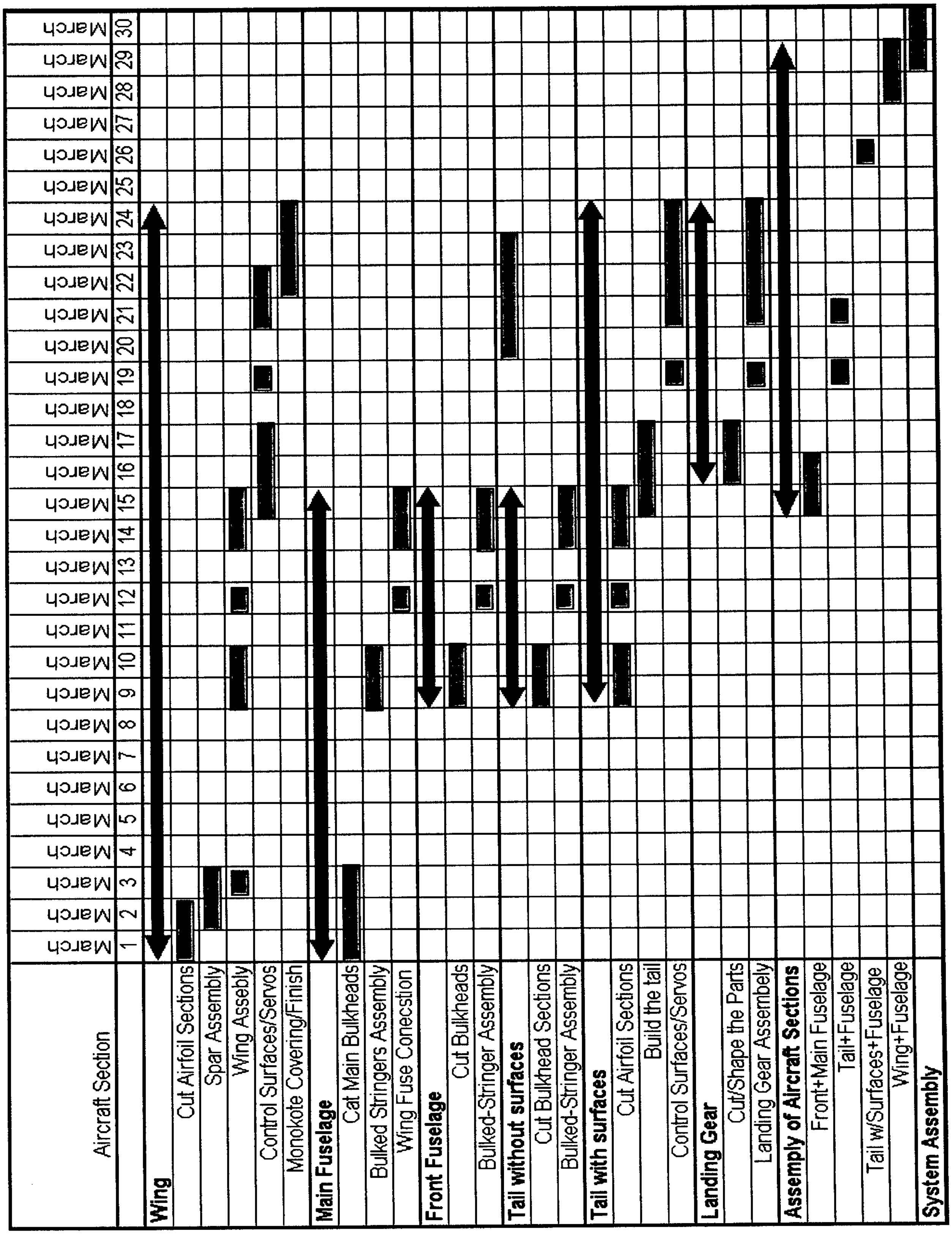


Figure 6. 1 Planned Manufacturing Schedule

Aircraft Section	March 1	March 2	March 3	March 4	March 5	March 6	March 7	March 8	March 9	March 10	March 11	March 12
Wing												
Cut Airfoil Sections												
Spar Assembly												
Wing Assebly												
Control Surfaces/Servos												
Monokote Covering/Finish												
Main Fuselage												
Cat Main Bulkheads												
Bulked Stringers Assembly												
Wing Fuse Conecstion												
Front Fuselage												
Cut Bulkheads												
Bulked-Stringer Assembly												
Tail without surfaces												
Cut Bulkhead Sections												
Bulked-Stringer Assembly												
Tail with surfaces												
Cut Airfoil Sections												
Build the tail												
Control Surfaces/Servos												
Landing Gear												
Cut/Shape the Parts												
Landing Gear Assembely												
Assemply of Aircraft Sections												
Front+Main Fuselage												
Tail+Fuselage												
Tail w/Surfaces+Fuselage												
Wing+Fuselage												
System Assembly												

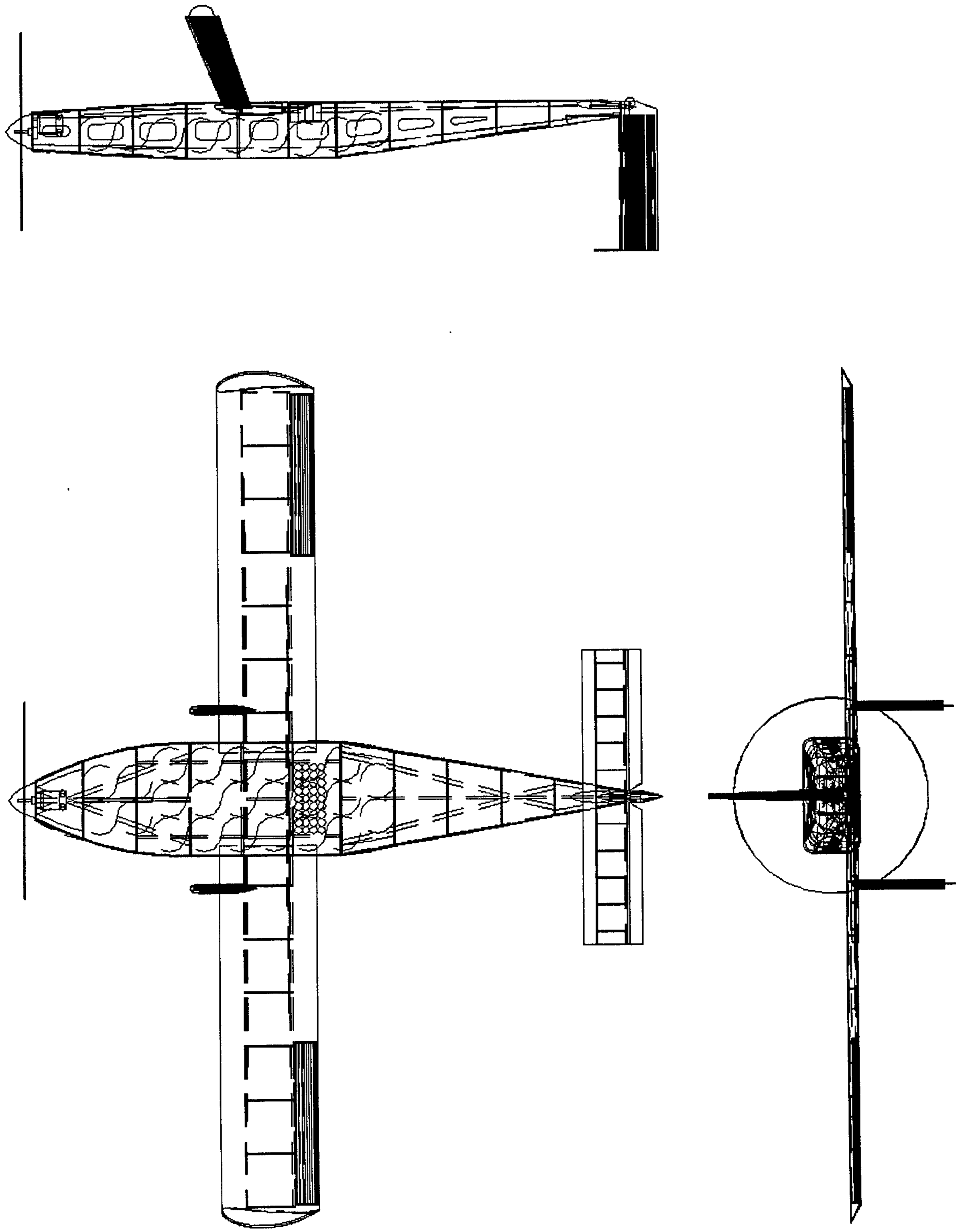
Table 6.2 Actual Manufacturing Schedule

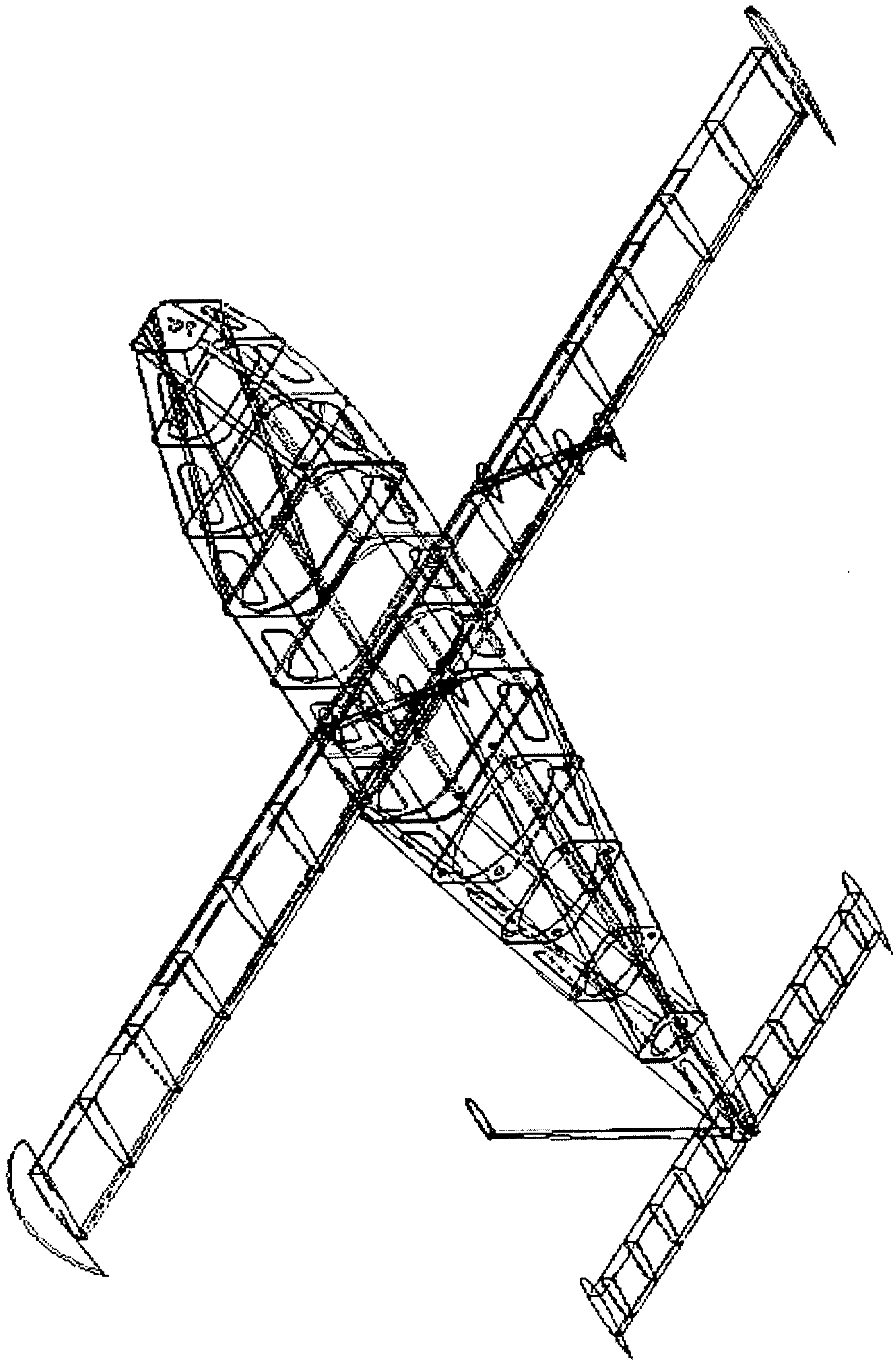
Material	Strength	Density	Cost	Skills needed	Final Score
Douglas fir	10	5	5	10	30
Balsa wood	4	10	10	10	34
Basswood	8	8	8	10	34
Spruce wood	6	7	8	10	31
Titanium	10	1	2	1	14
Aluminum	10	2	10	1	23
Carbon fiber	10	10	2	10	32

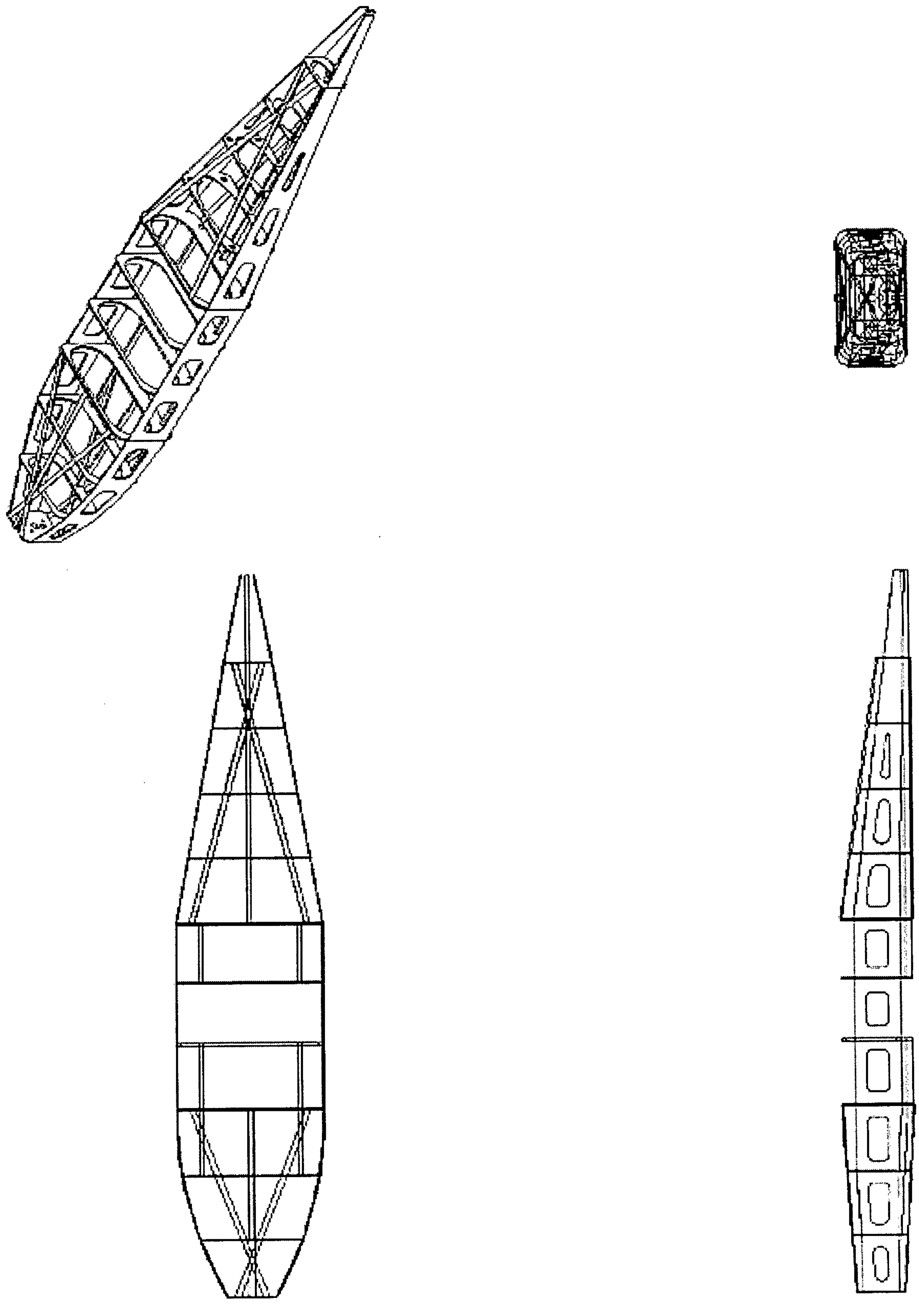
Figure 6.2 Material selections table

7.0 Final Aircraft Summary and Configuration

This section covers the final configuration of the Team Marooned aircraft. This includes the final calculated RAC, and the drawing package.







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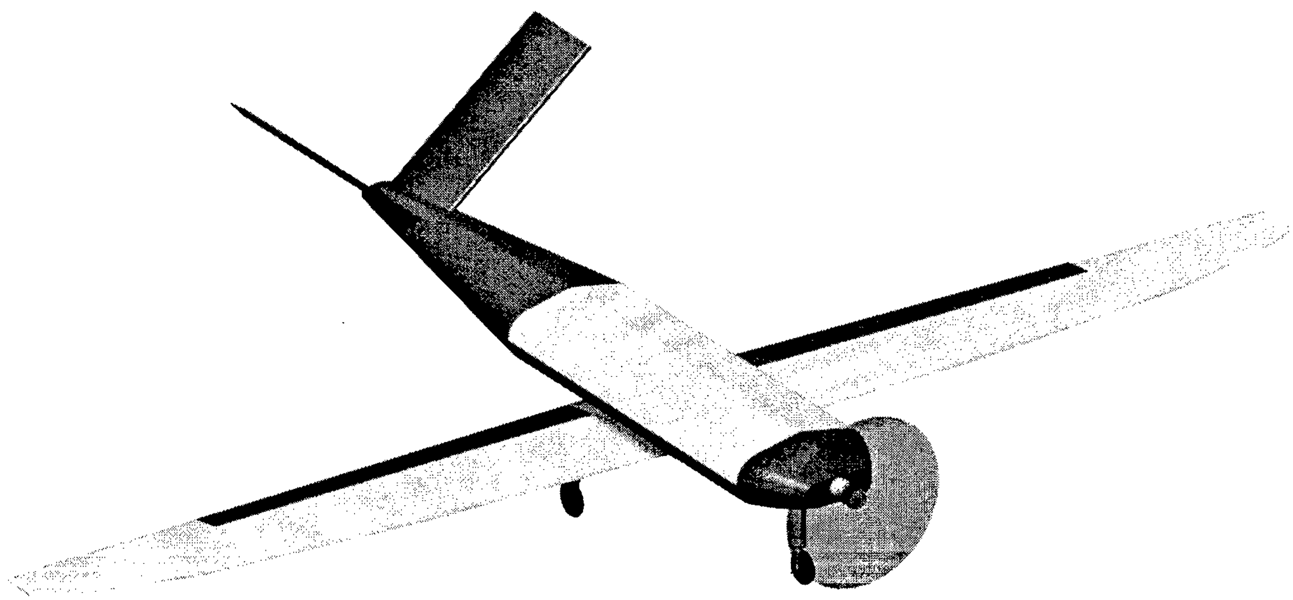
2001/2002 AIAA

Cessna/ONR Design/Build/Fly Competition

Design Report

PROPOSAL PHASE



California Polytechnic University

Third Base

Table of Contents

1. Executive Summary

California Polytechnic State University organized a small group of students comprised of both upper and lower classmen to compete in the Cessna/ONR Student Design/Build/Fly competition. The goal was to design and build a radio controlled, propeller driven, electric powered model airplane. The design mission was to fly two laps including a 360° turn in the opposite direction of the base without the payload, then pick up the designated payload and fly two laps with the same 360° turn, and finally fly for two laps without the payload and without any 360° turns.

1.1. Conceptual Design

The basic layout of the aircraft needed to be determined based upon the mission profile and method of scoring. A program specially developed to evaluate the performance of possible aircraft was used in choosing the right configuration and overall basic dimensions.

1.1.1. Conceptual Design Alternatives

In addition to the conventional layout, many alternative configurations and combinations were studied and reviewed. The team discussed the general configuration of the airplane, namely, a flying wing, a biplane, a canard, a blended wing/body, and a twin fuselage. Due to the necessity of good access to the payload, the placement of the wing was determined early in the design. In addition to its affect on wing placement, the payload affected the entire design. These affects were documented and became a dominating influence in the design of each component. One of these major impacts was on the fuselage. The arrangement of the balls determined its basic size and shape thereby determining the rated cost, frontal area, and the wetted area. Another consideration was where to locate the payload compartment relative to the center of gravity. The amount of payload that the aircraft was designed to carry was decided based on scoring methods. A ball trade study was done to minimize the amount of parasite drag.

For landing gear, the team discussed the tricycle and tail dragger configurations, as well as whether or not a retractable landing gear was optimal for the mission. Several tails were analyzed. Starting with the conventional tail, the team also looked at the cruciform tail, the T-tail, and the V-Tail. These were compared to performance records specifically from model aircraft. The advantages and disadvantages of a single engine/single propeller and a double engine/double propeller were weighed against each other and analyzed in a simulation program. The results of the conceptual design allowed the Cal Poly design team to investigate more detailed components that were addressed in the preliminary stage of design.

1.1.2. Conceptual Design Results

After discussing the different configurations, the team decided to build a conventional aircraft. The wing was placed on the bottom of the fuselage to allow for a door to be placed on the top for access to the softballs. To optimize the rated cost, the frontal area, the wetted area, and the aircraft s stability, the full payload allowance will be used and the softballs will be place in a three by eight set-up over the aircraft s center of gravity. The team decided to use one engine with one propeller. The team left the decision of the tail and landing-gear configurations to be made during preliminary design so further research could be made.

1.2. Preliminary Design

After the basic configuration of the aircraft was laid out, the group added more detail to the design and was able to perform more analysis of the aircraft components chosen. This is where the aircraft realy started to shape shape and develop a personality.

1.2.2. Preliminary Design Alternatives

An elliptical wing, a rectangular wing, a tapered wing, and a modified Shuemann wing were all investigated as possibilities for the wing planform. The body of the aircraft was fine-tuned and the fuselage configuration closeout was modified. The conventional tail and V-tail configurations were deliberated upon and the empennage was confirmed. Landing gear was investigated in more detail. The employment front-wheel steering was discussed. For the function of roll control, the ideas of roll coupling, conventional ailerons, flaps, and flaperons were all researched. Finally, the different materials available were analyzed.

1.2.3. Preliminary Design Results

The modified Shuemann wing was used in this design because it posed the best compromise between all of the concepts reviewed. The team chose a V-Tail over a conventional tail since it is slightly cheaper and performs better. The optimal landing gear for the mission was finalized as a non-retractable tricycle configuration for reasons of stability and simplicity. The landing gear was designed to have a steering front wheel without brakes.

1.3. Detailed Design

Once the general configuration was decided on, it was time to work on the specifics of the design, to define the components that would work together to achieve the best performance possible. Many aerodynamic components of the aircraft such as the planform, airfoil, empennage and high-lift systems configuration were finalized. These aerodynamic components were evaluated with programs such as X-

foil and the performance evaluation program specialized for the competition. An evaluation of the effect the gear selection will have on the ground handling qualities of the airplane was conducted. Batteries were selected and assembled into packs on the basis of energy density and discharge rates. The performance characteristics of various motors from Graupner and AstroFlight were compared and the best motor for the task was selected. Making a good choice of a propeller to mate with the selected motor was of big importance; various requirements were weighted against each other to come up with a viable solution. Having come up with a detailed overview of the components that will make up the airplane, simulations were run in order to determine various aspects of the design s performance. The exact position of components in the solid model of the airplane was modified in order to achieve an acceptable degree of balance.

1.3.1. Detailed Design Alternatives

When refining the aircraft design, many design alternatives were deliberated on. The exact airfoil choice was a difficult compromise between two drastically different types of airfoils during which more than 20 successive designs were evaluated in X-foil. Different models of motors from AstroFlight and Graupner had to be discarded when making the selection. Two types of battery cells both seemed quite well suited for the task of powering the airplane, and a choice had to be made between the two.

1.3.3. Detailed Design Tools

The tools used in the detailed design process were X-foil and the DBF performance simulation program. A solid model of the aircraft used for visualization and balancing was created in Solid Works 2001. Many of the other estimates and design methods used by the team originated from previous years of Cal Poly DBF.

1.3.3. Detailed Design Results

By the end of the detailed design phase, a solid aircraft design emerged. The estimations conducted by the design team predicted an RAC of 7.927, and an ideal final competition score of 62.3, with a flight time of slightly more than 6 minutes.

2. Management Summary

To increase productivity and keep the project on track, the group organized itself and set a timeline.

2.1. Design Team Architecture.

Cal Poly s Design/Build/Fly team consisted of only four members. This is about a third of the team size that would normally have been used for such a project. A small team of inspired and dedicated people working in a semi-autonomous way can often achieve much more in a short amount of time than a large and bureaucratic set-up, even if it has much better resources on paper. Because of the small group size everyone was integrated into each part of the design. Working in this fashion proved to be a benefit because everyone was exposed to everyone else s area of expertise. At the same time, each member was responsible for his or her own assigned areas of design. All of the team members met regularly to share new research and to finalize parts of the design. The personnel assignment chart is listed in Table 2.1.

2.2. Management Structure

John Asplund, bringing much experience and knowledge to the group as a DBF alumni, was an obvious choice for the project manager. Asplund assisted the group in getting a realistic schedule for the different assignments, which proved crucial to accomplishing the set goals. Each group member decided what they would be responsible for with the approval of Asplund and the rest of the group. Asplund aided each member in their respected assignments and all final decisions where presented to him at the end of each stage of design.

2.2.1. Personnel Assignments

The project manager helped to make the timeline by giving the rest of the team members a more realistic view point on how much time and effort each task would actually take. He met with the team during various times at each phase to affirm that the team was on track and to inquire as to what assistance was needed. The design team worked on a day-to-day basis with the pilot, Brian Buaas to ensure that the final design posessed adequate flying characteristics.

Two of the team members worked on getting funding and performed a large amount of the detailed design. One of these students made the CAD models and worked with the spreadsheet. The whole team worked together in organizing the team, making the timeline, designing and building the model, and writing the report. A list of the group members and their assigned areas is documented in Table 2.1.

Table 2.1 - Design Personnel and Assignment Areas

Design Personnel	Assignment Areas
Francesco Giannini/AE/JR	Public Relations; Funding; Conceptual Design; Preliminary Design; Detail Design; Manufacturing
Yevgeniy Gisin/AE/JR	Performance; Funding; CAD; Conceptual Design; Preliminary Design; Detail Design; Graphics; Manufacturing
Erin Clare/AE/JR	Report; Recorder; Conceptual Design; Preliminary Design; Graphics; Manufacturing; Stability and Control
John Asplund/AE/GRAD	Airfoil; Propulsion; Structure; Manufacturing

2.2.2. Schedule Control

A time chart kept the whole process organized and moving. Because of the many tasks that needed to be completed, deadlines for each stage of the design were set. The date of completion for each task was projected at the beginning of the project. As the project developed, the actual time of completion was recorded and entered into Table 2.2.

2.2.3. Configuration Control

Having a small team kept the lines of communications shorter and more direct, therefore, there was less opportunities for misunderstandings. While each group member was responsible for and had to report on a separate part of the project, the group was integrated into all parts of the project and constant communication between the members was maintained.

Event	Start	Finish	September	October	November	December	January	February	March	April
Team selection	9/25/2001	10/3/2001								
First meeting, orientation	10/3/2201									
Conceptual design stage										
scheduled	10/10/2001	11/14/2001								
actual	10/12/2001	11/19/2001								
Preliminary design stage										
scheduled	11/15/2001	12/23/2001								
actual	11/17/2001	1/10/2001								
Detailed design stage										
scheduled	1/9/2002	2/20/2002								
actual	1/9/2002	3/1/2002								
Report preparation										
Journal	10/10/2002	4/21/2002								
Letter of intent mailed	10/31/2001									
Proposal composition	2/15/2002	3/6/2002								
Proposal mailed	3/9/2002									

Table 2.2 - Milestone Chart

3. CONCEPTUAL DESIGN

Cal Poly team reviewed possible configuration concepts that would best fit the design parameters. Based on the mission profile and the DBF rules, the basic aircraft configuration was chosen and the operation of the aircraft and interaction of its components were discussed and laid out.

3.1. Design Concepts Investigated

3.1.1. General Configuration

Configurations considered during this early phase of design included flying wing, canard, biplane, conventional, twin fuselage, and blended wing/lifting body. Significant figures of merit in the selection of the configuration were ease of manufacture, rated aircraft cost, performance, and weight.

Table 3.1. — Figures of Merit for General Configurations

Description	Manufacture	Performance	Rated A/C Cost	Weight
Flying Wing	Difficult	Low	Good	Low
Biplane	Medium	Medium	Medium	High
Conventional	Easy	Good	Medium	Medium
Canard	Easy	Medium	Medium	Medium
Blended Wing/Body	Difficult	Good	Medium	Medium
Twin Fuselage	Medium	Low	High	High

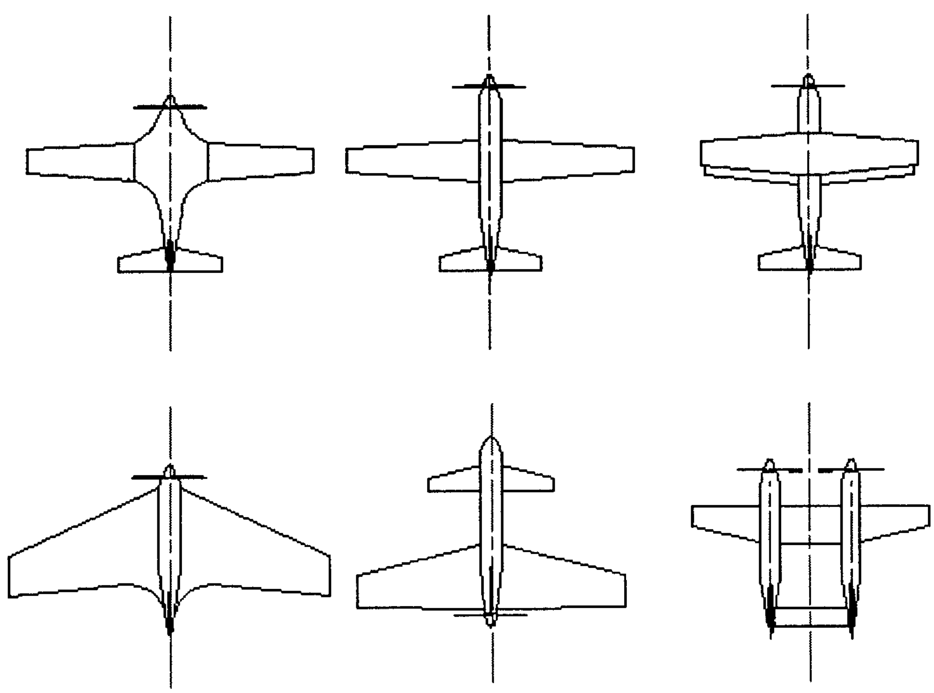


Figure 3.3.

Flying wings require airfoils with upwardly reflexed camber mean line near the trailing edge. This produces a positive pitching moment that obviates the need of a tail. Unfortunately, the maximum coefficient of lift for that airfoil is adversely affected, largely negating the benefits of this configuration. Flying wings make the installation of a high lift device problematic; the high pitch-down moment created upon deployment of these cannot be offset by the limited capabilities of a tailless airplane. The team generally felt that too much time would be required to fine-tune this configuration.

Biplanes achieve a greater total lift by increasing the wing area and the wing can be made lighter if externally braced, albeit at the expense of drag. The relative closeness of the sets of wings leads to interference drag and a reduction of efficiency. A monoplane with the same reference area but greater aspect ratio achieves greater efficiency. Since this year s rules place no real limit on the wingspan, this configuration was discarded.

The team seriously investigated the canard configuration. This concept seemed promising on the account that both foreplane and main wing act as lifting surfaces, and a smaller, more compact airframe can be built. On the other hand, the particular lift distribution between the two surfaces requires special considerations. At high angles of attack, the foreplane must stall before the main wing; this ensures that an unrecoverable departure does not result. In attempting a dive, however, the main wing must arrive at its angle of zero lift before the foreplanes achieve zero lift. If the foreplanes were to cease to lift while the main wing still lifts, a violent dive would result.[1] These and other complications involved with fine-tuning this configuration prompted the design group to decide against the canard.

The twin fuselage configuration proves attractive only in case the twin-engine solution is chosen. The mass of the fuselages is far from the centerline, which results in high moments of inertia and sluggish handling. The added expense of a second engine makes this arrangement unattractive.

The blended wing/lifting body configuration was rejected early-on, because of many reasons: For one, at the comparatively low speeds and Reynolds numbers at which the model operates, lifting bodies (basically low-AR wings with a low C_L/C_D) do more harm than good. Since the aircraft could not carry the payload in the wing, or anywhere outside of 16 inches from the centerline, it could not efficiently use the volume of a blended wing-body configuration. A blended wing-body would also be much more difficult to manufacture and operate.

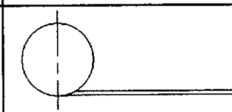
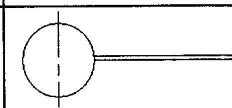
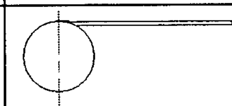
A conventional layout presents no technical challenges. It is easy to manufacture, its handling characteristics are well understood, and the design team generally felt that it conformed well to this year s set of rules.

3.1.2. Wing Placement

In terms of wing/fuselage arrangement, the wing can be classified in three categories: high wing, mid wing, low wing. Figures of merit include: lateral stability, ease of manufacture, interference drag, and serviceability. Cost remains unchanged from one configuration to another. Easy access to the softballs is

crucial since time is a big factor in the mission. The wings must be placed out of the way of any door or flap that open and the door/flap must have a reliable hinge and latching device. The wing placement which allowed for easiest payload access was the low wing design and so it was favored throughout the design process.

Table 3.2. — Figures of Merit for wing placement

Description	Manufacturability	Serviceability	Lateral Stability	Drag	Figure
Low-Wing	Easy	Easy	Medium	Medium	
Mid-Wing	Difficult	Difficult	Medium	Low	
High-Wing	Medium	Easy	High	Medium	

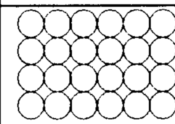
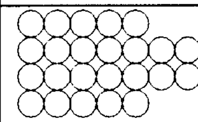
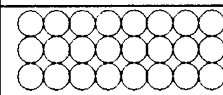
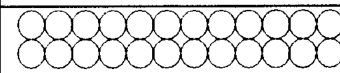
3.1.3. Payload

By the virtue of its size and importance, the entire aircraft was essentially designed around the payload. During the mission, the aircraft s weight will be changed multiple times as softballs are loaded and unloaded. To keep the payload independent of the aircraft s stability, the center of gravity of the payload compartment was placed directly over the center of gravity of the airplane. The size of the tail is affected as well since the length of the tail moment arm is determined in large part by the fuselage length needed to house the softballs. The payload had great impact on determining of the overall fuselage length, size, and shape.

3.1.4. Ball arrangement trade study

The design group studied several storage configurations for the payload. Figures of merit for the payload were: frontal area, fuselage length, wetted area, rated cost. Frontal and wetted areas are proportional to the airplane s drag, the lower values being preferred. For tail effectiveness and improved fineness ratio, a long fuselage would be preferred; however, fuselage cost is directly proportional to this figure.

Table 3.3. — Trade Study

Arrangement	Frontal Area	Fuselage Length	Wetted Area	Rated Cost	Figure
4x6	High	Short	Low	Low	
2x5, 2x7	High	Medium	Medium	Medium	
3x8	Medium	Medium	Medium	Medium	
2x12	Low	Long	High	High	

3.1.5. Fuselage Configuration

Data gathered from the conceptual design phase suggested that the highest score would be obtained by carrying the maximum number of balls. Therefore, the fuselage and the rest of the airplane were sized to carry 24 balls.

When sizing the fuselage, many variables were considered:

Frontal area: Being an unavoidable component of parasitic drag, frontal area was an aspect of the design that team worked hard to minimize.

Wetted area: Another component that, in conjunction with skin friction contributes to parasitic drag, and which, the design team obviously tried to make as small as possible.

Fuselage length: If the tail is too short, a large empennage (generating considerable drag) becomes necessary, affecting control and stability of the aircraft. A short tail also necessitates a large degree of closeout, turbulating the air and adding to drag. A long tail, however, adds to the total length of the fuse, making the airplane prohibitively costly.

Ease of access to the payload: Payload access was planned through a door at the top of the fuselage — a decision that is more so supported by the use of a wing mounted low on the fuselage and a load-carrying keel to connect the components of the airplane.

3.1.5. Landing Gear Configuration

The landing gear of a propeller driven airplane serves two major functions. The first is to provide adequate clearance between propeller tips and the ground. The second is to permit the plane to rotate on both takeoff and landing so that the wing s angle of attack comes close to the stalling angle of its airfoil. At that angle of attack, the wing is near the airfoil s maximum C_L. This permits the lowest landing and takeoff speeds of which the model is capable. There are two major types of landing gear: tricycle and tail-

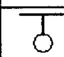
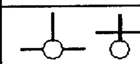
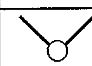
dragger. These tail configurations were further studied in the preliminary design.

3.1.6. Empennage Configuration

For an airplane in level flight at its selected cruise speed, the sum of the positive and negative pitching moments must be zero. Four major moment sources must be compensated. The main source is CG location. A CG that is ahead of the mean aerodynamic chord s quarter chord causes a nose down moment and results in a longitudinally stable airplane. Locating the CG further back decreases static pitch stability. The other sources of pitching moments are the wing s drag moment, the thrust moment and the airfoil pitching moment. The balancing is achieved in a conventional and canard design by the use of a horizontal tail or foreplane. The horizontal tail s angle of attack, relative to the wing s downwash, should be sufficient to provide lift or most often, downforce required to provide equilibrium.

The aircraft rated cost formula shows a small cost break for a v-tail compared to all other tail plane configurations. However V-tails could be more difficult to size properly which was also a consideration. As a result of conceptual design, conventional and the v-tails were both considered to have enough merit to be deliberated upon further in the design process.

Table 3.4 — Empennage configuration

Description	Manufacture	Rated Cost	Handling	Merit	Figure
T-tail	Difficult	Moderate	Good	Low	
Conventional/Cruciform	Easy	Moderate	Good	High	
V-tail	Easy	Low	Moderate	High	

3.1.7. Power Plant

The selection of the number and size of engines can easily be considered one of the most crucial decisions that the team had to make. The simulation program, mentioned in section 3.4.1., specified that the airplane would easily be able to take-off within the allotted distance using a single motor limited to 40 amperes; therefore eliminating the need for two motors/propellers. In addition, a single large engine is known to be more efficient than two smaller engines of equivalent power[2]; and a single large engine also greatly decreases the drag and complexity of the design, while increasing its reliability.

3.2. Conceptual design results

Based on results from previous model aircraft, a conventional layout was chosen, with a low mounted wing. This provides a good access to the payload section and good drag characteristics. The team selected a tricycle landing gear for good ground handling. The retractable solution does not present any significant advantage over a fixed solution, the reduced drag advantage being negated by the added cost and complexity, not to mention reliability concerns, so a non-retractable landing gear was chosen.

3.3. Possible Designs and Figures of Merit

The competing design concepts were mainly evaluated based on the following 5 Figures of Merit (F.O.M.s).

- **Best Total Score** — This FOM was not numerically used during the evaluation of components that comprised the bulk of conceptual design.
- **Ease of manufacture** — One of the most important F.O.M.s for our design was the ease of manufacture. Having very limited time allotted to manufacture the model, the team was willing to sacrifice some aspects of performance for this goal.
- **Serviceability** — A design that will be easy to service, modify and repair is very important. This FOM reflects the ease with which the design can be repaired after a crash and/or the modularity that the concept may possess.
- **Ease of handling** — In order to avoid a crash and to make the job of the pilot as simple as possible, good flight characteristics is a very important aspect of a conceptual design.
- **Aesthetics** — Although not nearly as important as the other 4 F.O.M.s, this was a consideration in some of the decisions we made. This not-quite-official FOM had no weight and wasn t used heavily in our decisions.

3.4. Assumptions/Design Methods

Numerical data need not be extensive at this stage, but should include as a minimum: a final ranking chart giving the quantitative value of each design for each FOM; the FOM importance factors or weighting; and an explanation of the features that produced the final configuration selection.

3.4.1. Analytical Tools

The initial design of the airplane was done using an interactive spreadsheet program created in Microsoft Excel. The setup of the program required the design group to make many assumptions in order to get valid answers. This program was created to achieve accurate performance predictions for an airplane of the configuration that resulted from our earlier conceptual design work.

The program estimated the overall performance of the airplane by combining many specific performance calculations.

- **Weight** — This estimate was created from weight/area constants that were obtained from weighing model parts created via different traditional RC-model construction methods. Multiplying these weight/area constants by their respective estimate areas created the weight estimate of the airplane.
- **Drag** — A flat-plate drag estimate was created based on CD estimates and experience compiled by the previous year s DBF teams.
- **Thrust** — Thrust estimate was created using a propeller efficiency approximation.

- **Energy accounting** — A propulsion system efficiency constant (efficiency in converting electric energy into volts — motor + propeller + drive efficiencies) was used to estimate the amounts of current/energy that will be drawn out of the batteries during the duration of the contest.
- **Cost** — A cost estimate was easy to create by simply applying the given rules to the parts, the size of which was estimated by other components described above.
- **Mission time** — This estimate was created by adding together the following separate mission segments: Takeoff, Climb, Turns, Cruising, Landing, Un/Loading. Constant acceleration and constant-G turns were assumed to simplify the approximation.
- **Mission score** — This result was obtained by using the preceding estimates in the context of the competition rules.

The 4 main variables that we systematically modified to get the highest possible flight score were Airplane Weight, Wing Loading, Aspect Ratio, # of batteries. A load of other assumptions and initial values were used, but those did not vary when trying to select an optimum airplane configuration.

The simulation program helped to define some of the directions in which the development of the airplane should head. For one, a trade study made it clear to the team that the design of the airplane should focus on an airplane to carry 24 softballs.

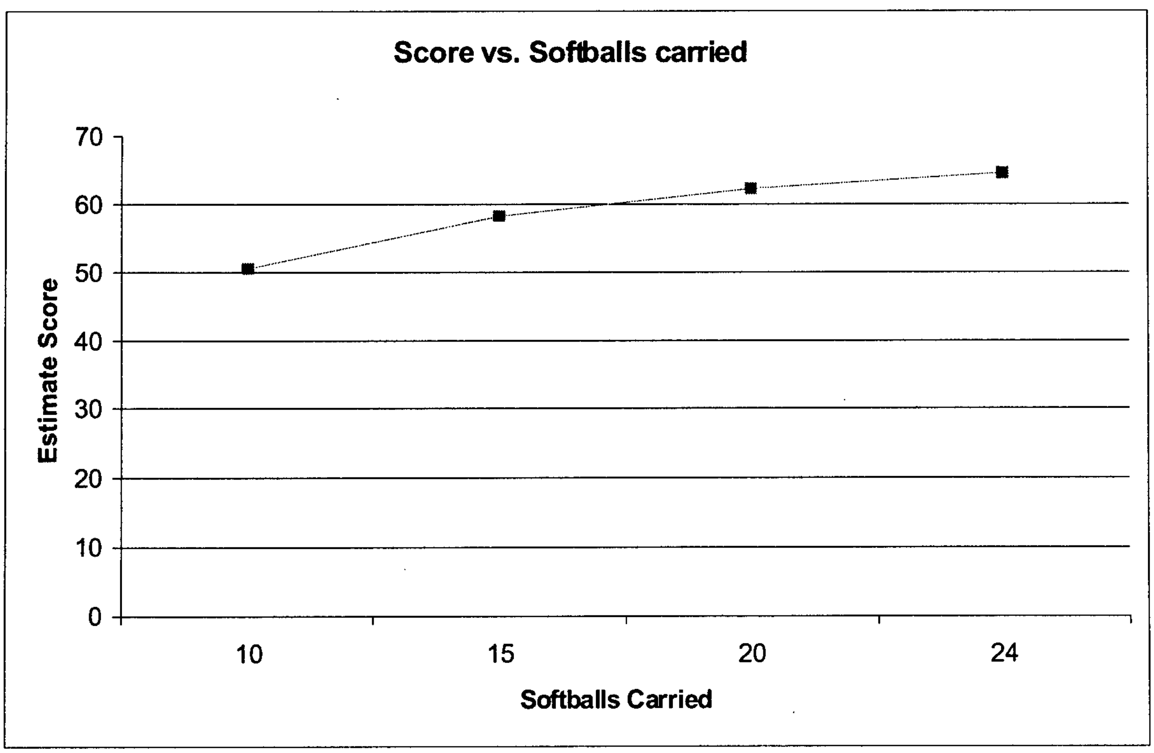


Figure 3.1.

Another trade study helped us determine which sections of the mission take the most amounts of energy — optimization of the airplane s performance on these mission segments would provide the most overall benefit.

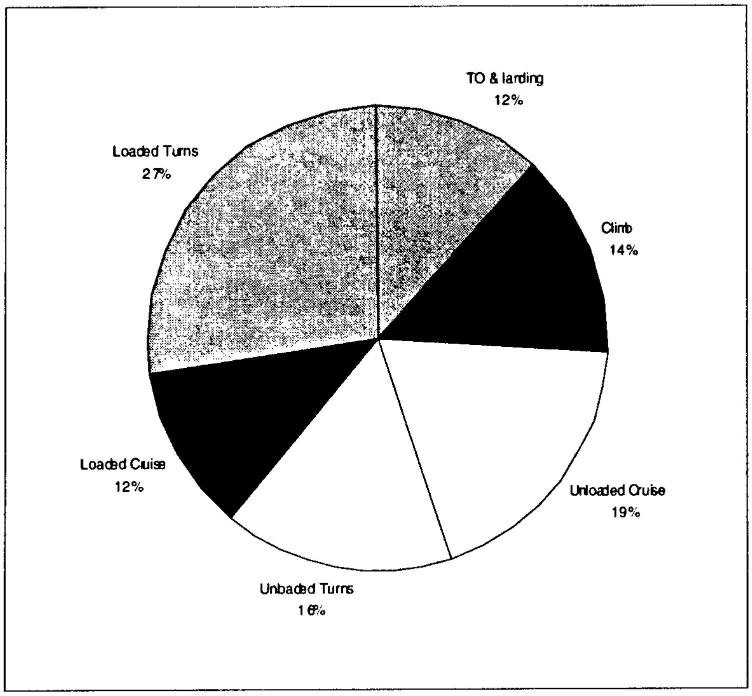


Figure 3.2. — Mission Energy Consumption Breakdown

Yet another useful piece of data that helped the design team streamline the design was feedback on how the cost of the design was being influenced by all the components.

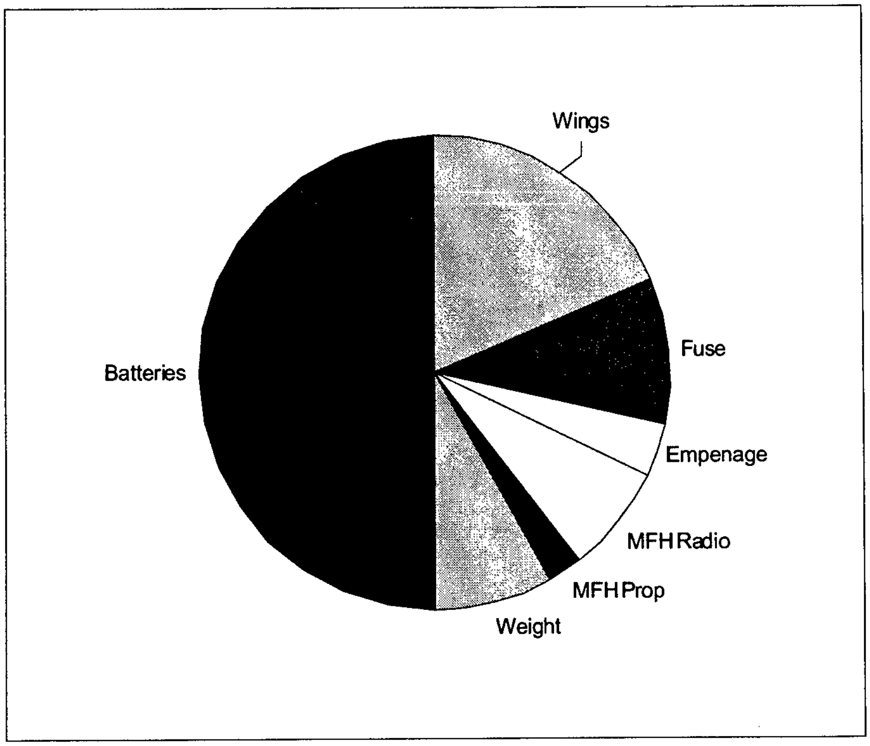


Figure 3.3. — Component Price Pie Chart

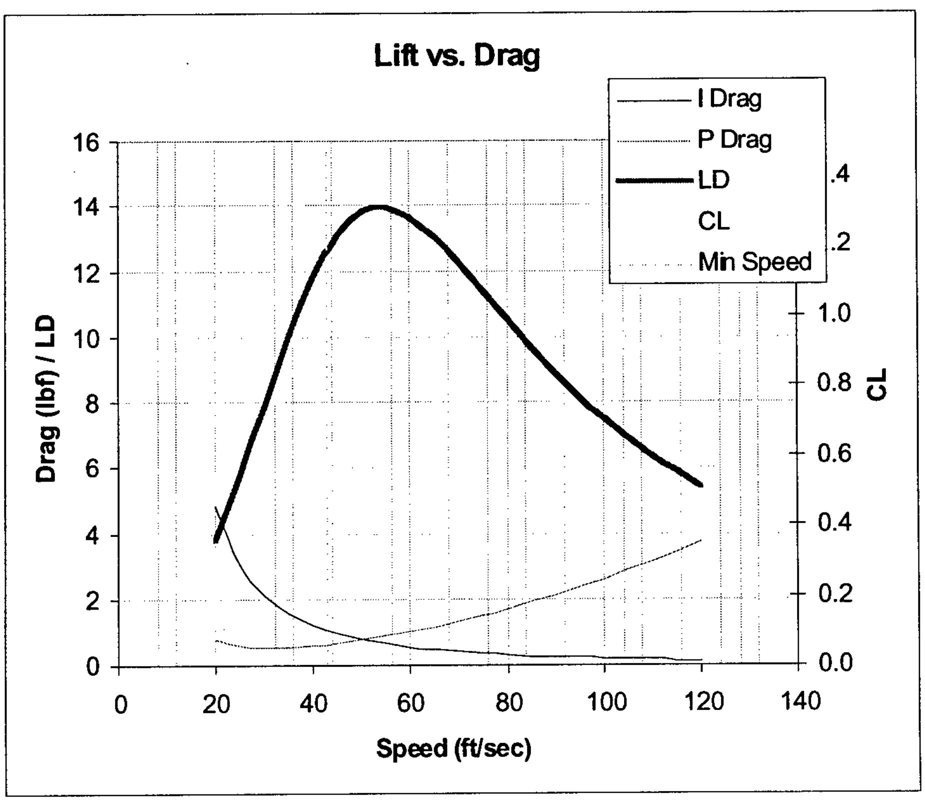


Figure 3.4. — One of the latest iterations of the optimum cruising speed chart

4. Preliminary Design

4.1. Alternative Configuration Concepts

4.1.1. Wing Planform

This is the wing s shape as viewed from above. It may be swept, elliptical, straight, tapered, or a combination of both. Swept wings were ruled out immediately because they only offer disadvantages at the expected flight conditions.

Elliptical wings: This is the ideal wing planform. It has the lowest induced angle of attack and induced drag, and stalls evenly across its span. These factors increase for rectangular and tapered wings. Structurally, the elliptical wing is difficult to manufacture, especially using some of the methods likely to be used on this type of an airplane.

Rectangular wings: Rectangular wings are the easiest to design and build. All chord sections are the same, and wing skins have a single chordwise curvature. While it suffers in comparison with an elliptical wing, for small models, it maintains a constant Reynolds number across its span. A tapered wing of the same area could have tip Reynolds numbers in the high drag/ lower lift and stalling-angle range of low Reynolds numbers, leading to premature tip-stalls at low speeds. Structurally, the wing roots need reinforcing, owing both to narrower root chords and higher bending moments that are generated as a result of the center of lift of each wing being farther from the centerline than an elliptical or tapered wing.

Tapered wings: A tapered wing with a tip chord of 40 percent of the root chord closely approximates the ideal elliptical planform both in induced angle of attack and induced drag. For wings of model aircraft, this taper ratio results in narrow tip chords and undesirably low Reynolds numbers at low speeds. Increasing the taper ratio produces larger tip chords. Lift is lost at the tips; the wider the tip chord, the greater the loss. The resulting loss in efficiency isn t great and is the lesser of the two evils.

Structurally, the tapered wing has lower root bending moments, and the wider, deeper root chord provides the greatest strength where it s needed most- at the root. A tapered wing can be lighter yet stronger than a rectangular wing of the same area.

Modified Schuemann wings: This planform has an elliptical leading and trailing edge for 70% percent of the semispan and a sheared wingtip. It comes close to the elliptical wing in efficiency and is more easily produced than an elliptical wing. The rectangular inner portion is wider in chord, which provides a strong root, and bending moments are lower than for a rectangular wing.

The design team chose the Modified Schuemann wing because it represents the best compromise in terms of performance.

4.1.2. Body

A blended wing-body/lifting body configuration was rejected early-on, because of many reasons: For one, at the comparatively low speeds and Reynolds numbers expected, lifting bodies (basically low-

AR wings with a low C_L/C_D) do more harm than good. Since the aircraft could not carry the payload in the wing, or anywhere outside of 16 inches from the centerline, the design could not efficiently use the volume of a blended wing-body configuration. A blended wing-body would also be much more difficult to manufacture and operate.

The team selected to adopt a 3x8 ball configuration for a fineness ratio of ~6 once nose and tail are added to the airframe. This configuration allowed the team to have a good balance of sufficient control arm moments and minimal fuselage length, while keeping the overall shape aerodynamically acceptable. Initially, a beaver-tail closeout was chosen for because it provided two axial points were the V-tails could be mounted. After further consideration it was realized that the because of the beaver tail closeout, the V-tail mounting points were away from the center of the fuselage, which is where the keel is located. In order to not modify the keel (which was likely to increase the manufacturing difficulties) it was decided to replace the beaver-tail closeout by a conventional closeout that can be seen in the current drawings.

4.1.3. Empennage

Although technically, there are many different tail configurations that could have been used, only two ended up being a point of serious deliberation. Configurations such as the T-tail, the cruciform tail, the canard configuration, and other unorthodox configurations were quickly discarded. The team weighted the moderate performance advantages offered by some of these configurations versus the increases in systems complexity and the amount of time required to build, test and fine-tune these advanced configurations and decided against using any of them.

The two configurations that the group deliberated on were the conventional tail (horizontal and vertical) and the V-tail configuration. Steering would have used a dedicated servo in either of the configurations to avoid the complexity of running a mechanical linkage (from conventional tail rudder servo to nose gear) through the entire length of the airplane.

Because a V-tail is priced at $15 versus $20 for a simple conventional tail, and offered better performance, the team chose the former.

4.1.4. Landing Gear

During preliminary design, landing gear was investigated more thoroughly. The team had to make a choice between two of the following landing gear configurations:

Tricycle: The CG is ahead of the main wheels and the nose wheel is steerable. This configuration is self-correcting directionally. The nosewheel steers, prevents the plane from nosing over and protects the propeller. Fuselage upsweep can be used to reduce the length of the landing gear legs.

Tail dragger: Tail draggers are directionally unstable. On takeoff and landing, coarse rudder use is needed to maintain directional control. As the tail comes up, prop torque and gyroscopic forces cause the

model to veer. Tail draggers are prone to tipping over. Moving the wheels forward worsens the directional instability while on the ground.

Adopting a tricycle gear was an easy decision. It affords better ground handling and takeoff characteristics. The first big decision made when selecting the type of landing gear was whether it would be retractable or not. On one hand, retractable gear offers better flight performance because it dramatically decreases an airplane s parasite drag. On the other hand, retracts (and the servos to power them) add to the complexity and the rated cost of the airplane. After considering the success last year s team had with their landing gear design and the short amount of time necessary to fine-tune and troubleshoot the design, the design group adopted a conservative approach and chose to use a fixed fiberglass arch landing gear.

4.1.5. Roll control

Roll coupling: Wings for this type of control require more dihedral. When rudder is applied, the model yaws. Because of the dihedral, there is an effective increase in angle of attack for the wing on the outside of the turn. This situation is reversed on the opposite wing. This causes the aircraft to roll. Models that adopt such control need good spiral stability.

Conventional ailerons: These are usually 25% of the wing chord and 30% to 40% of the semispan in length. The upgoing aileron experiences less drag than the downgoing one, introducing adverse yaw. A remedy to this is aileron differential, where the upgoing aileron angular travel is approximately one third that of the downgoing one. Their position limits the allowable span for flaps to the inner portion of the wing.

Flaps: Well-developed flaps allow a higher permissible wing loading for flight at high speeds and still permit slow speed landing. In order to accommodate ailerons, these are located on the outboard part of the wings. When extended, they cause a sharp change in the coefficient of moment that requires the use of the elevator.

Flaperons: Flaperons are a form of plain ailerons that can be operated ailerons and drooped simultaneously to act as flaps. They extend for most of the wing semi-span, like strip ailerons. When in fully lowered position as flaps, and then used as ailerons, a strong adverse yaw is created. Rudder control must be introduced to counter the adverse yaw of this type of roll control.

4.2. Materials

The material chosen for the aircraft had to be lightweight and strong. The wing is made out of a foam core covered by fiberglass. Unidirectional carbon fiber is used for the spar caps. The foam is easy to cut and shape, therefore, it is good for making complex aerodynamic shapes. Foam is also a good material to use in the wing because it is lightweight. A good solid form can be made without too much concern of adding to the overall weight whereas a balsa wood built-up wing would have to be made of stringers and ribs, making the foam construction simpler to build. To reinforce the wing, balsa wood is

often added to the foam-core in the places of highest stress since it is has a higher modulus of elasticity when compared to the foam. The team decided to instead use the foam core backed up by fiberglass and carbon fiber, seeing as how the technology was available and would produce better results.

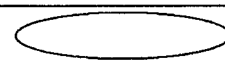
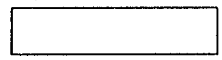
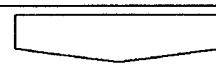
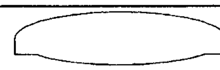
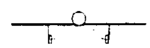
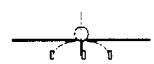
The fuselage has to be large to fit all of the softballs. A wood fuselage, similar to that of many RC models, would be strong throughout, but would also be quite heavy and not very aerodynamic. Instead, a keel and a shell construction are used for the construction of Third Base. The keel is very strong and takes all of the forces exerted on the aircraft during the flight, while supporting most of the aircraft s structures. The shell, being a non-structural component is free of any considerable stress, can be made lightweight and have holes cut into it without affecting the structural integrity of the aircraft. This is an especially important factor, considering that a door must be placed in the fuselage somehow to allow for access to the balls. The empennage consists of foam, fiberglass, and carbon fiber very similar to the wing. The main landing gear is made of a unidirectional carbon fiber bow that holds the rear wheels.

4.3. Figures of Merit and Design Parameters

- Weight — The take off gross weight was a major factor in the cost equation, so in addition to the performance and stability concerns, there was a weight limitation to be taken into account when selecting materials and components. Therefore, a lighter airplane was definitely one of the focus points of the design.
- Strength/Weight Ratio — Because the airplane is designed to be as strong as possible, while being light, the materials used to build the aircraft had to have a high strength to weight ratio.
- Cost — The cost is part of the total score and, therefore, it had to be as low as possible. (Written report score * Total Flight Score)/(Rated Aircraft Cost).
- Time to Manufacture — The amount of hours available to manufacture the aircraft was extremely limited due to school schedules of the team members, so the length of time to build the aircraft was a crucial design factor.
- Durability/Repair — If damaged propellers, motors, and other internal components can be replaced, the airframe structure, such as an entire wing, must be repaired. This means that the airplane needed to be as durable and as easily repairable as possible.
- Manufacturability — An easily manufacturable aircraft assures the overall timely completion of the airplane and is simpler to maintain and repair.
- Handling — Good handling of the aircraft is vital to the success of the flight. If an aircraft handles poorly, the pilot has to fight with the operation of the aircraft and spend less concentration on the flight mission. It is also more likely to crash if a problem occurs.
- Efficiency — The efficiency of coefficient of lift versus coefficient of drag had to be evaluated for all wing plan forms analyzed to produce the best possible performance.
- Complexity — The aircraft was designed to be as simple as possible so that the aircraft would cost less, be easier to build, and easier to maintain. For main components, such as the wing, where

an elaborated design will have a big positive effect on the aircraft performance, a more complex design is acceptable. A complex retractable landing gear would decrease drag, but the inconvenience and added cost of the electrical components needed would outweigh the benefits of reduced drag.

Table 4.1. — Figures of Merit

Wing Planform					
Description	Manufacturability	Efficiency	Handling	Merit	Figure
Elliptical	Difficult	Medium-High	Medium	Low	
Rectangular	Easy	Low	Good	Low	
Tapered	Easy	Medium	Medium	Medium	
Schuemann	Medium-Difficult	High	Medium	High	
Landing Gear					
Description	Manufacturability	Complexity	Handling	Merit	Figure
Tail Dragger	Medium	Low	Poor	Low	
Tricycle Retractable	Difficult	High	Good	Medium	
Tricycle	Easy	Low	Good	High	

4.4. Preliminary Design Results

At the end of the preliminary design phase, the Cal Poly design team decided on a modified Schuemann wing. For reasons of simplicity and stability, the team chose a non-retractable landing gear configuration. After analyzing the drag affects of different payload configurations, a three-by-eight ball configuration was chosen to be the best.

4.6. Summary of Key Features

The selection process that the team performed during conceptual and preliminary design contributed to creating a set of basic attributes to be possessed by the final configuration:

Gross Weight: 19.2lb
Cargo Capacity: 24 softballs
Battery Weight: 2.1lb
Wing Span: 9.8ft

Landing Gear: Tricycle
Roll Control: Flaperons
Construction: Composite
Empennage: V-tail

Wing Area: 6.1ft^2 Planform: Modified Schuemann

Max Wing Loading: 50 oz/ft^2

4.7. Assumptions/Design Methods

Analytical methods were used to determine the relationships of wing loading and span to the final score, and therefore allow for better sizing of both of these aspects of the aircraft. The tools used were parts of the DBF performance simulation program, which also affected the determination of the final gross weight of the aircraft.

5.1. Detail Design Configuration

5.1.1. Final Wing configuration

When determining the final configuration of the wing, the team was trying to achieve a very difficult goal. The wing needed to produce high levels of lift when operating during the loaded laps, but had to produce as little drag as possible when generating low levels of lift (when flying the empty laps).

The planform finally adopted was a modified Schuemann wing. *Refer to Figure 5.1.a.* The wing has elliptical leading and trailing edges for the inner 70% of the span and a sheared wingtip design. *Refer to Figure 5.1.b.* This closely approximates the ideal spanwise lift distribution, without incurring the low Reynolds numbers at the tips that can lead to the stall occurring at the tips earlier than at the roots. More specifically, the wing was designed so that at a given angle of attack, the outer section of the wing would be operating at a CL that is .1 lower than that of the root section. All the spanwise sections of the wing can thus fly near peak efficiency.

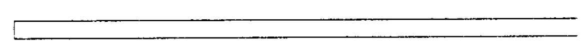


Figure 5.1.

Figure 5.2.

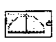


Figure 5.3.

5.1.2. High-Lift devices

In order to meet the takeoff constraint, and to attain efficient climb, the team investigated different ways to increase the maximum wing C_L. Simple leading and trailing edge devices were the most obvious

choice in this situation, with wing tilting, area increase devices (fowler flaps), and thrust vectoring being deemed too complicated and not overly efficient on this type of an airplane. Keeping in mind the penalty that the competition rules assign for extra servos (which would be required to power separate control surfaces), flaperons seemed to be the most obvious choice.

The flaperons extend for the inboard 70% of the wingspan and 30% of the chord, and are used for roll control. Model airplanes with moderate-chord flaps fully extended, and reduced throttle tend to porpoise upward. In order to prevent this behavior, elevator down trim must be applied. The sharp increase in angle of downwash from the flaps forces the tailplane down and creates a greater force than the increase in nose-down pitch.

Increasing the flap size to 30% of the chord produces a balance between nose-down and nose-up forces, producing little change in pitch trim. In addition to these benefits, the larger chord control surfaces proved to be better suited for airfoils designed with camber changing flaps like the JA-40, the airfoil designed for the airplane.

5.1.3. Airfoil Selection

Early in the design phase, it was recognized that the cost of the wing represented 20-40% of the total cost of the airplane. Wing cost was largely dependent on wing area, so in order to reduce cost as much as possible, the wing was designed capable of operating efficiently at high lift coefficients. This was a function mainly of the airfoil selected for the design. The wing planform was also considered in the design because of its impact on handling, aerodynamic efficiency, and the maximum lift achievable given a constant wing area.

In sizing the aircraft, many different airfoils were compared to find the airfoil and wing configuration that would maximize score and performance. The sizing results favored thick, highly cambered high lift airfoils for the two softball carrying payload laps. This allowed the takeoff requirement to be met easily and also was efficient during the relatively high CL cruise and turns. The airfoil chosen for the 2000-2001 Cal Poly DBF airplane is an excellent example of these high lift airfoils considered, the JA13. However during the four empty ferry laps, the optimal flight profile favored a high speed, low CL course in order to minimize flight time. With the lightly loaded airplane, the high lift airfoil is not desirable because of the lighter wing loading and the relatively low efficiency at low CL (0.3-0.4). During these four laps, a thinner, lower camber airfoil is optimal. These two different requirements were very difficult to meet with any single, commonly used model aircraft wing section. In attempt to achieve both of these requirements, an adjustable trailing edge camber changing airfoil was investigated. The goal of the airfoil design would be to achieve high lift and efficiency with the trailing edge surface deflected downward for the payload laps. The trailing edge would then be reflexed to a speed setting in order to shift the polar down towards the lower CLs for the empty ferry laps.

The design team investigated the camber changing airfoil concept using the same tool used to evaluate the commonly available model aircraft sections, X-foil. The design goal was to create an airfoil

with a maximum takeoff CL on the order of 1.5 during the loaded takeoff and provide higher efficiency than the other candidate airfoils during both the payload and ferry missions. Maximum lift coefficient, drag at loaded cruise (CL 0.8-1.2), usable CL range, unloaded fast cruise (CL 0.3-0.4), and behavior near stall were considered. The design Reynolds number range for the final configuration was Re(CL)^0.5 = 150,000. After much iteration, JA40, the final design of the airfoil was far more capable and well suited for the aircraft than any of the other considered airfoils. Figure 5.1.d shows the other candidate low Reynolds number airfoils considered for the design compared to the composite polar of the camber changing section.

Figure 5.1.e shows the JA40 with its characteristics in the three trailing edge configurations, cruise, high lift, and speed modes. Figure 5.1.f shows how the trailing edge position shifts the polar for the JA40. The airfoil has some different features compared to other low Reynolds number airfoils. One distinct feature is the kink in the top surface at the flap hinge line. This is designed to create a smooth upper surface when five degrees of flap are used. It was found that the boundary layer on top surface was very sensitive to the geometric disturbance at high CL on a conventionally flapped airfoil. Without the kink , this effect made it difficult to utilize relatively large flap deflections to achieve the broad CL range of the airfoil. In order to keep the drag low during the ferry laps, the airfoil thickness is fairly low compared to many of the other candidate airfoils. This tends to narrow the efficient CL range for most low Reynolds number airfoils but because of camber changing, the range of the JA40 is very broad.

5.1.4. Analytical Methods

Many of the tools used in the design process were created by the team members using Visual Basic or in Microsoft Excel. Nearly all the programs were written some time before work on the 01/02 Design/Build/Fly aircraft began and had been already well tested and validated. These programs include an electric motor/ propeller model, an aerodynamic model, a weight estimation spreadsheet, a longitudinal static stability model, takeoff and climb simulation, and a lift distribution spreadsheet. These tools were incorporated into a 00/01 DBF mission simulation model that included a cost model. To check validity, aircraft with known performance from the 99/00 competition were also run through the simulation successfully.

The only tool not developed by the team members was X-foil, written by Mark Drela of the Massachusetts Institute of Technology. After learning to use X-foil, results from low Reynolds number wind tunnel testing performed at University Illinois Urbana Champaign were compared with data produced by X-foil. At the relatively high Reynolds numbers that this aircraft will fly at, the correlation

Figure 5.4.

Figure 5.5.

Figure 5.6.

between test data and predicted performance was excellent. Experience gained with these correlations and the good results obtained with the JA13 airfoil designed using X-foil and flown during the 00/01 DBF contest gave the team confidence in the design tool.

5.1.5. Empennage

The V-tail was sized using the equations on page 124 of Aircraft Design[1]. To minimize the length of the fuselage, the team decided to calculate the tail moment arm based on the maximum horizontal surface span allowed by the rules. The appropriate areas were calculated for both the horizontal and vertical tails, which were then converted to their V-tail equivalents. The equation used are:

$$S_{VT} = c_{VT} b_W S_W / L_{VT}$$

$$S_{HT} = c_{HT}\_{w} S_W / L_{HT}$$

where L is the moment arm taken to be the length from the tail s quarter chord to the wing s quarter chord. S_W is the wing area, b_W is the wing span, and $\_{w}$ is the wing mean chord. The vertical and horizontal tail volume coefficients are c_{VT} and c_{HT}, respectively. These values were chosen based on experience gained from the past three Cal Poly DBF aircraft. The handling qualities of these airplanes were very good and the tail volumes were chosen using them as a model.

The Charles River Radio Controllers website provided a conversion equation to convert the horizontal and vertical tail area components into the V-tail area.

$$S_{V\text{-tail}} = S_{VT} + S_{HT}$$

$$\_ = \arctan[\hat{}(S_{VT} / S_{HT})]$$

$$S_{HT} = S_{V\text{-tail}} \cdot [\cos(\_)]^2$$

$$S_{VT} = S_{V\text{-tail}} \cdot [\sin(\_)]^2$$

$S_{V\text{-tail}}$ is the area of both halves together, rotated flat. _ is the V-tail s dihedral angle from the horizontal. The formulas taken from the Charles River Radio Controllers website are only to be used for large tail aspect ratios. These formulas do not account for the local interference and lift cancellation at the V-tail roots during yaw or rudder application. °

5.1.6. Controls

The aerodynamic controls consist of two flaperons and the V-tail. During the takeoff roll, the flaperons will be deflected downward five degrees to provide high maximum lift. During cruise, the flaperons will be returned to neutral deflection for efficiency. For the ferry mission, two degrees of negative flap will be used to improve the efficiency at high speed. The flaperons will also be electronically mixed to the elevator control to provide addition camber when up elevator is applied. The ground handling will be provided by steerable nose wheel on a tricycle landing gear system.

A single servo will control each flaperon, a servo for each tail control surface, and one to steer the nose landing gear. The controls consist of a small 4-cell battery to power the controls, and five Hitec HS-85 ball-bearing mini-servos. The speed controller will be next to the motor as will the nose landing gear steering servo.

5.1.7. Handling

Throughout the design process, the team ensured that the airplane would possess excellent handling qualities. These were recognized to be a major contributor to the success in the competition. The team wanted a predictable design that the pilot would be confident flying to the limits.

The design features include high lift devices enabling short takeoffs to be performed. The empennage was generously sized ensuring adequate control authority both in pitch and yaw. The wing planform and the associated airfoil gave the airplane a gentle stall. The main masses were located near the center of gravity; this resulted in minimum mass moments of inertia.

The tricycle landing gear, with its wide track and robust elements, made the airplane less susceptible than a taildragger to ground wind conditions while taxiing. The nosewheel was steerable, for effective and responsive ground handling and tight turns.

5.1.8. Battery Selection

The propulsion system must be powered by a nickel cadmium battery pack no more than 80 oz in weight as described in the contest rules. Many different types and sizes of NiCd batteries were available. Over half of the rated aircraft cost of most configurations was due to the battery cost, which is directly related to battery weight. Because of this, battery selection heavily drove the design. Choosing a battery that just met the energy and power requirements was vital in fielding a competitive aircraft. Depending on the characteristics and type of NiCd cell used, there are limits to the maximum power. Research on the characteristics of NiCd cells was conducted to choose the best battery to fulfill both the aircraft s energy and maximum power requirements. The voltage and maximum current capabilities of the available motors were also considered during the search. In addition, the contest rules state that a 40amp fuse must be used in series with the propulsion system. This current limit was also factored in when considering the available NiCd batteries.

Two styles of NiCd cells were investigated in detail, the high capacity Sanyo KR series, and the NiCd cells typically used for electric powered R/C models, the fast charge Sanyo R series. Table 5.1.e shows the characteristics of each cell type.

The fast charge cells are designed to handle high maximum currents and therefore are well suited for typical electric aircraft motors. The specific power density of these cells is on the order of 25 Watts/oz. These cells have a specific energy density of approximately 1.2 Watt-Hours/oz. The new Sanyo CP series cells have the highest energy density and excellent power density compared to all of the fast charge cells. Table 5.1.a shows the different NiCds considered for the design.

The high capacity KR cells were considered because of their higher energy density (1.6 Watt-hours/oz). The compromises in the cell design which gave it high capacity also hurts the cell s internal resistance their maximum discharge rate. Because of this, the power density is only half that of the fast charge cells, approximately 11 Watts/oz. Unfortunately, most of the design concepts required over 600 Watts peak power to meet the minimum climb requirement, making the high capacity cells unsuitable. Additionally, NiCd cells delivering power at their maximum rate are not operating efficiently which significantly reduces their delivered energy. Most electric motors available for powering a 20 lb aircraft require 20-50 Volts input instead of the near 100 Volts that a series pack of high capacity cells would be. A combination parallel-series pack was considered to reduce the voltage and increase the current capabilities, but the difficulties in charging and maintaining balance between the cells eliminated this idea.

The energy requirement for the battery was approximately 40 Watt-hours. Initially the aircraft was sized to use 13 CP-2400SCR cells because they provided the highest energy density, allowing the lightest, least costly battery pack. This pack would have been approximately 15 volts under load. Factoring the 40amp maximum current allows only 600Watts for takeoff and climb. This was unacceptably low and a higher voltage alternative was sought. The next smaller cell with competitive energy density was the CP-1700SCR. This cell arranged in a 19 cell series configuration fulfilled the energy requirement and was capable of over 800Watts.

The final NiCd cells chosen for the design were the Sanyo CP-1700SCR cells because of their high performance, availability, and suitability for the design.

Table 5.1.

Nicad Cell	*mAh*	*Weight (oz)*	*Package*	*mOhm/cell*	*Amps @ 1.0V*	*Watts/oz*	*Watt-hr/oz*
KR-1100AAU	1100	0.85	AA	20	10	11.8	1.55
KR-1500AUL	1500	1.09	4/5A	16	13	11.5	1.65
KR-1700AU	1700	1.25	A	14	14	11.4	1.63
N-1250SCR	1250	1.50	4/5sub C	4.5	44	29.6	1.00
RC-2000	2000	2.00	sub C	3.8	53	26.3	1.20
RC-2400	2400	2.15	sub C	3.6	56	25.8	1.34
CP-2400SCR	2400	2.1	sub C	3.6	56	26.5	1.37
CP-1700SCR	1700	1.65	4/5 sub C	4.5	44	26.9	1.24
CP-1300SCR	1300	1.25	_ sub C	6.5	31	24.6	1.25
N-3000CR	3000	3.00	C	3.3	57	20.2	1.20

5.1.9 Motor Selection

After initial sizing, it was determined that the aircraft needed a propulsion system able to run on approximately 16-24 NiCd cells, and be capable of handling 700-800 Watts for takeoff and climb performance. Both single motor and twin motor configurations were considered. It was determined universally that for a given power requirement, the single motor solutions outperformed the twins in final score. This was caused by the additional cost of having more than one motor and speed controller, because for a given power level, two smaller motors, gearboxes and propellers weigh more than a single (increasing empty weight), and because larger motors generally have higher efficiency. It was decided that unless no single motor of sufficient size was available, the aircraft would not have two motors.

To evaluate the performance of the motors available for the aircraft, a Visual Basic subroutine modeling electric motors was written. This model uses motor parameters including Kv (RPM/V), idle current, armature resistance, RPM and thermal limits to calculate output power and efficiency for DC motors. This program was used in conjunction with a propeller model and the mission simulation to compare the various motors offered by Astro Flight and Graupner.

Performance, reliability and availability were considered in selecting the motor. In the United States, the larger Graupner Ultra motors are not very popular nor are they regularly stocked by hobby vendors. The team members were also very familiar with the Astro product line and felt confident in their reliability. Although the Ultra motors outperformed the equivalent Astro Flight motors in the performance

model, the lack of availability and absence of suitable gearboxes for the Graupner s caused the team to choose Astro Flight products.

The power requirements for the design fell between three motors in the Astro product line, the Cobalt 25, the Cobalt 40 and the Cobalt 60. Table 5.1.b shows the Astro Flight motors seriously considered for the aircraft. The performance model showed that the electrical efficiency of these motors was very dependent on the voltage and current. The propellers indicated in Table 5.1.b were chosen to provide 700-800 Watts static power with a 19 cell CP1700SCR battery pack. Generally the motors with lower turn armatures were more efficient at full takeoff power while suffering in efficiency during the relatively low powered cruise. The higher turn motors showed lower efficiency at full power while being much more efficient during cruise. Most of the propulsive energy during the mission is used during cruise so the decision was made to optimize cruise efficiency.

Table 5.2.

Motor	***Weight (oz)***	***Max Watts***	***RPM/V***	***Ohms***	***Idle Current***	***Gear***	***Prop***	***Max***	***Cruise***
Astro 25 8T	11	500	971	0.093	2.5		9x6	75%	73%
Astro 25 8T	12.5	600	971	0.093	2.5	1.68	12x9	67%	66%
Astro 25 8T	13.5	700	971	0.093	2.5	3.1	14x12	67%	66%
Astro 25 5T	10	600	1475	0.039	4.5		10x6	76%	64%
Astro 25 5T	11.5	600	1475	0.039	4.5	1.68	12x9	68%	58%
Astro 40 10T	13.5	600	550	0.189	2		11x8	76%	78%
Astro 40 10T	15	750	550	0.189	2	1.68	14x11	71%	70%
Astro 40 10T	16	1000	550	0.189	2	3.1	17x15	71%	70%
Astro 40 8T	13.5	600	682	0.121	2.5		10x7	81%	73%
Astro 40 8T	15	750	682	0.121	2.5	1.68	13x10	73%	66%
Astro 40 8T	16	1000	682	0.121	2.5	3.1	16x14	73%	66%
Astro 40 5T	11	800	1161	0.05	4.5		11x6	78%	62%
Astro 40 5T	11	800	1161	0.05	4.5	1.68	13x10	70%	56%
Astro 40 4T	11	800	1364	0.05	5		11x6	76%	54%
Astro 40 4T	12.5	800	1364	0.034	5	1.68	13x10	68%	49%
Astro 60 11T	22	1100	347	0.103	2.5		13x10	82%	74%
Astro 60 11T	25	1200	347	0.103	2.5	2.75	20x16	74%	67%

All of the motors come available with various gear reductions. Although the gear reduction reduces the mechanical efficiency of the motor, the larger, lower disk loading propeller usually makes up the difference in efficiency while also providing significantly higher static thrust for takeoff. The Astro motors are capable of producing more power using a gear reduction because they can be run at higher voltages. The decision was made to use a motor with gear reduction.

The Astro 60 is capable of running reliably at 1200 Watts input power but is over half a pound heavier than the Astro 25 and 40. The Astro 25 and 40 are normally rated at 600 Watts input power direct drive which is lower than the limit for acceptable climb rate for the aircraft. The mission profile calls for short periods at full power for takeoff and climb followed by extended cruise periods well within the normal power limits for the Astro 40. The disadvantage of running the motor at high power levels is premature erosion of the brushes and commutator, and loss of efficiency due to increased resistance and demagnetizing of the permanent magnets. For a limited use contest aircraft, this was deemed an acceptable compromise. The team decided to use the Astro 40 10T with 1.68 gear reduction over the Astro 60 or lighter Astro 25 because of the light weight, highest cruise efficiency and improved scoring potential.

The propeller performance model indicated propellers 14 to 16 inches in diameter with 10 to 12 inches of pitch would load the selected motor to the required 700-800 Watts. The relatively high pitch of the propeller was chosen because of the aircraft s high cruise speed (60ft/sec). This pitch was a compromise between the advance ratio required for efficient cruise, and for good full throttle takeoff and climb performance. Because of variation in efficiency and power absorption between different propeller manufacturers, the propeller model was calibrated using data gathered from the motor manufacturer and with testing performed with an Astro 40. Final propeller selection will occur during flight testing.

5.1.10. Performance estimates

One of the most useful outputs of the program used in initial sizing of the aircraft were the tables of estimated performance. The tables presented below are the mission performance estimates for the final configuration of the aircraft.

Table 5.3.

Mission	***Condition***	***Time Used(s)***	***Time Total (s)***	***Energy Used (kJ)***	***Energy Rem. (kJ)***	***Score***
	Static	0	0	0.0	99.3	0.0
Mission 1	Takeoff	2	2	1.9	97.4	0.0
	Climb	3	5	3.0	94.4	0.0
	Turns	12	17	5.3	89.1	0.0
	Cruise	21	38	3.6	85.6	0.0
	Landing	14	52	0.6	84.9	0.0
	Cruise	34	85	5.8	79.2	0.0
	Landing	12	97	5.3	73.9	0.0
	Loading/un loading	30	127	0.0	73.9	0.0
Mission 2	Static	0	127	0.0	73.9	0.0
	Takeoff	7	134	6.3	67.6	0.0

	Climb	8	142	7.9	59.7	0.0
	Cruising	16	158	3.6	56.0	2.3
	Turns	26	184	13.6	42.5	2.3
	Hv. Cruise	35	218	8.1	34.4	2.3
	Turns	26	245	13.6	20.8	2.3
	Landing	14	258	0.6	20.2	2.7
	Loading/un loading	30	288	0.0	20.2	2.7
Mission 3	Takeoff	2	290	1.9	18.3	2.7
	Climb	3	293	3.0	15.3	2.7
	Turns	6	299	2.7	12.7	2.7
	Cruising	21	320	3.6	9.1	2.7
	Landing	14	334	0.6	8.5	3.4
	Cruising	34	368	5.8	2.7	3.4
	Turns	6	374	2.7	0.1	4.8

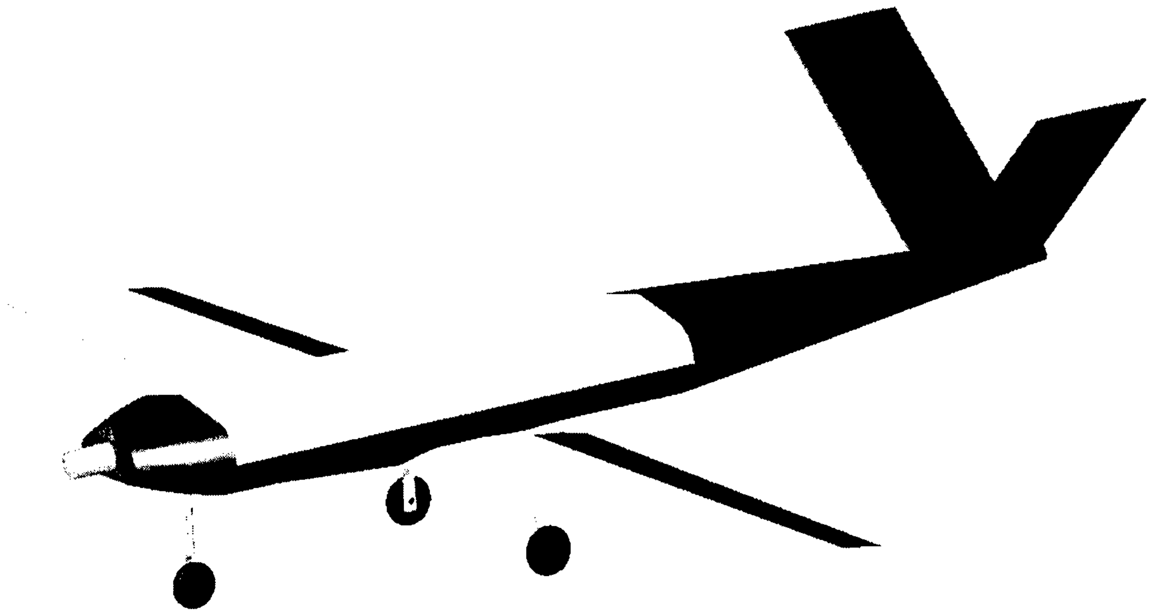
The design team was able to incorporate the weight estimates from the performance program into the solid model of the airplane. The solid model was balanced to maintain adequate stability in both loaded and unloaded configurations using the following balance worksheet.

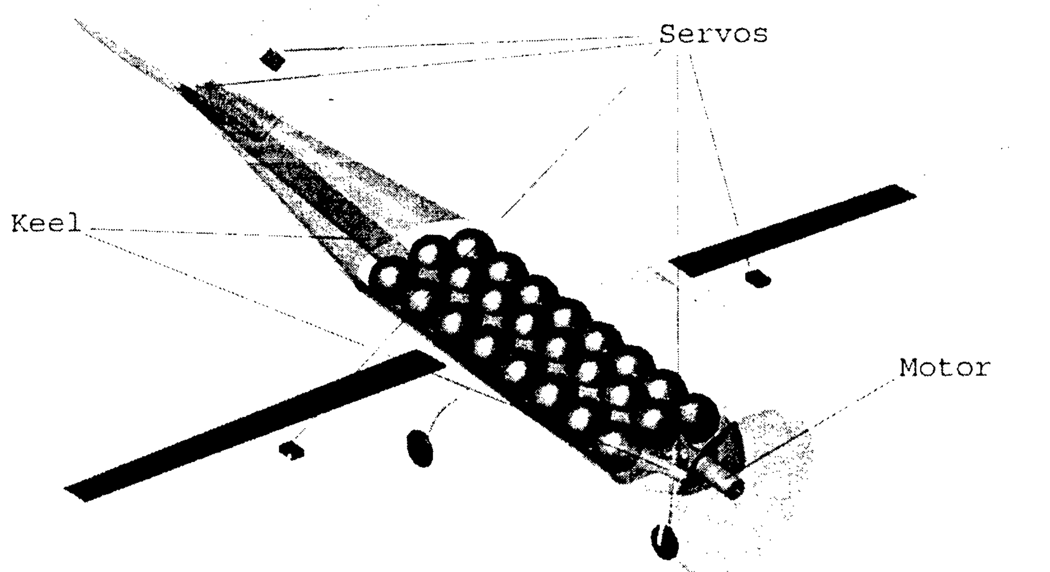
Table 5.4.

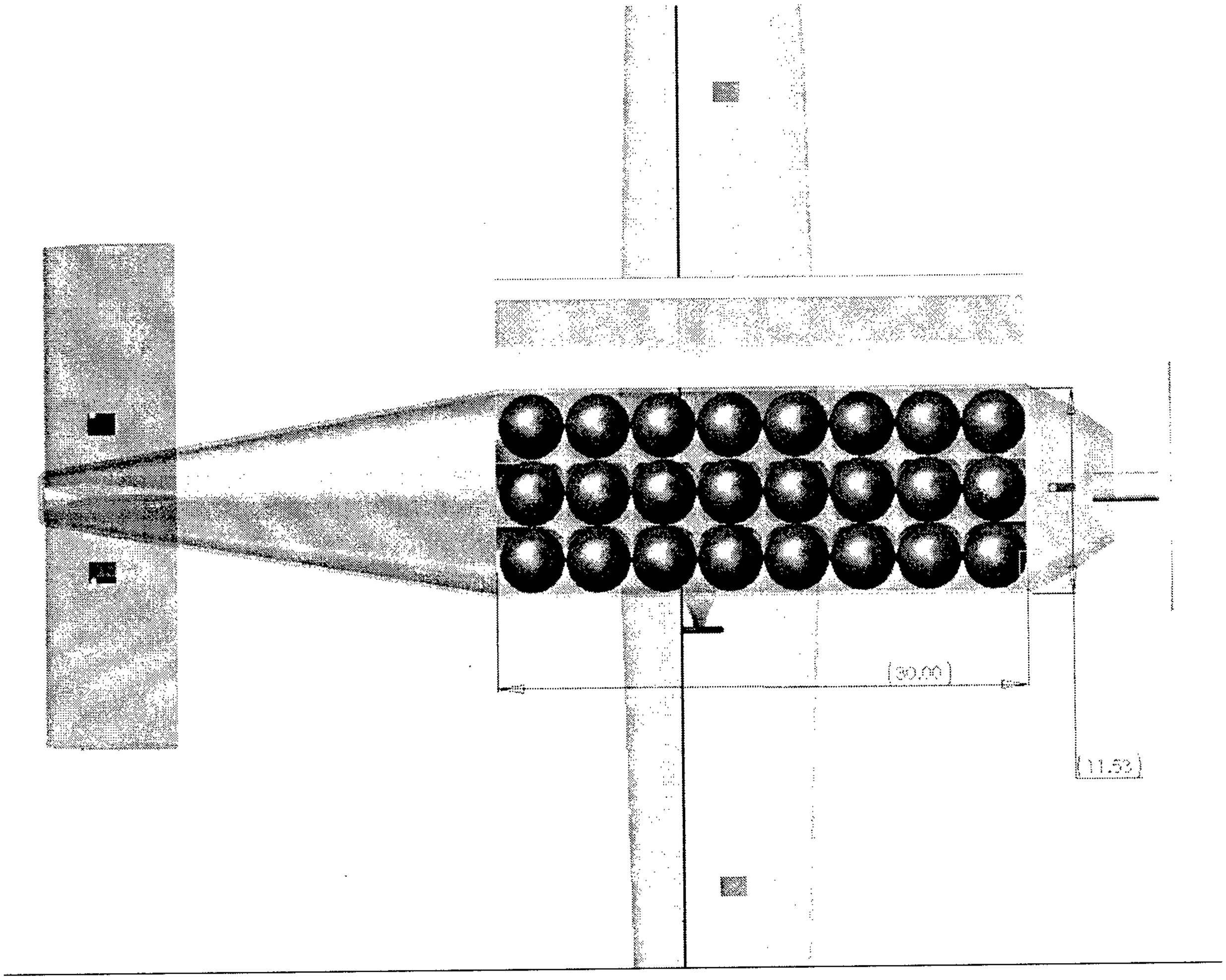
Component	***Weight (lb)***	***Arm (in)***	***Moment (lb*in)***
Fuselage	2.6	27.75	72.15
Balls	12	26	312
Astro. F. 40 + prop	1.3	5.2	6.76
Empennage	0.3	62.3	18.69
Main Gear	0.4	30	12
Nose Gear	0.5	10.5	5.25
Batteries	2.1	27.55	57.855
Wing	1.5	22.15	33.225
Wing Servos (2)	0.106	27.76	2.9495
V-Tail Servos (2)	0.106	63.4	6.73625
Gear Servo	0.053	9	0.478125
Receiver	0.063	42	2.625
Speed Controller	0.031	6	0.186
Main Gear Wheels	0.250	30	7.5
Nose Gear Wheels	0.094	10.5	0.984375
Totals:	21.402875		539.38925
Resultant CG (in):	25.20		

5.1.11. Rated Aircraft Cost

5.1.12. Drawing Package



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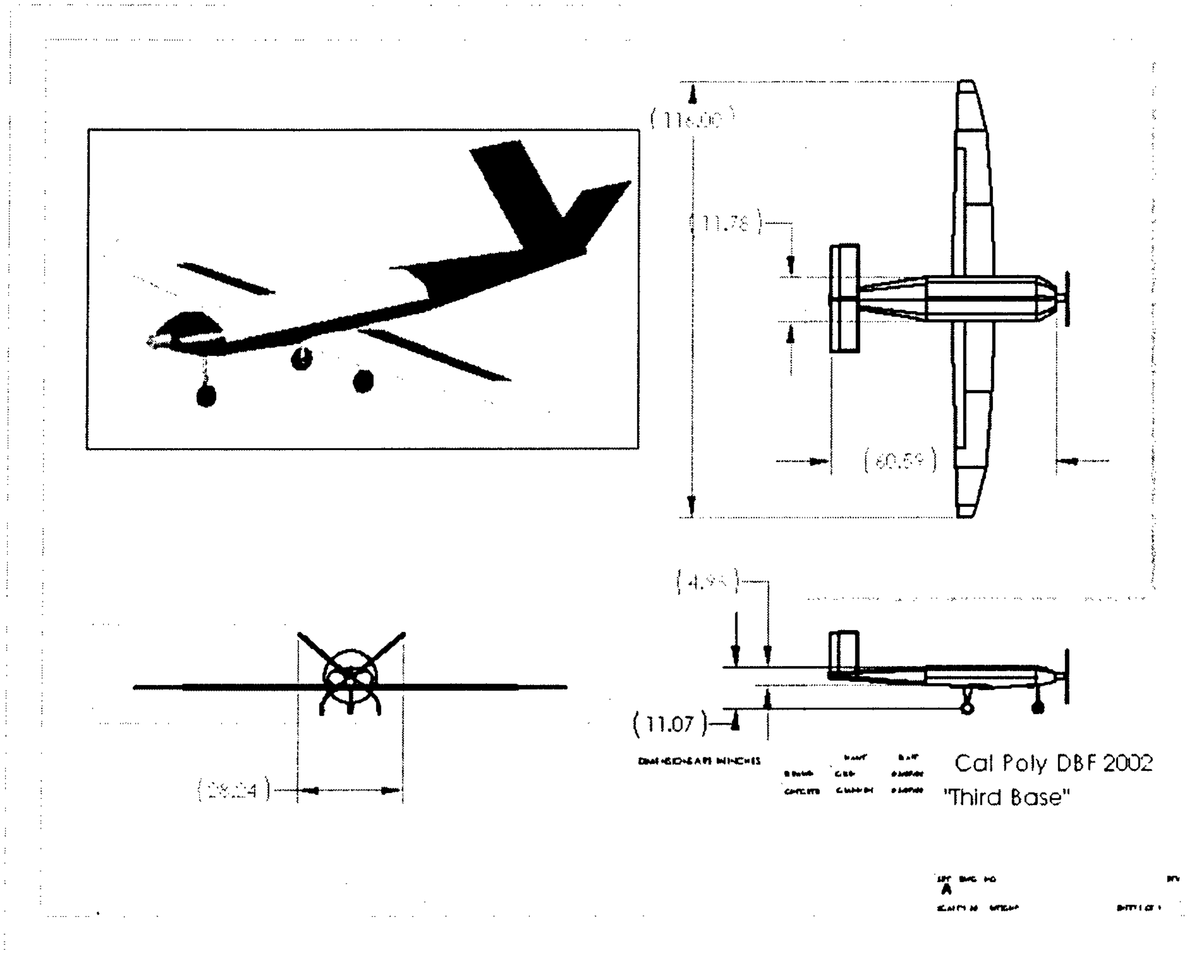

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Cal Poly DBF 2002
"Third Base"

6. Manufacturing Plan and Processes

6.1. Manufacturing Process for Major Components

Wing:

The wing will be made using standard vacuum bag foam core construction, wrapping a foam core with thin sheets of fiberglass and layers of carbon fiber arranged in the spanwise direction to act as the spar. The wing core will be constructed out of 2 lb density blue foam, commonly used for home insulation. Unidirectional carbon fiber will be placed on the quarter chord to resist bending moment in the wing. We selected this comparatively advanced method of construction despite of its higher level of complexity because of the potential it offers to builders experienced enough to utilize it. Although the method of construction is complex to learn; once learned, it delivers results that are superior in most respects to the alternatives.

The direction of the fiber will be placed from the root to the tip of the wing because this is the direction of the bending force and the fibers only have strength length wise. The fiberglass has fibers placed ninety degrees to each other and will be placed at a forty five degree angle with respect to the leading edge of the wing. The fiberglass will provide strength for the wing in torsion.
After the carbon fiber and fiberglass are attached with epoxy, the wing is placed in a vacuum bag. The vacuum bag is very convenient. It provides uniform pressure around the part while curing holding everything together and producing consistent parts.

Tail:

The tail will be made via foam core construction. The tail is made much like the wings, except that the fiberglass on the tail will be thinner than that on the wing.

Keel and Shell:

The major load-bearing component of the airframe will be a keel made out of dense foam (Rohacell) wrapped by multiple layers of epoxy-impregnated carbon fiber. The keel will run down the length of the fuselage, connecting the engine, wings, landing gear and empennage. The outside of the fuselage will be made up of a light non-load-bearing fiberglass shell in which the payload will be contained. The shell will be laid up over a foam blank which will then be hollowed out to make room for the softballs leaving foam in the tail boom and motor mounts. The shell will have a hinged door to allow for quick access to the softballs. A keel and shell configuration will be more compact and lightweight than a conventional balsa fuselage while being stronger. Using this method will simplify the creation of a fuselage that has good aerodynamics since the foam can be easily cut and sanded to any desired shape. As mentioned before, the team chose to use the materials and construction methods described above

because of the experience team members gained in previous DBF competitions working with composites. Had the team been composed of more inexperienced members, a less complicated manufacturing technique would have likely been selected.

Landing Gear:

The rear landing gear is a unidirectional carbon fiber arch. The bow configuration is basically a one-piece spring shock absorber. The carbon fiber arch is strong yet quite flexible so it translates the shock from landing into bending and shear stress in the landing gear diverting it away from the plane. The round shape of the landing gear also has the benefit of not having any stress concentrations because there are no corners.

6.2 Assemblies of the Final Design

The fiberglass shell will be attached to the keel with fiberglass. A door allowing access to the softballs will be hinged to the shell. The V-tail, wing, and landing gear will attach to the keel. The propeller and the motor will be placed in the nose. The speed control will be positioned near the motor. The rest of the electrical components will go wherever they fit as long as they are situated away from the motor, especially the receiver. Wires from the speed control and the servos will need to be directed to the receiver.

6.3. Alternative Manufacturing Processes and Figures of Merit

- Plain balsa wood, the built up construction, has many benefits. It is cheap, requires little skill, and few pieces of equipment. The problem is that it would not have the desired strength, especially in shear stress and is heavy.
- A foam only construction is simple, but would not provide enough strength.
- A straight boxy fuselage would be very strong, but would also be too heavy and too big, and not very aerodynamic.

One would think that the keel and shell method of construction compromises strength when compared to a fiberglass fuselage with load-bearing walls. In fact, we chose to not build the fuse using a construction with load-bearing walls because of the big hole that we planned on cutting in the top of the fuselage. This door would have seriously compromised the structural integrity of a fuselage built without a load-bearing keel. Another advantage of the keel-shell configuration is that it is much lighter and easier to modify, making it more aerodynamic and giving the airplane higher performance.

Design Alternative	Availability	Cost	Time Needed	Required Skill Levels	Strength to Weight	Durability	Easy Repair
Built Up Construction	H	L	M	L	L	M	L
Foam With Spars Wrapped in Tape	H	L	M	L	L	H	M
Foam With Fiberglass and Carbon Fiber	M	M	L	H	H	H	H
Wood Frame Covered With Wood Panel	H	L	M	L	L	M	L
Keel With Shell	M	M	M	M	M	H	H
Molded Shell	L	H	H	H	H	M	H

6.4 Analytic Methods

Describe the analytic methods (cost, skill matrix, scheduling time lines) used to select the final set of manufacturing processes.

	Friday 3/22/02	Saturday 3/24/02	Sunday 3/25/02	Monday 3/26/02	Tuesday 3/27/02	Wednesday 3/28/02	Thursday 3/29/02	Friday 3/30/02
Wing		Cut Foam Mainframe	Apply Carbon Fiber and Fiberglass	Attach to Keel				
Keel		Set Main Plan-Form	Apply Carbon Fiber	Prepare for other components				
Shell		Cut Foam	Apply Fiberglass	Shell out and Attach to Keel				
Tail		Cut Foam	Apply Carbon Fiber and Fiberglass	Attach to Body				
Landing Gear	Retrieve Landing Gear				Install Landing Gear			

Servos						Insert and Adjust		
Motor						Install and Adjust		
Receiver						Install and Adjust		
Speed Control						Install and Set		
Aesthetics							Apply	

Table 6.1 - Schedule of Events

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1 — Lennon, A., (1996), Basics of R/C Model Aircraft Design , P. 103

2 — http://www.elec-toolbox.com/formulas/motor/mtrform.htm

3 — Raymer, D., (1999), Aircraft design: a conceptual approach, Air Force Institute of Technology.

4 - http://www.charlesriverrc.org/articles/design/markdrela_vtailsizing.htm

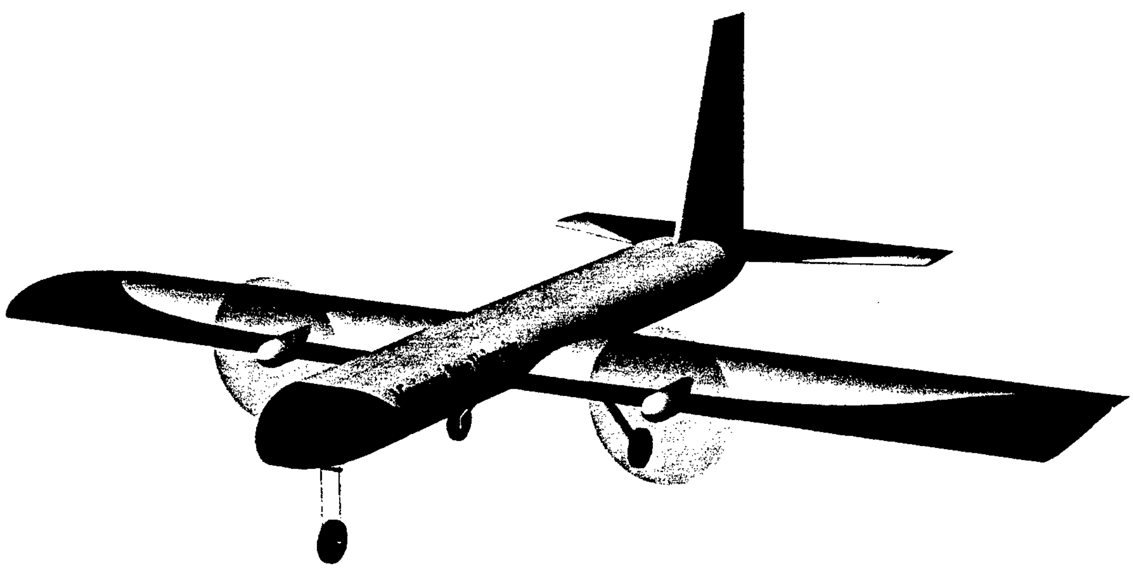
2001/2002 AIAA Foundation

Cessna/ONR Student Design/Build/Fly Competition

DESIGN REPORT
MARCH 12, 2002

Department of Aerospace Engineering

Mississippi State University



"FAST PITCH"

1. Executive Summary

1.1 Introduction

The *Fast Pitch* represents the design and construction efforts of 12 students from the Aerospace Engineering Department at Mississippi State University. Blending simplicity with performance, the airplane offers a highly manufacturable platform, which sacrifices very little mission capability. An electrically-powered, remotely piloted vehicle (RPV), *Fast Pitch* carries 24 softballs at speeds of up to 75 miles per hour. A summary of its performance and a picture of the design are presented below:

Table 1.1 Performance Summary

	Empty	Loaded
TOGW	20.6 lb	30.2 lb
Stall Speed (clean)	26 mph	28 mph
Stall speed (flaps)	23 mph	26 mph
Takeoff Distance	110 ft	140 ft
Climb Rate @ 40 mph	560 ft/min	450 ft/min
Max Cruise	85 mph	73 mph
Max Turn Rate @50 mph	120 deg/sec	83 deg/sec
Static Margin	23.5%	23.5%
Limit Load	±8 G	±5 G
Endurance @ 100% throttle	4 min 15 sec	3 min 49 sec

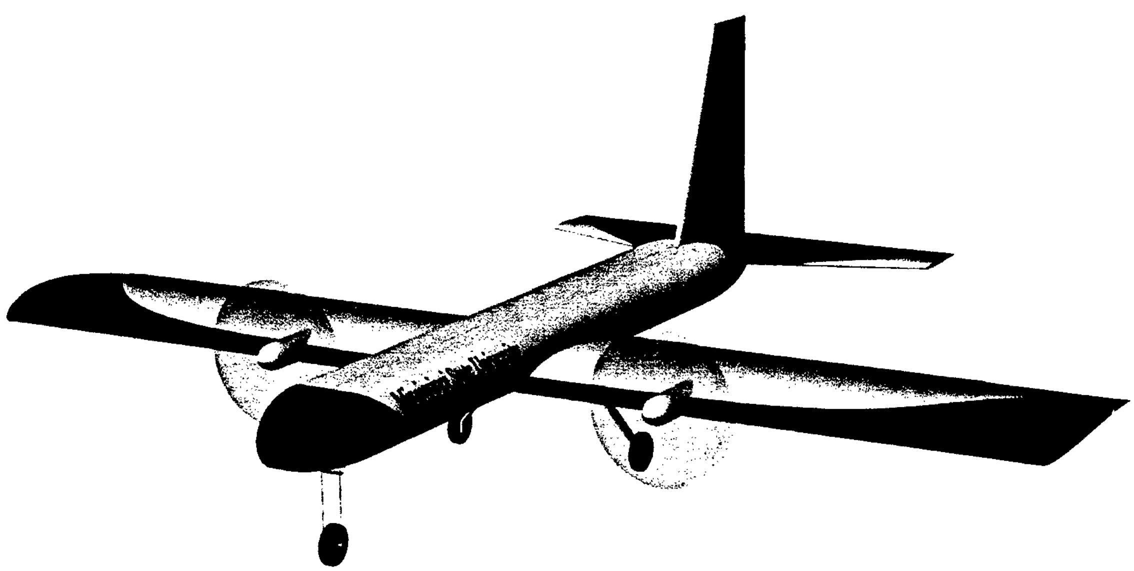


Figure 1.1 Perspective View of *Fast Pitch*

1.2 Design Development

The design of an aircraft such as this requires a long development process. Initially, possible configuration concepts must be created and evaluated for fitness to the overall design. This phase of design was carried out by the entire team, and eventually resulted in the development of 17 distinct configuration concepts. These concepts were evaluated using a series of analytical methods, and the results indicated that a "conventional" aircraft was most suited the task. This concept included the use of a low aspect ratio wing (4 to 7) a conventional fuselage, and either a conventional tail or a "V"-tail. This configuration was chosen not for any outstanding feature, but rather for its lack of an outstanding flaw. All "unconventional" aircraft had some feature that acted as an "Achilles' heel", dramatically reducing their fitness to the task. Additionally, all convential-winged designs utilized a constant-chord wing in order to maximize wing area for the cost specified in the competition rules. This also contributes to simplicity of manufacture.

Carrying forward the configuration selected in this phase, preliminary design of the aircraft was begun. This phase is where most of the actual design of the aircraft – at least from a performance standpoint – takes place. In this phase, the team performed trade studies on wing area, aspect ratio, number of engines, and engine type. It was immediately realized that in order to perform the mission at a relatively high speed, the aircraft would most likely be of a twin-engine configuration. This resulted from the use of higher-speed, lower-thrust engine configurations in order to raise cruising speed above 35-40 mph. These configurations, which utilize direct-driven propellers, do not offer as much static thrust as a geared-down system, but offer a great deal more speed, and if properly selected, can offer very high efficiencies for the airspeed ranges most common for the aircraft. However, a single-engine configuration utilizing these faster drivetrains has a very difficult time meeting the takeoff requirement. Upon completion of these trade studies, the team settled on an aircraft with a total wing area of 10.125 ft^2 and an aspect ratio of 5.6. The aircraft's propulsion system was selected to be two Astro Flight 60's driven by 33 Sanyo KR-1400AE cells each.

With this data in hand, it became possible to complete the detail design phase. During this phase, the sizing data given was transformed into an airplane. Utilizing a low wing with zero dihedral, and a tubular fuselage with space-shuttle-style top cargo doors, this airplane was also enhanced by such features as automatic doors and ground and in-flight braking systems. The tail was designed using a conventional approach, with horizontal and vertical tail volumes being obtained from tabulated historical data on similar aircraft types. Manufacturing techniques and materials for the aircraft were selected through a rigorous cost function based on manufactured weight and completion time, which resulted in a hybrid aircraft – part wood, part metal, part foam, and part advanced composites. A very simple and manufacturable

design, this aircraft sacrifices very little in terms of performance. While still retaining ease of construction, the performance enhancements added offer outstanding mission capabilities.

1.3 Design Alternatives Investigated

Throughout the design process, it was necessary to perform "reality checks" to make sure that the design path had not deviated substantially from the optimum. In the beginning, this meant evaluating a large host of design options. These options ranged from relatively standard configurations to deltas, canards, lifting bodies, and blended wing-body configurations. Initially, in fact, the team proceeded to the first phases of preliminary design on a configuration that was found to be lacking in takeoff performance. A concept for a lifting-fuselage airplane with a split "V"-tail seemed very promising until it was determined that the induced drag term for this airplane caused the power required for level flight at takeoff speeds to increase far beyond the power available. This led the team to select what it had thought to be an alternative design (that of a conventional airplane) as optimal.

Throughout the preliminary design phase, it was necessary to evaluate different power configurations, and finally select among them based on data points from the optimization routines. Astro Flight motor sizes from 15 to 90 were considered, and props ranging in diameter from 9 to 20 inches and in pitch advance from 4 to 20 inches were considered. While time-consuming, this broad analysis range allowed the team to select a drivetrain well suited to the mission. Further, wind tunnel testing was performed on the chosen motor (Astro Flight 60) with propellers similar to the analytically determined optimum. This data was then used to modify the results slightly, with the optimum propeller selection being changed from a 12x10 to a 12x6.

During detail design, the team evaluated myriad design methods, settling on a geometrically simple, yet high-performance aircraft. During this phase, manufacturing was also considered, with materials and methods from molded composites to hand-cut balsa being evaluated. The resulting aircraft blends high- and low-tech structures to create a strong, light, and manufacturable airframe. In fact, while maintaining good strength and manufacturability, this airplane achieves an empty weight fraction on the order of 0.5 – not at all bad for a configuration of this type.

After many months of evaluation, design, and redesign, the team emerged with what it feels to be an excellent airplane – one balanced very well between performance and manufacturability, with a simplicity of outward design that does not impede its capability to fly the mission, and fly it well.

1.4 Design Tools Used

1.3.1 Conceptual Design Tools

During this phase, the team developed an analytic model to perform very rough analysis on the weight, performance, and Rated Aircraft Cost (RAC) of each concept. This code, while quite rough, did in fact predict the specifics of the final aircraft reasonably well. Weight was developed from a part-by-part buildup of all structures present on the airplane, while performance and RAC were based on geometries determined to be necessary for flight.

1.3.2 Preliminary Design Tools

This phase of development relied on an expanded version of the Conceptual Design model for the configuration selected. By adding takeoff simulation, cruise speed and turn rate calculations, and optimization code, potential designs could be evaluated for performance. In addition, risk factors could be taken into account, as the optimum point was found to exist with the takeoff constraint active. Obviously, if the airplane were to gain weight through the construction phase, or the estimates of performance parameters were to be slightly erroneous, this could result in an aircraft incapable of flight, and therefore useless. In addition, for the development of thrust functions, both off-the-shelf software and newly developed analytic code were used, and found to agree reasonably well.

1.3.3 Detail Design Tools

For this phase, which included manufacturing design, the team expanded the weight model to develop structural weights for different components based on required strengths, material, and construction technique. Also, an analytic model based on prior experience was developed to evaluate time to complete these tasks. This led to the selection of manufacturing techniques and materials, which defined the final design for these assemblies.

2. Management Summary

2.1 Introduction

In the Fall of 2001, an interest meeting was conducted for the 2002 Design/Build/Fly competition. Because of last year's success in this competition, an overwhelming number of students were interested. This lead to two teams entering the competition from Mississippi State University: "Fast Pitch" and "Milk Run". The Design/Build/Fly Team "Fast Pitch" consisted of twelve members: three freshmen, three sophomores, two juniors, three seniors, and one graduate student – all majoring in aerospace engineering.

In their first official function, the team selected a faculty advisor and a team leader and divided into task-oriented groups. It was intended that the advisor would suggest on the design and manufacturing of the aircraft, and the team leader would perform the administrative duties for the team as wells as help with the designing and construction. The team held regular weekly meetings with each lasting approximately two hours.

2.2 Design Team Architecture

Because the overwhelming task ahead of the team, it was necessary to divide the team into groups based on their choice preference and skills. Each group had a leader that was responsible of informing the team leader of major developments and assigning tasks to other group members. Table 2.1 was generated to define the individual groups and responsibilities of each, and Table 2.2 was generated to show the involvement of each member in that group.

2.3 Scheduling of Major Events

The scheduling of major tasks are presented in Table 2.3. On most tasks, it was difficult for the team to keep on schedule. There are many factors that were involved with these delays. The major factor is the fact that the team is composed of college students that are constantly under the pressure of homework and examinations in their classes. Since this competition is extracurricular, some delays were inevitable. The major delay that the team experienced was with the motor/propeller tests to be conducted in a wind tunnel. The wind tunnel at Mississippi State University is constantly being used for classes and special projects. The only time available was during February.

Table 2.1 Team Architecture and Responsibilities

Group	Members Team Leader: Jeremy Sebens Faculty Advisor: Dr. Robert King	Responsibility
Finance and Accounting	Bryan Gassaway (GL), Michael Vitale, Jeremy Sebens	Maintaining and tracking of funds and the purchasing of materials.
Public Relations	Viva Austin (GL), Erin Wahlers, Chris Cureton	Generating press releases to the university and soliciting funds.
Documentation	Bryan Gassaway (GL), Erin Wahlers	Taking notes during meetings and collecting any material deemed necessary for documentation.
Scheduling	Erin Wahlers (GL), Joe Green	Informing the team leader of any tasks that are behind schedule.
Conceptual Design	Jeremy Sebens (GL), Joe Green, Peter Sakalaukus, Bryan Gassaway, Michael Vitale	"Brainstorming" ideas of possible airplane designs as well as issuing Figures of Merit.
Performance and Sizing	Joe Green (GL), Bryan Gassaway, Nathan Alday, Peter Sakalaukus	Sizing the aircraft based on performance issues and stability and control issues as well as issuing Figures of Merit.
Propulsion and Systems	Peter Sakalaukus (GL), Matt Phillips, Chris Cureton, Michael Vitale, Jennifer Esper	Researching the subject of thrust models for electric engines and experimental testing of potential motor/propeller combinations.
Structures and Design	Jeremy Sebens (GL), Viva Austin, Nathan Alday, Chris Cureton	Taking the sizing of the airplane from the performance group and putting in structure that would carry the loads.
Flight Testing	Bryan Gassaway (GL), Matt Phillips, Allen Hammack	Developing a flight plan for testing the flight envelope of the airplane, and obtaining an airfield to perform testing.
Construction	Matt Phillips (GL), Allen Hammack, Joe Green, Erin Wahlers (team is considered to help with the construction)	Scheduling of manufacturing and building of the airplane.
Report Generation	Jeremy Sebens (GL), Bryan Gassaway, Jennifer Esper, Joe Green, Erin Wahlers	Writing the report.

GL -- Group Leader

Table 2.2 Involvement of Each Member

	Jeremy Sebens	Bryan Gassaway	Joe Green	Michael Vitale	Viva Austin	Erin Wahlers	Chris Cureton	Nathan Alday	Matt Phillips	Peter Sakalaukus	Jennifer Esper	Allen Hammack
Finance and Accounting:												
Budget	5	5	0	2	2	0	0	0	0	0	0	0
Purchasing	5	0	0	0	0	0	0	0	0	0	0	0
Manufacturing:												
Wing Assembly	5	2	1	0	2	2	0	2	5	2	0	0
Fuselage Assembly	4	3	2	2	2	2	2	3	5	1	3	3
Horizontal and Vertical Tail	5	0	0	0	3	0	0	0	5	0	3	3
Systems Integration												
Report:												
Executive and Management	5	5	0	0	0	0	0	0	2	0	0	0
Conceptual Design	5	4	0	0	2	2	0	0	0	0	0	0
Preliminary Design	3	5	0	2	0	0	0	0	0	0	0	0
Detail Design	5	3	0	0	0	0	0	0	2	0	0	0
Manufacturing Plan	5	4	0	0	0	0	0	0	3	0	0	0
Flight Testing:												
Pilot	0	0	0	0	0	0	0	0	5	0	0	0
Ground Crew	5	5	2	2	2	2	2	2	5	2	2	2
Propulsion:												
Thrust Model	5	0	0	3	0	0	3	0	3	3	3	0
Motor/Propeller testing	3	3	0	0	3	3	0	3	3	3	3	3
Performance:												
Sizing and Optimization	4	4	3	0	0	0	0	2	0	1	0	0
CG and Neutral Point Analysis	3	3	0	0	3	3	0	0	0	0	0	0
CAD:												
Components	5	3	0	0	0	0	0	0	2	0	0	0
Engine Mounts	5	3	0	0	0	0	0	0	2	0	0	0
Wing	5	3	0	0	0	0	0	0	2	0	0	0
Fuselage	5	3	0	0	0	0	0	0	2	0	0	0
Horizontal and Vertical Tail	5	3	0	0	0	0	0	0	2	0	0	0
Stability & Control:												
Horizontal and Vertical Tail Sizing	4	0	0	0	3	3	0	0	0	0	0	0
Servo Selection	4	0	0	0	0	0	0	0	4	0	0	0
Control Surface Sizing	4	1	0	0	0	0	0	0	4	0	0	0
Administration:												
Scheduling	4	2	2	2	3	3	2	2	4	2	2	2
Soliciting Funds	4	2	0	0	4	0	0	0	0	0	0	0
Documentation	0	4	0	0	0	4	0	0	0	0	0	0

0 – No Involvement 5 – Maximum Involvement

Incomplete activity

Table 6.3 Milestone Chart of Major Events

Task Name	Start	Finish	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr
Conceptual Design										
Planned	18-Sep	2-Oct								
Actual	20-Sep	10-Oct								
Propulsion Model and Testing										
Planned	16-Sep	27-Nov								
Actual	26-Sep	18-Feb								
Submission of Entry Form		31-Oct								
Performance and Sizing										
Planned	27-Sep	23-Oct								
Actual	27-Sep	30-Oct								
Obtain Half Funding										
Planned	18-Sep	25-Oct								
Actual	17-Sep	20-Sep								
Detailed Design										
Planned	26-Oct	30-Nov								
Actual	1-Nov	10-Jan								
Manufacturing										
Planed	10-Jan	28-Feb								
Actual	2-Feb	22-Mar								
Report										
Planned	12-Feb	5-Mar								
Actual	10-Feb	10-Mar								
Submission of Report		12-Mar								
Design/Build/Fly Competition		26-Apr								

3. Conceptual Design

3.1 Introduction

The conceptual design phase of the aircraft design process defines the "look" of an aircraft. The rough layout of the aircraft is defined during this phase, and many of the major design decisions that will later affect the airplane's performance must be addressed, at least in part. This phase was carried out by the entire team, with members suggesting configurations until a list of 17 different layout concepts were developed.

3.2 Configuration Concepts

Upon examination, it was determined that three major features defined each of the design concepts presented: fuselage, wing, and tail (empennage). Open suggestions taken from the entire team resulted in the following table of design configurations:

Table 3.1 Concept Descriptions

Concept Number	Fuselage	Wing	Empennage
1	Conventional	High AR	Conventional
2	Conventional	High AR	V
3	Conventional	High AR	T
4	Conventional	High AR	Canard
5	Conventional	Low AR	Conventional
6	Conventional	Low AR	V
7	Conventional	Low AR	T
8	Lifting Body	High AR	Conventional
9	Lifting Body	High AR	V
10	Lifting Body	High AR	None
11	Lifting Body	Low AR	Conventional
12	Lifting Body	Low AR	V
13	Lifting Body	Low AR	None
14	Lifting Body	Swept High AR	None
15	Lifting Body	Forward Swept	Conventional
16	Blended	Delta	None
17	Blended	Swept	None

3.2.1 Fuselage Concepts

The fuselage design was categorized as being either conventional, lifting body, or blended body. The conventional fuselage consists of tubular cross-sections. The lifting and blended body fuselages are designed such that the fuselage contributes to the overall lift of the aircraft. The difference between a

lifting body and a blended wing body is noticeable. The lifting body has a distinct break between the fuselage and wing, but the blended fuselage lacks this distinction.

3.2.2 Wing Aspect Ratio

The wing configuration was categorized as being either high aspect ratio or low aspect ratio. A low aspect ratio wing was assigned a value between 4 and 7, and a high aspect ratio wing was assigned a value above 7. High aspect ratio wings have the advantage of having less induced drag when compared to low aspect ratio wings, but high aspect ratio wings suffer in performance and handling qualities due to the extra inertia and roll damping as well as a tendency to tip stall; whereas, low aspect ratio wings have better performance and handling qualities.

3.2.3 Empennage Configurations

The empennage was categorized as being either conventional, "V", "T", or none. A conventional tail consists of a horizontal and vertical stabilizer with the horizontal stabilizer located on the fuselage. A "T"-tail appears like a conventional tail, but the horizontal stabilizer is located above the fuselage. A "T"-tail is better than the conventional tail because at small angles of attack the downwash from the wing is negligible, but during high-angle of attack maneuvers the "T"-tail could be blanketed from the wing and thus losing its effectiveness. The "V"-tail has less wetted area when compared to conventional tails, but it is less efficient in pitch inputs and does require radio mixing. For no empennage to be added, there must be sufficient sweep in the wing to allow elevons to be utilized

3.3 Figures of Merit

3.3.1 Design Parameters Investigated

Each of the configurations investigated was ranked based on a series of Figures of Merit. The Figures of Merit or design parameters chosen for this phase are speed, performance and handling, takeoff performance, and rated aircraft cost. The parameters are weighted based on their importance to the mission from a value of 5 (highest importance) to 1 (lowest importance). The Figures of Merit chosen, and the mission features affected by each, are detailed in the following table.

Table 3.2 Design Parameters and Weighted Value

Design Parameter	Weight Factor	Mission Feature
Performance and Handling	5	Handling qualities of aircraft, flyability for ease of piloting on closed course
Speed	4	Time to complete entire airborne portion of mission
Takeoff Performance	3	Ease of takeoff within 200 ft
Rated Aircraft Cost	-1	Rated Aircraft Cost

The first important design parameter was performance and handling. This quality was deemed very important due to the fact that the pilot has to feel comfortable with the airplane. If the airplane has a very low performance and very heavy on the sticks, the pilot will be "over-working" the airplane as well as becoming frustrated. It was deemed very important to keep the pilot satisfied with the airplane at all times.

Since this competition is timed, speed is a necessity. One of the variables that effects the speed is the drag. Drag is composed of two terms: parasite (friction) drag and induced (pressure) drag. The influence of the parasite drag is great at high speeds and is low at low speeds. The influence of the induced drag is opposite. Induced drag is high at low speeds and is low at high speeds. Since most of the time the airplane is operating at the high end, parasite drag must be considered. The parasite drag can be estimated by comparing the exposed surface area ("wetted area"). If the wetted area is less, than the parasite drag would be less.

One of the restrictions place on the competition is that the aircraft must takeoff within 200 feet. During ground roll and takeoff the aircraft is operating at low speeds. At low speeds induced drag is high. Lifting bodies tend to suffer with induced drag due to the low aspect ratio of the fuselage body, which is essentially a wing. Therefore, lifting bodies are severely penalized when compared to the conventional fuselage.

The last quality, RAC, even though last is considered to be an important design criterion. The objective in any competition is to win, and to win the overall score must be greater than the opponents'. The RAC is a penalty in the overall score. The higher the rated aircraft cost the less the overall score will be. This is the reason why the weighing factor is negative. It acted as a penalty in the ranking of the design concepts.

3.3.2 Scoring of Design Concepts

Based on the design parameters discussed, each design concept was evaluated by giving a score of 5 (excellent) to 1 (poor). The scores from these design parameters were multiplied by their corresponding weighting factor and summed to obtain the overall score for each concept. In other words,

$$\text{TOTAL} = \sum_{i=1}^{3} s_i \cdot WF_i$$

where s and WF are the score and weighting factor, respectively, for each design parameter. Evaluating each configuration, the following table was generated. The top total scores were further considered.

Table 3.3 Scoring of Design Concepts

Concept	Speed	Perf/Handling	RAC	Takeoff	TOTAL	Action
1	3	2.5	15.382	3.5	19.618	E
2	3	2.5	15.377	3.5	19.623	E
3	3	2	15.46	4	18.54	E
4	3	2	15.382	3.5	17.118	E
5	3	4	15.086	3.5	27.414	FC
6	3	4	15.084	3.5	27.416	FC
7	3	3	15.382	3.5	22.118	E
8	3.5	2.5	14.67	4	23.83	E
9	4	4	14.664	1.5	25.836	E
10	4.5	1	14.57	1.5	12.93	E
11	3.5	2.5	14.32	1.5	16.68	E
12	4	4	14.314	1.5	26.186	E
13	4.5	1	14.18	1	11.82	E
14	4.5	3.5	14.18	1.5	25.82	E
15	3.5	0.5	14.33	2	8.17	E
16	5	2	12.1	2	23.9	E
17	5	2	13.12	2	22.88	E

E - Eliminate
FC - Further Consideration

3.4 Factors affecting Rated Aircraft Cost

Rather than discussing individual designs, we will describe the effect of different types of components. These effects are presented in the following table.

Table 3.4 Rated Aircraft Cost Factors

Component	Type	Effect on RAC
Fuselage	Conventional	Longer than other two types, so costs more Added length required additional structural weight
	Lifting	Shorter length costs less Lower structural weight
	Blended	Same as Lifting
Wing	High AR	Higher cost per square foot due to cost formula Greater structural weight
	Low AR	Lower cost per square foot Reduced structural weight
	Swept	Same as High AR
	Delta	Same as Low AR
Tail	Conventional	No effect
	V	Slightly reduced cost from formulas
	T	No effect
	Canard	Increased cost due to necessity to score as a wing.

3.5 Assumptions Made

In order to score the individual concepts it was necessary to hold all other factors constant. In essence, the assumption made to this end was that the score for an airplane capable of flight carrying 24 balls with twin Astro Flight 40 engines could be extrapolated to other loading configurations and powerplant options. The team felt that this assumption was valid since any aircraft would most likely be scaled the same amount for a given change in loading and powerplant.

3.6 Features of Final Design Selection

The design that emerged from the conceptual design phase was that of a conventional airplane with a low Aspect Ratio wing (AR of 4 to7) and either a conventional or a V tail. The empennage decision was felt to be minor enough to be deferred to a later design phase, but the basic configuration was locked in. The features that led to this design's selection were not anything that was vastly superior to all other designs, but rather its lack of an "Achilles' heel", something possessed by all other concepts investigated. For example, the Low AR Lifting body with a split V-tail was initially quite attractive, but suffers in low speed flight because of the tremendous induced drag developed by placing the lifting body (essentially a wing with AR=0.4) at an angle of attack. Canard types do not offer enough advantage to overcome the risk associated with their use for stability and control purposes. A delta, while very simple and cost-effective,

also suffers for low speed flight because of the necessity to essentially "decamber" the wing in order to achieve high angle of attack flight, lowering C_{Lmax} substantially. The fact that a conventional airplane lacks a negative feature like these led to its eventual selection. Finally, a Low AR wing was chosen over a High AR wing because of the simplification of manufacture, as the score difference was actually quite small.

4. Preliminary Design

4.1 Introduction

In designing an airplane, there are many variables to consider such as engine selection, number of engines, empty weight, wing surface area, wing aspect ratio, wing chord, and airfoil selection. Since there are many combinations of these variables to produce an aircraft that flies, a method must be developed to rate these configurations as a potential winning design. This method involves a trade study based on the number of engines, engine/propeller selection, wing surface area, and wing aspect ratio. These design parameters are substituted into the RAC matrix specified from the competition rules, and comparing the scores of these possible configurations will determine the best design.

4.2 Design Parameters and Sizing Trades

4.2.1 Airfoil Selection

Given the varied nature of the mission to be performed, airfoil selection is not a trivial task. The selected airfoil must have a high C_{Lmax}, but must also have good drag characteristics at cruising C_Ls. After evaluating several candidate airfoils, the team settled on the Selig-Donovan SD7062 airfoil. This airfoil offers excellent lift characteristics, without substantial drag penalty. Also, the team had excellent experiences with this airfoil in previous designs.

For the tail sections, the NACA 0009 was used. This airfoil section is in common use for tail surfaces, and offers very predictable performance.

4.2.2 Engine Selection

Selecting a power plant was not an easy engineering decision. The thrust is a function of the engine/propeller selection and the number of cells used to power the engine. Further, the endurance of the engine is a function of the current draw of the engine/propeller combination and the cell pack. Astro Flight tabulates some engine performance data for various engine/propeller combinations at there website. Unfortunately, this data was found to be somewhat unrealistic in previous designs, and cannot be taken at face value. In lieu of this, over 60 various battery/engine/prop combinations were evaluated using both off-the-shelf analysis tools and analytic methods. The resulting best configurations for several motors are presented below.

Table 4.1 Engine Performance Data for Optimal Engine/Propeller Configurations

	Astro15 10x9	Astro25 11x9	Astro40 12x8	Astro60 13x10	Astro90 16x10
Max. Amp draw	29	30	40	25	22
Number of cells	28	33	33	33	33
Static Thrust (oz)	65	66	72	88	108
Max. speed (mph)	103	108	98	109	89
Weight (oz)	8.5	11	12.5	22	32

4.2.3 Weight Model

In actuality, there is no "standard" empty weight model for an electric-powered, RPV with a maximum weight of fifty-five pounds. To solve for the empty weight of the RPV, an idea of how to build the airplane must be known. Then, the basic components of the RPV can be itemized for weight.

It can be shown that the empty weight can be solved as a function of the number of engines, engine/propeller selection, wing surface area, and wing aspect ratio. In other words,

$$W_{empty} = f(N_{engines}, Engine_{type}, S, AR)$$

Further, the above expression can be expanded to include terms for aircraft components, i.e. fuselage, empennage, wing, landing gear, electrical equipment, and engines.

$$W_{empty}(N_{engines}, Engine_{type}, S, AR) = W_{fuselage} + W_{empennage}(S) + W_{wing}(S, AR) + W_{gear} + W_{electrical} + W_{engines}(N_{engines}, Engine_{type})$$

The cargo consists of softballs with each weighing at a estimated three-eighths of a pound. When multiplying this weight by the number softballs to be carried, the cargo weight can be found. The 2002 Design/Build/Fly competition rules state the minimum number of softballs to be carried is 10 and the maximum number of softballs is 24. By carrying more softballs, the flight score increases. It was assumed that the maximum number of softballs is to be carried, which is consistent with the assumption made in the Conceptual Design phase. Thus, the cargo weight is nine pounds. By adding the cargo weight to the empty weight of the RPV, the takeoff-gross weight, TOGW, of the RPV can be calculated.

$$TOGW(N_{engines}, Engine_{type}, S, AR) = W_{empty}(N_{engines}, Engine_{type}, S, AR) + 9\text{ lb}$$

4.2.3 Takeoff Equations

From the rules of the competition, the performance stipulation on the aircraft is that it must takeoff within 200 feet. Therefore, the performance analysis of the aircraft will be based from takeoff equations.

By summing the forces parallel to the ground, and employing Newton's second law,

$$m\frac{dV}{dt} = T_{avl} - D - \mu_r \cdot (TOGW - L)$$

where T_{avl} is the thrust available from the engine, D is the aerodynamic drag, L is the aerodynamic lift, m is the mass, and μ_r is the coefficient of rolling resistance of the runway. Solving for dV/dt or the acceleration,

$$a = \frac{dV}{dt} = \frac{T_{avl} - D - \mu_r \cdot (TOGW - L)}{m}$$

The aerodynamic forces of lift and drag are expressed as

$$L(V,S) = C_L(\alpha) \cdot 0.5 \cdot \rho \cdot V^2 \cdot S$$

$$D(V,S,AR) = \left(C_{D0} + \frac{1}{\pi \cdot AR \cdot e} C_L{}^2\right) \cdot 0.5 \cdot \rho \cdot V^2 \cdot S$$

where C_{D0} is the zero-lift drag, ρ is the density of air, e is the Oswald efficiency factor, and α is the angle of attack (AoA). The zero-lift drag and Oswald efficiency factor were estimated as 0.022 and 0.8, respectively. The lift coefficient, C_L, during the ground roll was assumed to be the same as the approximate cruise lift coefficient of 0.4.

The thrust available from the engine can be approximated by the observation of the engine's characteristics. With a given engine/propeller selection, the engine has its highest thrust, T_{static}, when the RPV is not moving, and the engine has maximized its velocity, V_{max_eng}, when there is no more thrust available. From this relationship, the thrust available from the engine was assumed to be linear with velocity and also depends on the number of engines and engine/propeller selection. This assumption, while not exact, results in an error of less than 5% for all cases evaluated. In addition, this error is on the side of caution, as the actual thrust curve is concave downward. In any case, this linear relationship can be expressed as

$$T_{avl}(V, N_{engines}, Engine_{type}) = N_{engines} \cdot \left(T_{static}(Engine_{type}) - \frac{T_{static}(Engine_{type})}{V_{max\_eng}(Engine_{type})} \cdot V \right)$$

Once all these expressions are combined into the acceleration equation, it can be shown that the acceleration is a function of the velocity, number of engines, engine/propeller selection, wing surface area, and wing aspect ratio. In other words,

$$a = \frac{dV}{dt} = f(V, N_{engines}, Engine_{type}, S, AR)$$

Utilizing an Euler integrator algorithm, the liftoff velocity, V_{LO}, can be calculated at 180 feet (a factor of safety of 11%) using the expression for acceleration. By comparing V_{LO} and the stall speed, V_{stall}, using a factor of safety of 1.2, the aircraft's takeoff capability can be evaluated.

4.2.5 Sizing Equations

There are several parameters that are important when sizing an aircraft. For the wing, these parameters include the wing span, surface area, aspect ratio and chord length. Keeping to the standard of setting variables as functions of surface area and aspect ratio, the span and chord length can be expressed as

$$b(S, AR) = \sqrt{AR \cdot S}$$

$$c(S, AR) = \frac{b(S, AR)}{AR}$$

For the empennage, these parameters are horizontal and tail surface area. To calculate these parameters, it was assumed that the horizontal tail and the vertical tail surface areas are 20% and 10% of the wing surface area. These rough initial estimates were based on tabulated historical data from model aircraft.

4.2.6 Scoring of Configurations

The scoring of a configuration is evaluating using a modified version of the overall score for the competition. The overall score of the competition is a combination of the Report Score (RS), Flight Score (FS), and Rated Aircraft Cost (RAC). This is expressed as

$$SCORE = \frac{RS \cdot FS}{RAC}$$

Since the RS is a just a multiplier and the FS is a constant (number of softballs are fixed at the maximum and an assumed mission completion time), a new modified score can be expressed.

$$MODIFIED\ SCORE = \frac{const.}{RAC}$$

The RAC is composed of the empty weight, rated engine power (REP), and the manufacturing hours (MFHR) function. The matrix that defines these components can be referenced in the competition rules, but the main point to be made is that the RAC can be expressed as a function of the number of engines, engine selection, wing surface area, and wing aspect ratio. In other words,

$$RAC = f(N_{engines}, Engine_{type}, S, AR)$$

There were several assumptions made for the calculation of the MFHR function. The fuselage length was 5.3 feet, and the number of servos was five.

4.2.7 Results of Trade Study of Possible Configurations

The expressions developed are given as functions of the configuration such as the number of engines, engine/propeller selection, wing surface area, and wing aspect ratio. For a given number of engines and engine/propeller selection, a three-dimensional analysis can be done on a configuration based on the wing surface area, wing aspect ratio, and modified score with the stipulation that the RPV must takeoff within 180 feet. If the RPV cannot takeoff within 180 feet a score of zero is given.

Varying the wing surface area and wing aspect ratio from 4 to 13, the RPV cannot takeoff with a single engine for the possible engine/propeller combinations, but can takeoff using twin engines. Also, the modified score of the possible engine/propeller combinations ranges from 0.370 to 0.395 with no actual peak in the contours to indicate an optimal design. A contour plot of the twin Astro Flight Cobalt 60 motor is shown in Figure 4.1, and the trends from this plot are similar to other engine selections. From this plot,

it can be noted that the design that gives the highest modified score lies on the boundary between a successful takeoff and a scrubbed mission. Since this design point is unacceptable, the design point was chosen to be in a "safe" region. Table 4.1 shows the results of the possible configurations. The values presented in this table represent the required "safe" values for takeoff and flight, and are considered optimal. In addition to performance and geometry, the resulting wing and power loading requirements are presented in this table.

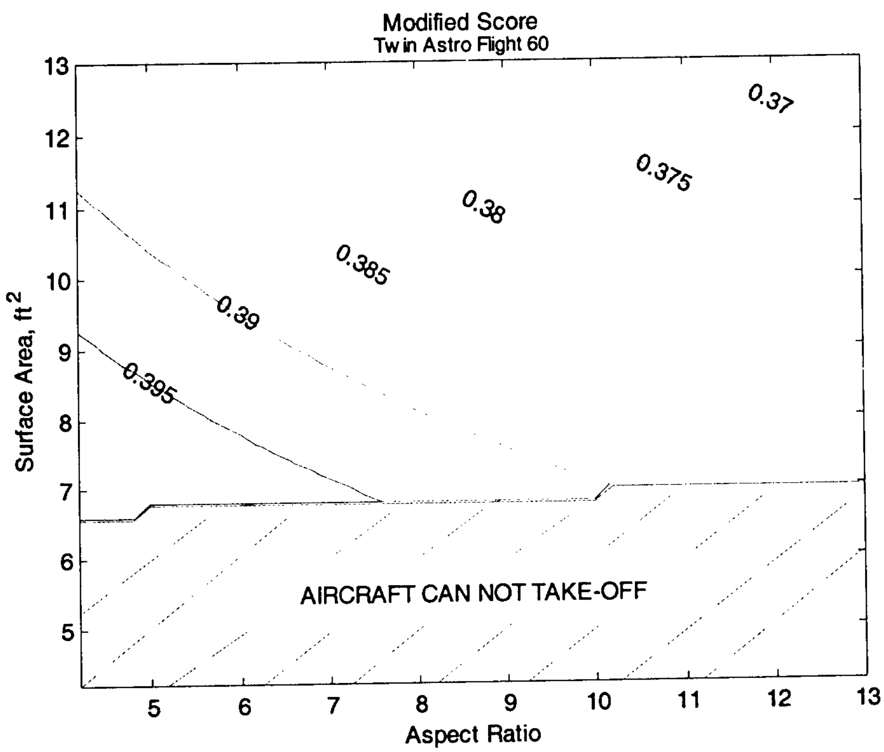


Figure 4.1. Contour Plot for Twin Astro Flight Cobalt 60

Table 4.1. Results of Trade Study of Possible Configurations

Design Concept	Engine/Propeller (twin)	Score	Empty Weight (lb)	Wing Loading (lb/ft^2)	Power Loading (lb/Watt)	S (ft^2)	AR	Vmax of RPV (mph)
1	AstroFlight 15	0.367	17.32	2.45	0.131	11.6	5.6	56
2	AstroFlight 25	0.374	17.11	2.48	0.138	11.4	5.6	61
3	AstroFlight 40	0.38	17.03	2.67	0.146	10.6	5.6	74
4	AstroFlight 60	0.39	16.54	2.83	0.143	10.2	5.6	81
5	AstroFlight 90	0.36	18.52	2.2	0.136	11.1	5.6	62

4.3 Figures of Merit

The selection of the best design was based on the Figures of Merit (FOM). The FOM are parameters that are considered important in the design. These parameters are maximum speed of the airplane (not to be confused with the maximum speed of the engine), the modified score, and the ratio of the liftoff velocity at 180 feet and the velocity at stall. Since the competition is a timed event, the maximum speed of the RPV is important. The maximum speed of the airplane can be found by finding the intersection of the power available and power required curves.

The FOM scores were tabulated according to the formula

$$FOM\ Score = \sum_{i=1}^{6} s_i \cdot WF_i$$

where s_i and WF_i are the score and weight factor for the i^{th} FOM, respectively. The scores are shown in Table 3.2

Table 3.2 FOM Scoring

	Design Concept				
FOM (Weight Factor)	1	2	3	4	5
Vmax (2)	1	2	4	5	3
Modified Score (2)	2	3	4	5	1
V_{180ft}/V_{stall} (1)	1	2	3	5	4
Total	**7**	**12**	**19**	**25**	**12**

4.4 Resulting Configuration

From the analysis performed in this phase, a final set of sizing data was put forward to the detail design phase. This configuration called for a twin-engine airplane powered by Astro Filght 60's with a wing area of 10.125 ft^2, an aspect ratio of 5.6, and capable of carrying 24 balls. This configuration represents a safe point on the design curve, while not sacrificing much score.

5. Detail Design

5.1 Performance Data

The final configuration for *Fast Pitch* represents a moderately high-performance electrically-powered aircraft. Final performance specifications are given in the following table:

Table 5.1 Performance Data

	Empty	Loaded
TOGW	20.6 lb	30.2 b
Stall Speed (clean)	26 mph	28 mph
Stall speed (flaps)	23 mph	26 mph
Takeoff Distance	110 ft	140 ft
Climb Rate @ 40 mph	560 ft/min	450 ft/min
Max Cruise	85 mph	73 mph
Max Turn Rate @50 mph	120 deg/sec	83 deg/sec
Static Margin	23.5%	23.5%
Limit Load	±8 G	±5 G
Endurance @ 100% throttle	4 min15 sec	3 min 49 sec

The handling qualities of the aircraft were evaluated from the standpoint of static stability. With a positive static margin, the airplane is obviously reasonably longitudinally stable. The aircraft is somewhat unstable in the spiral mode, due to the low-wing, zero-dihedral configuration and larger vertical tail, but the time to double amplitude was calculated to be approximately 36 seconds, far longer than the aircraft will be left unattended in any phase of the mission. This behavior was, as a result, considered acceptable. Dynamic stability analysis was not performed, as these calculations do not always resemble real-world performance, and as the aircraft will never be out of visual control at any time. If a signal loss were to occur, the fail-safe mode required would eliminate the possibility of such a condition. In order to eliminate the possibility of an underdamped dutch roll, a large value of the vertical tail volume coefficient was used.

5.2 Estimated Mission Performance

The aircraft development process involved a great deal of time spent on improving mission performance. This was primarily accomplished through the integration of features used in terminal flight phases and ground handling, and is further documented in section 5.4. The addition of these enhancements improved projected mission time by approximately 45 seconds. The estimated timetable for a complete mission profile is presented below:

Table 5.2 Mission Times

Mission Phase:	Time to Complete
2 unloaded laps, including 360 degree turns on downwind	1:12
Load payload, reset airplane at starting line	0:35
2 loaded laps, including 360 degree turns on downwind	1:30
Unload payload, reset airplane at starting line	0:35
2 unloaded laps, without 360 degree turns on downwind	0:56
TOTAL TIME	**4:48**

5.3 Weight and Balance Sheet

The weight and balance sheet references all x-locations in inches from a datum line located at the interface between the fuselage and nose cone.

Table 5.3 Weights and Balances

	x (in)	W (lb)	x*W (lb*in)
Vertical Tail	58.6	0.4	23.44
Horizontal Tail	62.9	0.6	37.74
Tail Box	56.47	0.2	11.294
Fuselage	26	4	104
Wing	26	4	104
Batteries	22.9	4.8	109.92
Motors	19.5	2.75	53.625
Main Gear	30.5	0.7	21.35
Nose Gear	0	0.6	0
Tail Servos	52	0.225	11.7
Wing Servos	32	0.45	14.4
Nosewheel Servo	-0.5	0.13125	-0.06563
Radio battery	-0.5	0.25	-0.125
Receiver	25.5	0.09375	2.390625
Door Servos	25.5	0.2625	6.69375
Wheel Brakes	30.5	0.3	9.15
Wiring	23	0.2	4.6
Wing Joiner	24	0.7	16.8
W (lb)	**20.66**		
Aero. Center of Wing (in)	**25.5**		
Neutral Point (in)	**30.0**		
C.G. (in)	**25.69**		

5.4 Component Selection

All off-the-shelf (OTS) components to be used on the aircraft were carefully selected from available parts. A listing of components selected follows, including comments on each.

Table 5.4 Component Selection

Component	Selected Part	Comments
Motors (2)	Astro Flight 661	Selected by preliminary design optimization
Speed Controls (2)	Astro Flight 204D	Appropriately sized for selected motor
Battery Packs (2)	33 Sanyo KR-1400AE cells each	Offer best available energy density in a moderate-resistance cell. 33 cells chosen to keep packs below 2.5 lb each
Servos (control surfaces)	Futaba S9151 Digital	Offer both high speed and high torque
Servos (ground-active)	FMA S355M	High torque (Speed not needed for these applications)
Main Landing Gear	TNT .90 size	Relatively light, properly sized for configuration
Nose Landing Gear	Fults RF800	Dual strut configuration for stability under landing and takeoff loads
Propellers	APC 12X6	Selected through wind-tunnel testing of motor/prop combinations
Brakes	Kavan Magnetic Brakes	Allow braking without use of extra servo

5.5 Systems Architecture

5.5.1 Wing

Fast Pitch is a low – wing aircraft with removable wings. The low wing configuration was chosen in order to allow the cargo to be loaded through an entirely open top, and the wings were made removable for several reasons. The first of these is simply the ease of transport, but also important were the ability to manufacture the spars in one piece for each wing half, and the ability to load batteries into the wing through the wing root, eliminating protuberance drag from another fuselage bay. The wings are of hybrid construction, utilizing a carbon-epoxy spar with a Rohacel core, foam ribs, and balsa skins and leading/trailing edges. The spar offers most of the strength of an all-composite wing, without the

complexity of manufacture. Foam ribs are easier to form than balsa, and are substantially lighter than a solid-core wing. Finally, for skins and leading/trailing edges, balsa offers an outstanding strength to weight ratio, as well as easy manufacturability. Ailerons and flaps are both set at 25% of chord, with ±30 degrees of travel. The wings are joined to the fuselage by a 1 ½ inch aluminum joiner tube. The socket for this tube is bonded o the spar with chopped glass/epoxy, and then wrapped with aramid/epoxy. The wings also contain the engine mounts and battery bays.

5.5.2 Fuselage

The fuselage of the airplane is a very low-technology structure. Constructed entirely of wood, it has a central I-beam for the carriage of bending about pitch axis and shear loads in yaw axis. Bonded to this keel are plywood formers with stringers attached around the perimeter. These stringers provide bending resistance about yaw axis, and support the balls. Finally, the stringers and formers are sheeted with balsa. This sheeting is responsible for the torsional loads about roll axis. The doors are constructed in a similar fashion, obviously omitting the keel, and are secured to the fuselage with offset hinges. At the midpoint of the cargo area, there is a 2 inch gap for the wing joiner socket and radio system components. The tail attachment is accomplished by bending extensions of the I-beam caps to a point, and sheeting the sides for torsional rigidity.

5.5.3 Tail

The tails are constructed using traditional balsa-sheeted foam-core techniques. The control surfaces are cut at 33% of chord and have ±30 degrees of travel. The horizontal tail is carried through the fuselage tail box by a small aluminum blade-type joiner. To eliminate interference between the elevator and rudder, the trailing edge of the rudder is just ahead of the hinge line for the elevator.

5.5.4 Landing Gear

The aircraft is of a tricycle gear configuration, and utilizes OTS landing gear for simplicity. The main gear is simple spring aluminum type, and the nosewheel is supported by a dual-strut spring-type nosegear. This arrangement offers good ground handling, with predictable responses. Braking is provided on the main wheels by means of electromagnetic resistance.

5.5.5 Performance Enhancements

In order to minimize time spent in mission completion, the team chose to focus most of its efforts on the landing and ground handling phases of the mission. It was felt that these phases offered greatest opportunity for time savings. An increase of cruising speed might offer time savings, but only with a cost or takeoff penalty. These penalties arise from the manner in which speed can be increased. The first, is obviously to increase power available by increasing cell count. However, doing so not only increases battery weight (and therefore cost) but decreases endurance due to the increased current draw. The

second method for increasing speed is to go to a "faster" prop. A smaller diameter, higher pitch prop will result in a higher V_{max} of the RPV, but penalizes the low-speed side of the thrust curve, decreasing acceleration available for takeoff. These gains are further offset by the amount of maneuvering required by the mission profile, which holds speeds down to a level substantially below maximum cruise. In simulations, an increase of cruising speed of 20 mph was found to have a practical gain of only 15 seconds over the entire mission profile. This does increase flight score somewhat, but is actually unattainable due to the battery weight constraint. Operating within the battery constraint, speed can be increased by selecting a smaller-diameter, higher-pitch propeller, but this actually adds to the time to complete the course, due to decreased acceleration on takeoff and out of maneuvers. On the other hand, if the airplane can be effectively decelerated, allowing a higher-speed approach and/or landing, several seconds can be removed from each landing. In addition, if the loading/unloading time can be reduced, mission time can be substantially reduced. These realizations led the team to make three enhancements to the basic airplane developed. These were the addition of ground brakes, the implementation of an in-flight braking system, and the addition of automated loading/unloading doors.

5.5.5.1 Ground braking system

In order to cut down on ground handling time required, it was realized that accurate placement of the aircraft at the end of the landing roll would be extremely advantageous. Elimination of the need for a long ground-crew run and the need to reposition the plane for the next takeoff offers substantial time savings. Also, the brakes allow the airplane to land at a higher speed, essentially being flown into the ground, rather than forcing a long deceleration and flare. This reduces the amount of time spent in approach and landing, increasing the benefit. Addition of the brakes was found to have very little effect on Rated Aircraft Cost, since they are controlled via a microswitch activated by an existing servo, and therefore contribute only to the Manufactured Empty Weight. Calculations indicated an approximate time savings of 3.5 seconds per landing due to increased approach speed, 5 seconds per load/unload due to decreased ground-crew travel distance, and 3 seconds per takeoff due to the elimination of the need to drastically reposition the aircraft.

5.5.5.2 In-flight braking system

As an extension of the lessons learned from the ground braking system, the team began investigating the possibility of an in-flight braking system. Spoilers were initially considered, but their tendency to destroy lift detracted from their usefulness. The objective of the system in question was to decelerate rapidly in level flight. Upon further consideration, the team settled upon the use of "crow" mixing (also know as "butterfly" mixing). When deployed, crow raises both ailerons, and lowers both flaps. This has the effect of dramatically increasing drag, while not destroying lift completely. In fact, the effect is that of extreme washout, and actually stabilizes the aircraft against tip-stalling, an obvious concern on landing. This

feature eliminates the need for the pilot to make a traditional approach, but rather allows him to run the final straightaway at full speed, throttling back and engaging crow at the threshold, rapidly decelerating and settling gently to the runway. This actually has a dramatic effect on the time required to land, as the aircraft necessarily has relatively low drag in order to complete the mission. Using traditional methods, even flaps, to shed the energy carried by the airplane at 80 mph is quite time consuming. Initial simulations indicate a possible time savings of 10-15 seconds per landing. This is offset somewhat by the requirement to add two servos to the airplane, but was considered very worthwhile.

5.5.5.3 Automated Doors

In order to further cut down on ground-crew time, it was decided to automate the cargo doors of the airplane. While the task of opening and closing a door may not seem time consuming, upon further examination, the unlatching and opening of a 51-inch long door is necessarily complex due to the need to adequately secure the door. This penalty also applies to the closing and resecuring of the doors. This time could be eliminated if the doors were capable of opening and closing on a command from the radio. Implementing this system does require the addition of two more servos, but offers a time savings of at least 10 seconds per loading/unloading

5.6 Other

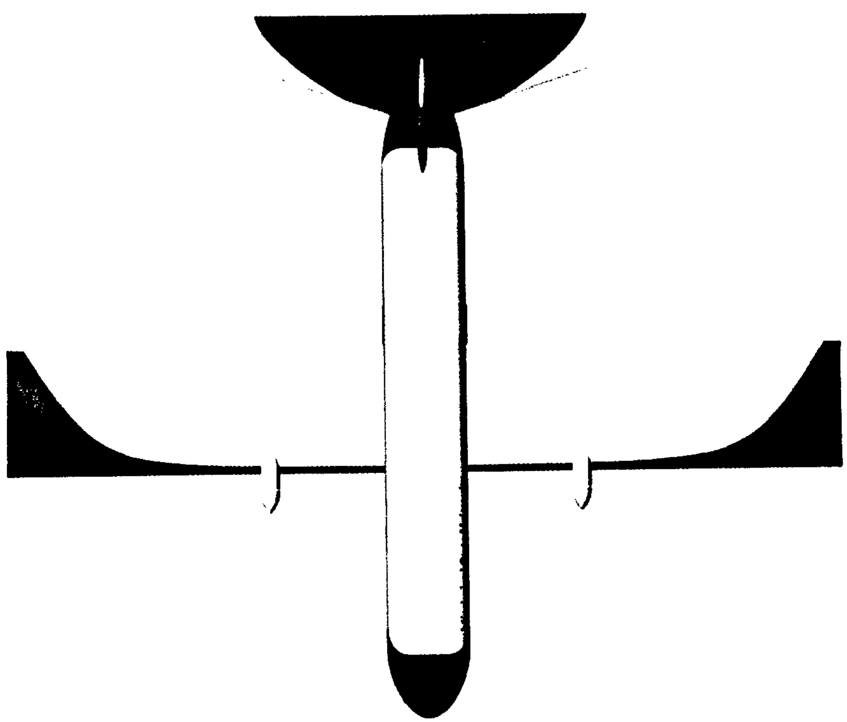
There are several other items of note in system architecture. The first of these is the use of two independent drivetrains that are separately throttleable for elimination of potential asymmetric thrust from the non-programmable speed controls used. The battery packs are located in the wing roots to eliminate the need to place them in the fuselage or add a drag-creating projection for them. This does require the removal of the wings to change packs, but eliminates a great deal of system complexity, as all high-power wiring is concentrated in the engine nacelles. Also, the nosewheel is steerable through a its own servo which is mixed to the rudder. Finally, all control surfaces are directly driven by servos located nearby. In the case of wing surfaces, these linkages are accomplished entirely internally, while the tail does require external control horns.

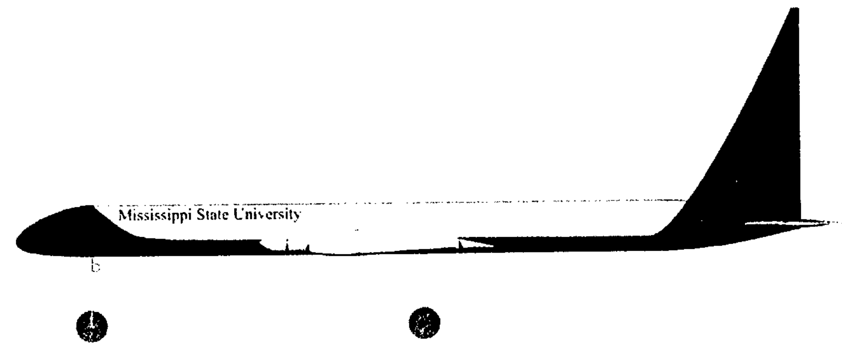
5.5 Rated Aircraft Cost Worksheet

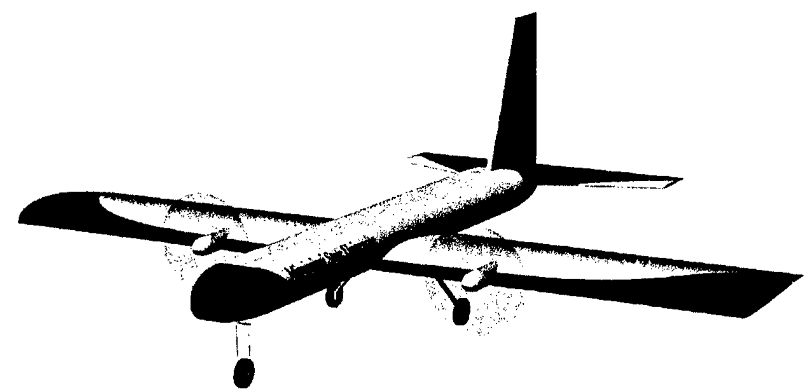
Table 5.5 Rated Aircraft Cost

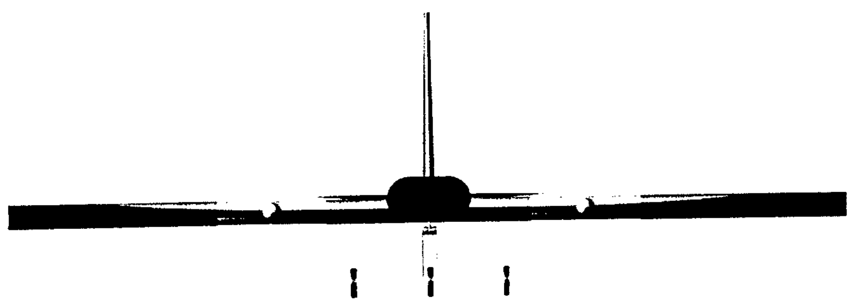
	Measurement	Scale		Cost:
Manufactured Empty Weight (lb)	20.7	100		2070
Number of Engines	2			
Battery Weight (lb)	4.7			
Rated Engine Power	5.875	1500		8812.5
MFHR: Manufacturing Hours				
Item			Hourage	
Wing Span (ft)	7.521	8	60.168	1203.36
Wing Chord (ft)	1.5	8	12	240
Horizontal Tail Span (ft)	3	8	24	480
Horizontal Tail Chord (ft)	1	8	8	160
Wing Control Surfaces	4	3	12	240
Horizontal Tail Control Surfaces	1	3	3	60
Fuselage Length (ft)	6.33	10	63.3	1266
Vertical Tail	1	10	10	200
Servos and Motor Controllers	11	5	55	1100
Engines	2	5	10	200
Propellers	2	5	10	200
TOTAL MFHR			267.468	5349.36
TOTAL RAC				**16.232**

5.6 Drawing Package

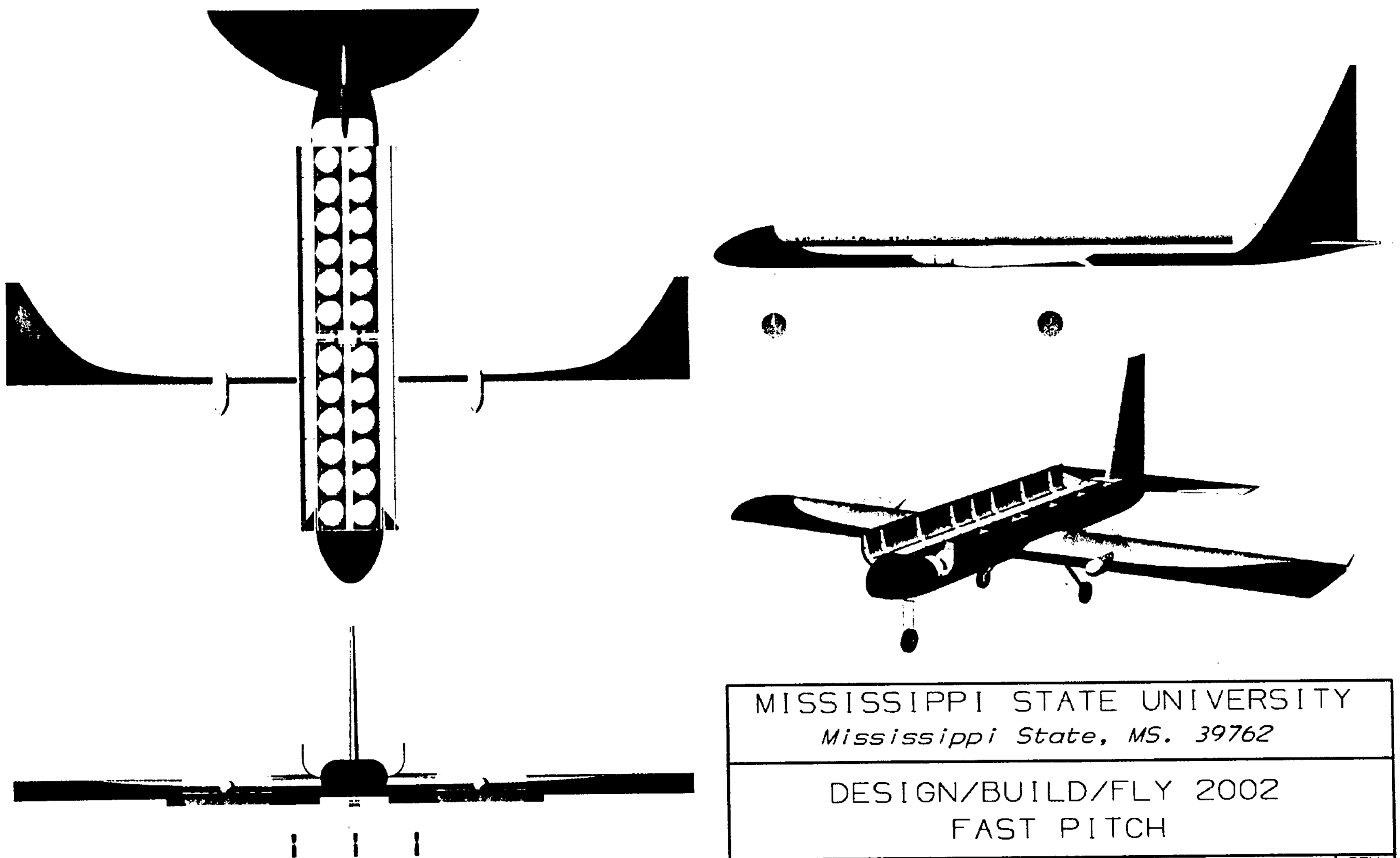








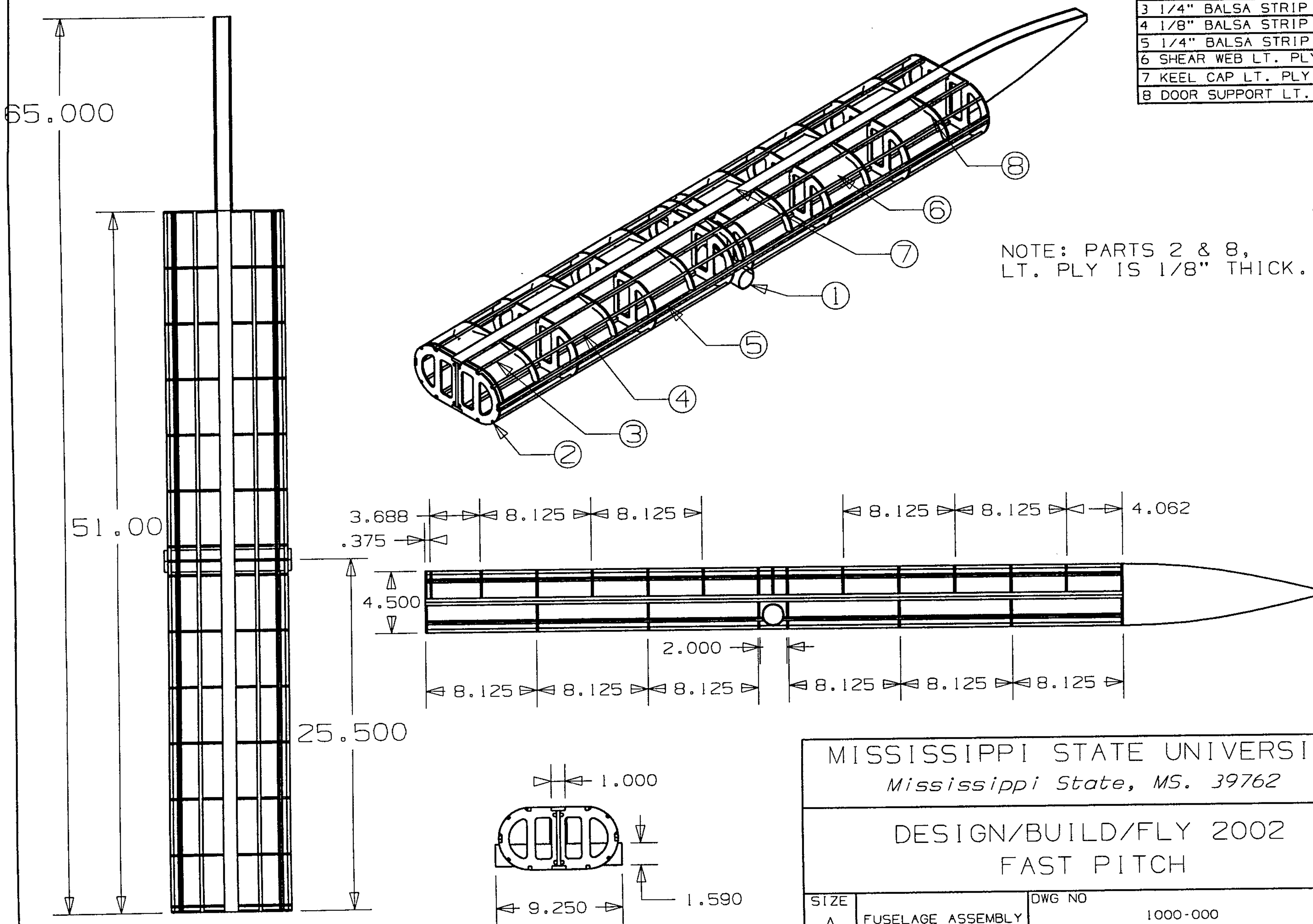
MISSISSIPPI STATE UNIVERSITY *Mississippi State, MS. 39762*			
DESIGN/BUILD/FLY 2002 FAST PITCH			
SIZE A	3-VIEW & PERSPECTIVE	DWG NO 0000-000	REV OR
SCALE NTS	DRAWN J. SEBENS	SHEET 1 OF 1	



MISSISSIPPI STATE UNIVERSITY *Mississippi State, MS. 39762*			
DESIGN/BUILD/FLY 2002 FAST PITCH			
SIZE A	3-VIEW & PERSPECTIVE	DWG NO 0000-001	REV OR
SCALE NTS	DRAWN J. SEBENS	SHEET 1 OF 1	

PART LIST
1 WING TUBE JOINER
2 FUSELAGE FORMER LT. PLY
3 1/4" BALSA STRIP
4 1/8" BALSA STRIP
5 1/4" BALSA STRIP
6 SHEAR WEB LT. PLY
7 KEEL CAP LT. PLY
8 DOOR SUPPORT LT. PLY

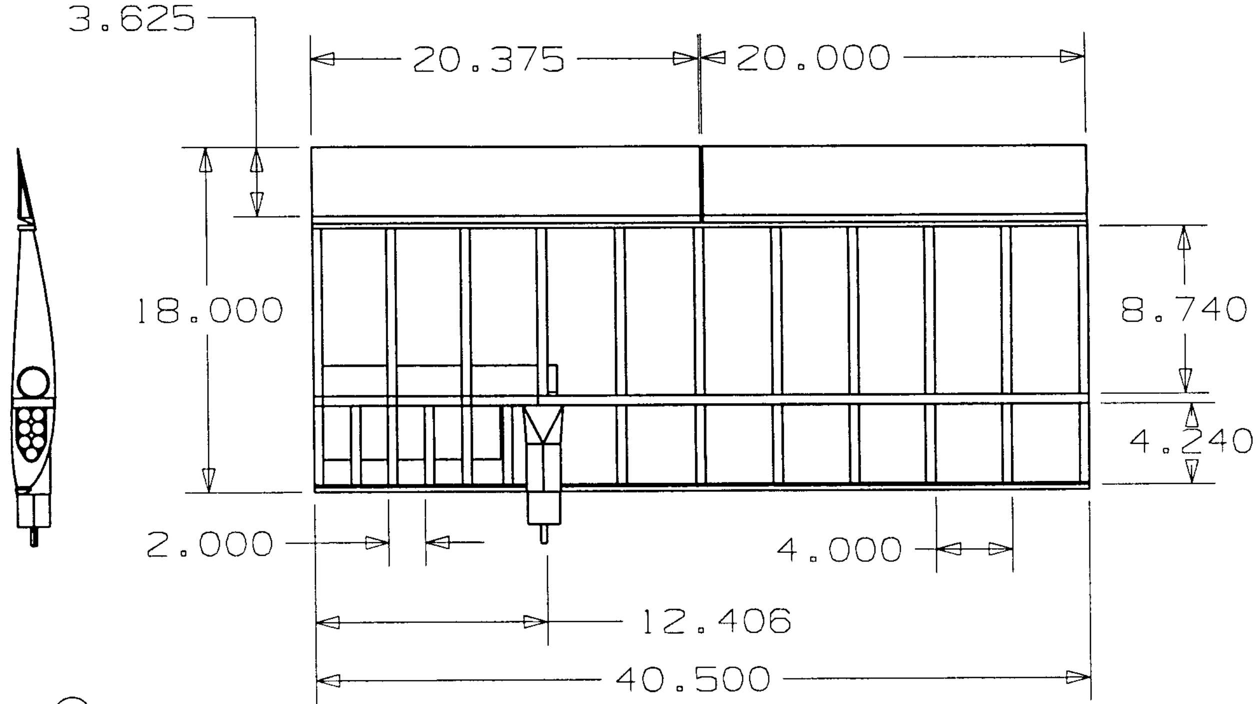
NOTE: PARTS 2 & 8,
LT. PLY IS 1/8" THICK.

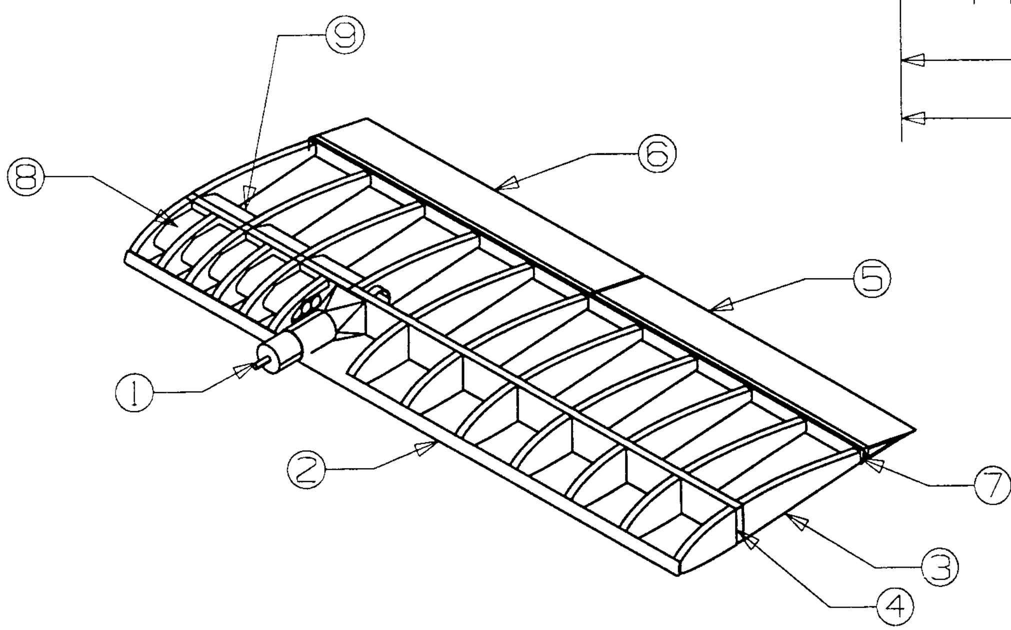


MISSISSIPPI STATE UNIVERSITY *Mississippi State, MS. 39762*			
DESIGN/BUILD/FLY 2002 FAST PITCH			
SIZE A	FUSELAGE ASSEMBLY	DWG NO 1000-000	REV OR
SCALE 1/10	DRAWN J. SEBENS	SHEET 1 OF 1	

PART LIST
1 ENGINE/ENGINE MOUNT
2 BALSA LEADING EDGE
3 FOAMULAR RIB
4 COMPOSITE SPAR
5 AILERON
6 FLAP
7 REAR SPAR
8 BATTERY SLEEVE
9 WING TUBE JOINER

NOTE: WING IS SHEETED WITH
1/8" BALSA





MISSISSIPPI STATE UNIVERSITY *Mississippi State, MS. 39762*						
DESIGN/BUILD/FLY 2002 FAST PITCH						
SIZE A	LEFT WING ASSEMBLY	DWG NO 2000-000				REV OR
SCALE	1/10	DRAWN	J. SEBENS	SHEET	1 OF 1	

6 Manufacturing Plan

6.1 Introduction

In order to build the airplane designed previously, it is necessary to select among the myriad possible construction methods and materials. For each structure on the airplane, various methods and materials were evaluated for two figures of merit: projected weight and projected construction time. Eventually, a hybrid aircraft, part wood, part metal, part foam, and part advanced composites emerged. The team is confident that this mix of high- and low- technology components results in an airplane that effectively balances manufacturability and performance.

6.2 Construction Methods Investigated

Four major methods for construction were considered for the aircraft. These methods are designated by the somewhat terse names of off-the-shelf (OTS), built-up, molded, and moldless construction. The terms are not necessarily self explanatory, so the following table is included for clarification. Methods are presented from most to least manufacturable.

Table 6.1 Construction Methods Investigated

Method	Description
Off-the-shelf	Components that can be purchased from hobby stores (only applicable to certain structures).
Moldless	Refers to solid-core structures in which skins (if any) are applied directly to the underlying core.
Built-up	Refers to structures that are assembled from individual components that may be either hand- or machine-cut. Skins, if any, simply conform to the underlying structure.
Molded	Internal structures may be built-up or solid-core, but skins are pre-molded to exact shapes before assembly to structure.

6.3 Construction Materials Investigated

Four basic materials were investigated: wood, foam, metal, and advanced composites. Wood includes balsa, lite-ply, aircraft plywood, and hardwoods such as spruce and basswood. Foam includes low- and high-density EPS, as well as 1 lb white foam. Metal indicates aluminum structures, and advanced composites refer to glass-epoxy, Kevlar-epoxy, and carbon-epoxy. Obviously, the use of each of these materials requires certain technical skills. Ranking these materials from the perspective of the

inexperienced hobby builder, advanced composite construction requires the most skill, and wood construction requires the least skill. The two other construction techniques, metal and foam, are categorized as being somewhere between wood and advanced composite construction. Of these two, foam cutting and shaping is the harder skill to learn for the beginning builder. There are many tasks and skills associated with foam cutting such as making templates, knowing the correct temperature for the hot wire, and the correct speed to move the wire over the templates to cut the foam. However, with metal construction, the beginning builder can be taught relatively easily the skills necessary for the forming of parts. The following table summarizes the construction methods investigated as well as the rank of the skill required ranging from 4 (most skill) to 1 (least skill).

Table 6.2 Construction Methods Investigated

Materials	Skill	Description
Advanced Composites	4	Refers to glass-epoxy, Kevlar-epoxy, and carbon-epoxy. Skills associated may include: composite layouts, mold construction, and vacuum bagging techniques.
Foam	3	Includes low- and high-density EPS, as well as 1-lb white foam. Skills associated may include: template construction and hot wire cutting.
Metal	2	Refers to aluminum construction. Skills associated may include: cutting, bending, and drilling.
Wood	1	Includes balsa, lite-ply, aircraft plywood, and hardwoods such as spruce and basswood. Skills associated may include: gluing, cutting, and sanding.

6.4 Selection of Major Component Construction Methods and Materials

Using numerical models developed from sample structures, each method and material combination that was considered feasible for a given part was evaluated for four parameters. The first parameter, projected part weight, refers to the weight of completed parts, but does not include assembly weights such as adhesives. Projected assembly weight, the weight of adhesives, fasteners, etc., is added to the projected part weight to construct the total part weight. The other two parameters, time to construct the bare components and the time to assemble the bare components into a completed structure, are added together to form the projected construction time.

A cost function was developed to rank the construction method and material used for each major structural component. This cost function, which is a function of the two figures of merit for this phase, estimated weight and projected construction time, was used as a decision factor:

$$\mathrm{Cost} = W^2 + \frac{1}{4}t$$

The squaring of weight was chosen in order to make weight more important on large, heavy parts than on smaller ones. For example, if a small light part could be manufactured in half the time with an additional weight cost of ten percent – perhaps half an ounce – the weight increase is acceptable. However, in a similar situation with a large part in which ten percent could mean half a pound, the lighter, less manufacturable part would be the appropriate choice. For each assembly, the method and material resulting in the lowest value of the resulting cost function were selected. The results from this analysis are presented in Table 6.3 with the chosen major component construction method and material highlighted.

6.5 Manufacturing Timeline

The construction chosen represents a mix of high- and low-skilled methods in which both the experienced hobby builder and the inexperienced hobby builder can participate. The ratio of members that are experienced to inexperienced hobby builders is 1-to-2. This ratio offers a positive experience for the team, as the more experienced members can train up the less experienced, building both their own expertise and team continuity for future entries.

Scheduling a task such as this construction project is very difficult due to the constant pressure of homework and exams and prior commitments. The schedule was constructed based on the major structural components listed in Table 6.3. Table 6.4 shows the actual and projected timelines of the manufacturing process. At the time of the completion of this report some tasks are not completed, and are indicated with a completion time of TBA.

Table 6.3 Selection of Major Component Construction Methods and Materials

Description			Projected weight (lbs)			Projected Construction Time (hrs)			COST
Structure	Method	Material	part(s)	assembly	TOTAL	part(s)	assembly	TOTAL	
Fuselage (internal)	built-up	wood	2.1	0.2	2.3	10	14	24	11.29
	moldless	foam	2.6	0.5	3.1	5.5	13	18.5	14.235
Fuselage (skin)	molded	adv. comp.	1.2	0	1.2	55	5	60	16.44
	built-up	wood	1.1	0.2	1.3	2	10	12	4.69
Wing (spar)	built-up	wood	0.8	0.2	1	6	3	9	3.25
	moldless	adv. comp.	0.4	0.1	0.5	8	3	11	3
	OTS	metal	4	0.5	4.5	0	4	4	21.25
Wing (other internals)	built-up	wood	0.6	0.2	0.8	12.5	12	24.5	6.765
		foam	0.6	0.2	0.8	5	12	17	4.89
		adv. comp.	0.5	0.1	0.6	15	12	27	7.11
	moldless	foam	2.2	0	2.2	5	8	13	8.09
Wing (skins)	built-up	wood	2	0.2	2.2	2	8	10	7.34
	molded	adv. comp.	1.5	0.2	1.7	55	5	60	17.89
(only abailable with moldless internals)	moldless	wood	2.2	0.3	2.5	2	6	8	8.25
		adv. comp.	1.8	0.3	2.1	5	12	17	8.66
Empennage (internals)	built-up	wood	0.2	0.1	0.3	9.5	6	15.5	3.965
		foam	0.2	0.1	0.3	4	6	10	2.59
		adv. comp.	0.15	0.05	0.2	12	6	18	4.54
	moldless	foam	0.35	0	0.35	3	0	3	0.8725
Empennage (skins)	built-up	wood	0.25	0.05	0.3	2	6	8	2.09
	molded	adv. comp.	0.15	0.05	0.2	30	5	35	8.79
(only abailable with moldless internals)	moldless	wood	0.25	0.1	0.35	2	4	6	1.6225
		adv. comp.	0.2	0.1	0.3	4	6	10	2.59
Landing Gear	OTS	metal	0.7	0.1	0.8	0	4	4	1.64
	moldless	metal	0.6	0.1	0.7	10	4	14	3.99
		adv. comp.	0.4	0.1	0.5	20	4	24	6.25
Wing joiner	OTS	metal	1	0.1	1.1	0	4	4	2.21
		adv. comp.	0.5	0.1	0.6	0	4	4	1.36
	moldless	metal	0.8	0.1	0.9	7	4	11	3.56
		adv. comp.	0.4	0.1	0.5	15	4	19	5

Table 6.4 Manufacturing Timeline

Task Name	Start	Finish	February	March
RPV Construction Period	2-Feb	22-Mar		
Fuselage Internal Assembly				
Planned	9-Feb	28-Feb		
Actual	9-Feb	2-Mar		
Fuselage Skin Assembly				
Planned	2-Mar	13-Mar		
Actual	3-Mar	TBA		
Wing Spar Assembly				
Planned	16-Feb	17-Feb		
Actual	16-Feb	20-Feb		
Wing Internals				
Planned	2-Feb	24-Feb		
Actual	2-Feb	TBA		
Empennage Internals				
Planned	2-Mar	2-Mar		
Actual	2-Mar	2-Mar		
Empennage Skins				
Planned	7-Mar	11-Mar		
Actual	7-Mar	TBA		
Landing Gear Integration				
Planned	12-Mar	12-Mar		
Actual	TBA	TBA		
Final Assembly				
Planned	18-Mar	22-Mar		
Actual	TBA	TBA		

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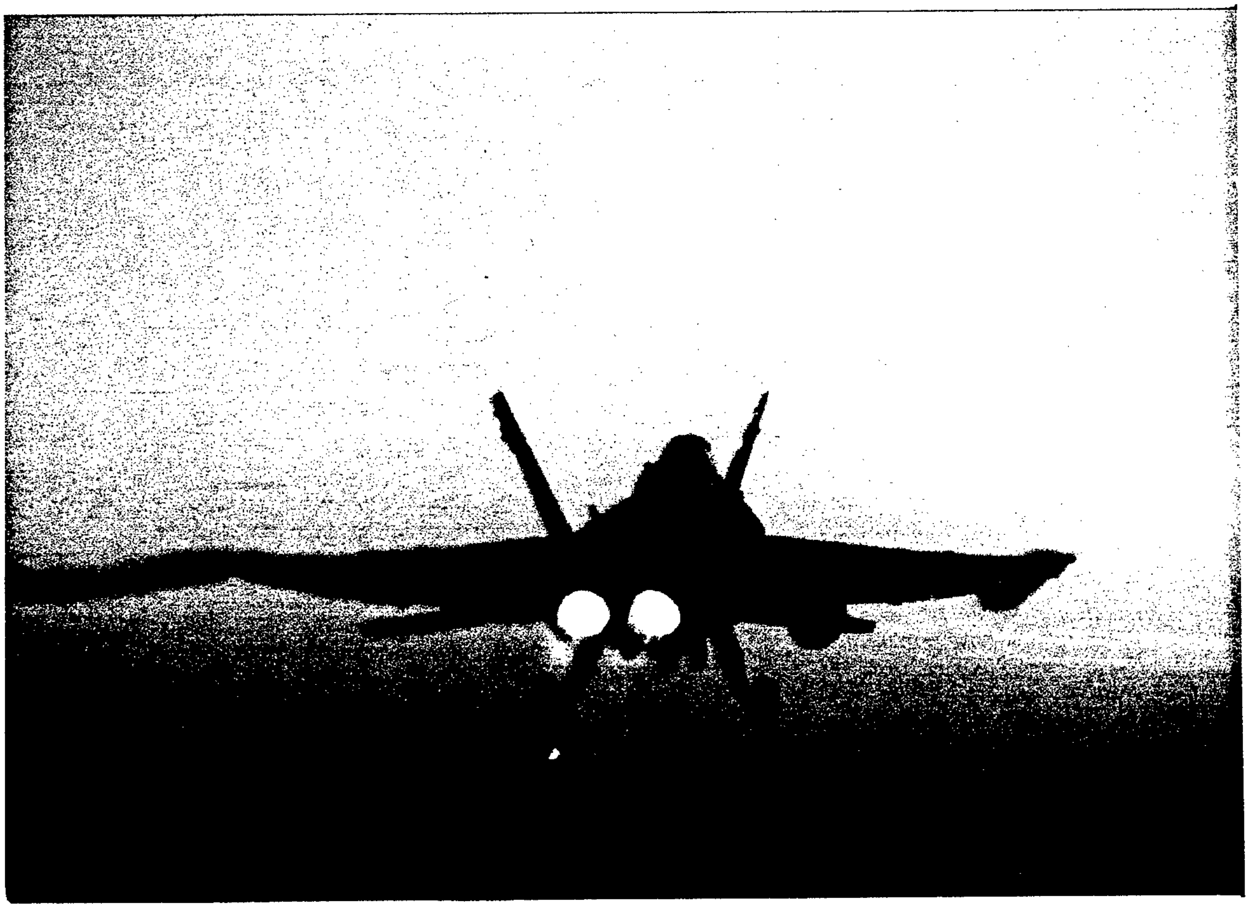
3. *AIAA Student Design/Build/Fly Competition*, http://amber.aae.uiuc.edu/~aiaadbf/, University of Illinois at Urbana-Champaign, Urbana, IL, 2001.

THE CESSNA/ONR 2002 DESIGN – BUILD – FLY COMPETITION

Design Report
Proposal Phase



THE UNITED STATES NAVAL ACADEMY

TEAM TANGO

USNA BLUE

TABLE OF CONTENTS

LIST OF SYMBOLS

a .... Lift curve slope of the aircraft
a_F .... Lift curve slope of the vertical fin
a_o .... Lift curve slope of the infinite airfoil section for the wing
a_{wb} .... Lift curve slope of the wing body (finite section)
a_t .... Lift curve slope of the horizontal tail
a_e .... Lift curve slope of the horizontal tail
α .... Angle of attack
$\alpha_{L=0}$ .... Angle of attack zero lift
AR .... Aspect ratio of the aircraft
AR_{vt} .... Aspect ratio of the vertical tail
b .... Wingspan of the aircraft
b_v .... Wingspan of the vertical tail
C_{m_α} .... Change in pitching moment with angle of attack
C_{m_o} .... Pitching moment at geometric angle of attack
$C_{m_{acwb}}$ .... Pitching moment of infinite airfoil about quarter chord
C_{L_α} .... Lift curve slope of aircraft (same as a)
$C_{L_{\delta e}}$ .... Change in lift with elevator deflection
$C_{m_{\delta e}}$ .... Change in pitching moment with elevator deflection
C_{n_β} .... The weathercock stability derivative
$C_{n_{\beta w}}$ .... The above value due to the wing
C_{l_β} .... The dihedral effect
$(C_{l_\beta})_w$ .... The dihedral effect due to the wing
$(C_{l_\beta})_\Gamma$ .... The dihedral effect due to the amount of dihedral built into the wing
$C_{l_{\delta a}}$ .... Change in lift due to aileron deflection
C_r .... Root chord length
C_{l_p} .... The damping in roll derivative
$C_{l_{\alpha w}}$ .... Lift curve slope of the wing
$\bar{c}$ .... Mean aerodynamic chord of the wing
δ_e .... Change in elevator deflection
e .... Oswald efficiency factor
m.a.c .... Mean Aerodynamic Chord
h .... Ratio of distance of placement of cg to the mean aerodynamic chord
h_n .... Ratio of distance of placement of fixed stick neutral point to m.a.c.
$h_{n_{wb}}$
Ratio of distance of placement of the aerodynamic center of the wing to the m.a.c.
i_t .... Incidence angle

K...........A constant found in DATCOM to aid in finding the damping roll derivative
l_{t_v} .... Distance from c.g. to aerodynamic center vertical tail
$l'_{t_{n(H)}}$ ....
Distance from the fixed stick neutral point to thė aerodynamic center of the horizontal tail
l_F .... Distance from the c.g. to the aerodynamic center of the vertical tail
m ....Mass of the aircraft
m_q ....Damping in pitch value
p_{ss} ....Steady state rolling rate due to aileron deflection
S... Surface area of the wing
S_t .... Surface area of the horizontal tail
S_{vt} .... Surface area of the vertical tail
S_F .... Surface area of the vertical fin (same as above)
μ....................A constant used for calculating change in g's for elevator deflection
u_o .... Forward flight velocity
V_H .... Horizontal tail ratio
V_{VT} ....Vertical tail ratio
V'_H ....Horizontal tail ratio to find stick fixed neutral point
w...Average width of the fuselage
y_1 ...Distance from the root chord of the wing to the inboard section of the aileron
y_2 Distance from the root chord of the wing to the outboard section of the aileron
β .... Yaw angle
$\frac{\partial \varepsilon}{\partial \alpha}$ ....Change in downwash in change in angle of attack
ε_o ....Downwash angle
ρ .... Density of air

1.0 Executive Summary

1.1 Summary of the Development of the Design

The 2002 U.S. Naval Academy's Design Build Fly team began its challenge to compete in the Cessna competition in the fall of 2001. The team felt that the best way to start a project like this would be to learn from the experience of last year's team. At the same time, the team set out to gain initial, hands on experience with aircraft construction techniques. These were put together to come up with individual conceptual designs based on each team member's personal knowledge. After submitting these individual conceptual designs for review and getting feedback on what was done, the group was able to move on to a more focused preliminary design. The group looked at designs having different engines, fuselages, tails, and wing configurations. To determine which parameters would yield the best flight score, several computer programs, some written by the students themselves, were used. These parameters were then optimized in the final detailed design on the aircraft.

1.2 Major Areas in the Development Process for the Final Configuration

By having each member of the team develop their own conceptual design, it allowed the team to explore a wide range of designs and compare the benefits and drawbacks of each aircraft before preliminary and detailed design began.

For the preliminary design, the team broke into small groups, where each was assigned a specific part of the aircraft, for example the wing or tail. Each group then compiled all the best aspects of the conceptual designs that the team came up with, and a better and more refined design of each part of the plane was determined. Full-scale drawings were then completed of each design, and these drawings were shown to craftsman to determine if they could be constructed, and how difficult it would be to build them. These plans were then modified, and the team moved on to the detailed design.

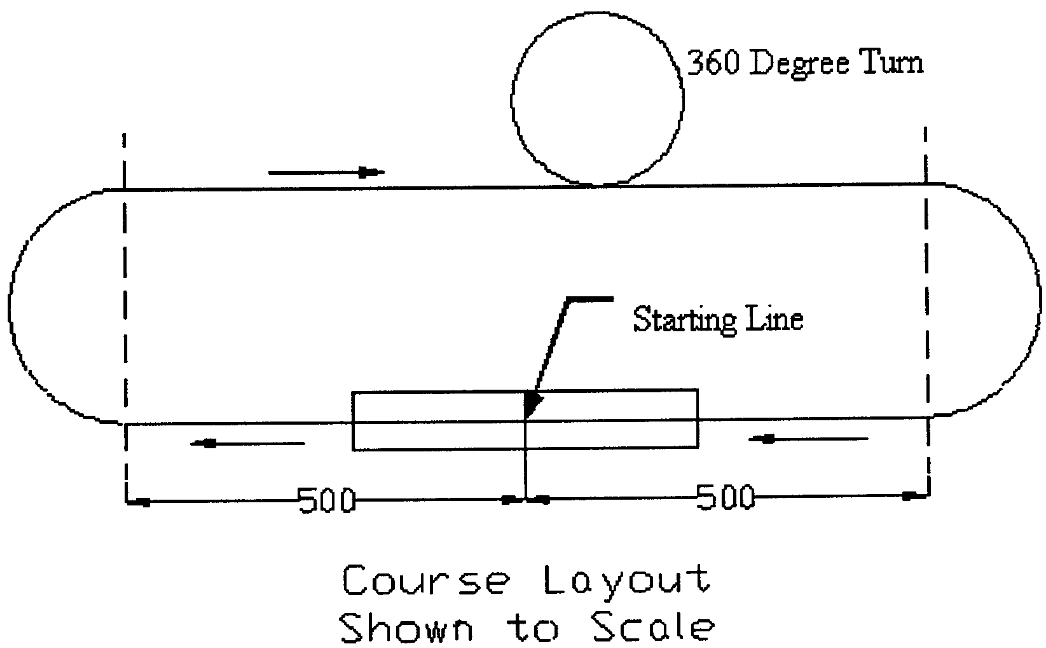
The detail design of the aircraft included structures, propulsion, stability and control, and aerodynamics. Each of the small groups' final designs were taken to the shop where the team worked with the technicians to build the aircraft. The team wanted to have the aircraft completed before the middle of March, so that there was ample time to perform flight-testing and allow the team's pilot to become familiar with the aircraft.

1.3 Range of Alternatives Investigated

The competition would be composed of three sorties made in the span of ten minutes. Each sortie would make two laps around the course shown below.

Figure 1

Course Layout



The first sortie would be made without payload. For the second flight, the aircraft would be loaded with ten to twenty-four softballs. These would subsequently be removed for the third sortie. The third sortie also excluded the 360-degree turn on the downwind leg.

The score would consist of the total number of laps flown plus the number of balls carried divided by the total time of the flight. This number would also be divided by rated aircraft cost. So immediately, the objectives of the design were to maximize the number of balls carried and number of laps flown, while minimizing the time to complete the course and the rated aircraft cost. Each member of the team explored a great range of alternatives in their conceptual designs. These covered several areas. The first was payload and weight, since the team wanted to know how many balls they could carry and in what configuration. For example, the team needed to decide if it would be better to build hard points and canisters to attach to the wings, or to have the balls carried internally as cargo. The second major area was propulsion. The team wanted to maximize speed, be able to take off within the limit distance, and still complete the course, so a variety of single and twin engine and propeller configurations were explored. The third major area was aerodynamics, and, specifically, what would be the best wing and fuselage configuration. The team looked at high, mid, and low wing setups, along with varying sizes, tapers, aspect ratios, and setups. The aspect ratios selected ranged from 7 to 12, which was based on both the structures available for building and turn handling characteristics. High lift designs were the most heavily considered because of the large amount of weight that the wing was expected to be carrying

at slow speeds. Wing position was also subject to a lot of debate for both aerodynamic and assembly reasons.

1.4 Overview of the Design Tools

There were three central tools used to do the major calculations for the conceptual, preliminary, and detailed designs. The first a MATLAB algorithm written by one of the team members that took into account all of the major design parameters, to include propulsion, payload and weight, aerodynamics, and course performance. Based on these inputs the code projected a theoretical score based on the design competition formula, and was accordingly named DBF. The second design tool used was a program called ECALC, which modeled engine, battery, and propeller performance and gave predicted results to include thrust, propeller speed, and time to battery drain. The final design tool was MFOIL, which was a database of major airfoil sections and their lift, drag, and moment characteristics at varying angles of attack.

2.0 Management Summary

Team Tango, registered as team Blue, had to pull together all the available resources that could be found at the Naval Academy to help them complete the task of designing, building, and flying a remotely controlled aircraft. The team was made up of members from 3 classes, and all members' experience levels ranged from none at all, to Aeronautical Engineering majors, to skilled RC pilots, and finally experienced machinists and builders.

2.1 Architecture of the Design Team – List of Personnel & Assignment Areas

Once Team Tango was established, positions within the team were selected. Midshipman Peter Scheu was chosen to be team manager for his organizational skills and his overall background with RC airplanes. Midshipman Robert Frantz was chosen to do the aeronautics side of the project, based on his ability to conceptually model the wing and the fuselage. Because of his long-term prior building experience with RC models, Midshipman Nathaniel Swift was chosen to be in charge of the structures. Midshipman Dong Park took over the propulsion plant design and testing. Finally, Midshipman Brandt Sword was put in charge of the stability and control of the aircraft.

The teams MATLAB expert, Midshipman Michael Stanfield, worked on a software project to write code that would give a projected score for an aircraft based on different input variables. This code allowed Team Tango to conceptually test aircraft in the program before they began the actual construction. The program's real power lay in its ability to allow the user to lay down the fundamental design parameters and vary individual design parameters to see how they would affect the aircraft's performance. This allowed a great amount of conceptual optimization for the aircraft in the pre-construction period. Midshipman Blazel wrote a spreadsheet in Microsoft Excel to determine the wing root

bending moment produced from varying load conditions, and wingspan. Both teams used this spreadsheet to determine the necessary structural requirements for the wings.

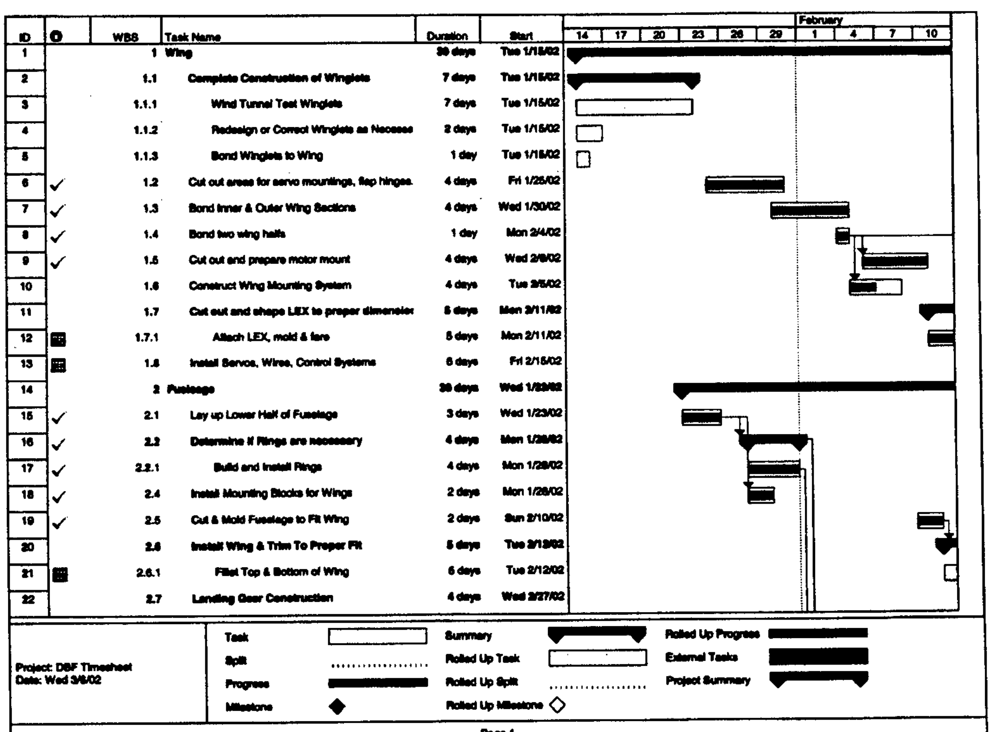
There were also several additions to the team that were not Aerospace Engineering majors at the Naval Academy. These people joined the team because of their interest in RC modeling and their wealth of experience in the area. Midshipman Eric Sorensen, USNA's 2002 DBF team pilot, was the main advisor to both teams during their design and construction phase. His input from a practical applications and experience standpoint was a great help to both teams, as neither had a great deal of experience building RC aircraft, or flying them for that matter. Additionally, 2/C Nicholas Keller, 3/C Collin Gaines, and 3/C Michael Bernard joined the team as underclass to add their insight and to learn about the design process for when they become team leaders in the future. Lastly, professional advise from the USNA faculty and Boeing engineers was considered when the team's views narrowed and needed opening.

2.2 Schedule and Configuration Control & Work Breakdown Structure & Milestones

The work breakdown structure for the team focused primarily on the construction of the aircraft and the completion of the report. Microsoft Project was used to organize and keep all of the tasks and their required stop and finish times. This program was convenient because it allowed each member to see the team's progress with each item, and what items needed to be completed before others could be started.

Figure 2

Work Breakdown Diagram



While all of the items that needed to be completed can not be seen in Figure 2, they included the wing, fuselage, tail section, propulsion, control systems, landing gear, systems integration test, flight testing, and the report.

Because there was a very large amount of work required to complete the task of designing and building a remotely operated vehicle, it was important to divide the work into parts and set deadlines for each large amount of work to be done. To help give the team a sense of the important deadlines for the competition, major milestones for the team were set. A major milestones summary is given in Table 1.

Table 1

Major Design Milestones

Milestone	Date
Complete Conceptual Designs	October 9, 2002
Begin Aircraft Construction	November 26, 2002
Complete Preliminary Aircraft Design	December 9, 2002
Complete Detailed Aircraft Design	January 28, 2002
Present Detailed Aircraft Designs to Boeing Engineers	January 29, 2002
Submit Design Report	March 12, 2002
Complete Aircraft Construction	March 22, 2002
Begin Flight Test	March 23, 2002
End Flight Test	April 13, 2002
AIAA Competition	April 27, 2002

2.3 Summary of Major Design Tools

Because the development of an aircraft, both in the conceptual and final design stages, required a great deal of calculations and mathematical iterations, the team used several computer programs to aid them in their work. These programs are summarized below.

2.3.1 ECALC

ECALC was used extensively throughout the design process in order to modify and refine the aircraft design. ECALC allowed the user to input all the pertinent aircraft väriables such as weight, wing area, drag coefficient, motor, propeller, and batteries, and then gave an output of the total amp drain at a certain motor speed setting. This allowed the user to not only efficiently match a propeller to a motor, but also allowed the user to observe the change in flight velocity caused by changes in aircraft design. For example, as the wing area went down, the maximum attainable velocity went up. The results of ECALC were further tested using an Eiffel wind tunnel in order to verify the precision of the ECALC predictions, which were shown to be slightly lower than actual. The efficiency parameters in ECALC were then reset based on this testing to more accurately model real results. ECALC was an essential tool in the early

stages of design, allowing quick rejection of ideas that were shown not to realistically work within the five-pound battery limit.

2.3.2 MFOIL

MFOIL was an invaluable tool in designing the wing. MFOIL is a computer program with a database of over 1300 airfoils that also has the ability to allow the user to specify a certain airfoil shape. The MFOIL program not only shows the user the shape of the airfoil, but also allows the user to add flaps and then outputs an airfoil lift coefficient based on an input angle of attack. The program then gave the stall characteristics and separation points based on the user's desired Reynolds number. This allowed model airfoils to be grouped according to performance or stall characteristics. An extensive search of this database and specific group-entered designs eventually allowed the team to select the 5412 airfoil through trial and error. After selecting the airfoil, MFOIL allowed the user to print out different airfoil section templates along the entire span of the wing for later use in the manufacturing process.

2.3.3 DBF

Early on in the design process, the team was confronted with the problem of testing possible designs. Because the calculations required for each design that the team wanted to test were very extensive and time consuming, the team needed some way to expedite this processes. Otherwise, it would be very difficult for the team to compare how well different designs could complete the required course. Also, once a final design was decided on, the team needed a way to optimize this design by slightly varying aircraft characteristics. Again, however, the amount of calculations involved was very large. Because it was obvious that computing power was needed to rapidly do these calculations for the team, a MATLAB code was written.

The solution to both of these problems was to create a computer program that would perform major design and competition calculations, with the final outputs being score and battery consumption. One of the requirements of the program was that it would have to be easily modified, and comparisons between proposed designs shown to the user. A second version of the program was created, which added an elementary trend evaluation of each of the input variables. This trend evaluation showed the weighted percentage increase or decrease in score and battery consumption relative to changes in each of the input variables. This provided the team an opportunity to modify selected variables to produce desired outcomes in score or battery usage.

The input variables for the DBF program were divided into three groups: basic aircraft variables, battery variables, and flight variables. The basic variable input consisted of important aspects of the plane's size, turning flight, attainable efficiency assumptions, and attainable zero-lift drag assumptions. The specific input variables were the plane's empty weight, wing surface area, wing aspect ratio, turning Gs, Oswald efficiency factor, and zero-lift drag coefficient. The battery variable input was composed of the number of batteries, volts per battery, and current rating per cell. The flight variable input contained

payload considerations, cruising and turning velocities, maximum lift assumptions, and assumed wind velocities on the flight course. The specific input variables were the plane's softball payload, cruising velocity, turning velocity, maximum coefficient of lift, time to load/unload the payload, and the downwind wind velocity on the course.

Following these inputs, the DBF program displayed the drag data for cruising, turning, and approach flight when the plane was empty and fully loaded. The team used ECALC to find the currents required for flying at their desired speeds. If the thrust available estimated by ECALC was the drag from the DBF program, the design was infeasible. The DBF program used this last input to display the total score for the course, total amps for the course, total time for the course, and a breakdown of time and amps consumed for each required maneuver. If the data was satisfactory, the data was saved in a text file for further analysis and comparison to other designs.

The trend evaluation program required the same inputs and eliminated the user inputs from ECALC. This elimination of ECALC user inputs was achieved by fitting lines to drag vs. amp curves for a given plane configuration at different velocities. The output provided the team with a sensitivity table for the following design aircraft variables: zero-lift drag coefficient, empty weight, payload size, wing surface area, wing aspect ratio, Oswald efficiency factor, cruise velocity, turn velocity, and turning Gs. These variables were selected because they were the easiest way to change the plane whether by physical dimensions, streamlining, or control inputs. This sensitivity data could be saved to a text file for future reference and gave the team possible design modifications to enhance the performance of the plane.

2.3.4 Wing Root Bending Moment Spreadsheet

The wing root bending moment was a key structural aspect of the design that was also rather tedious to calculate by hand. The bending moment was important to know because it influenced the size and aspect ratio of the wing, and the amount of materials necessary to ensure that the wing would not fail under high loads. To make these calculations much easier, Midshipman Blazel constructed a spreadsheet that used the Schrenk's approximation technique to estimate the lift distribution along the span and the resultant bending moment. The program used the aircraft's full and half span lengths, weight, G loading, and taper ratio as inputs. Using the spreadsheet, all of these inputs could be easily changed to explore different conceptual designs, or to optimize a potential final design.

3.0 Conceptual Design

The conceptual design of the aircraft began after a preliminary introduction to the important parameter considerations in aircraft design. Based on experimental and historical trends, the initial conditions set forth general requirements. The team wanted the aircraft to have a C_{Lmax} of 1.3 to 1.8, a wing loading between 5 to 6, a zero lift drag coefficient around .04, a lift to drag ratio of 8 to 12, and finally

a thrust to weight ratio of no less than 1:3. These numbers all came from historical data and represented what the group felt was the most characteristic of aircraft with missions similar to the one the team would be designing for. As almost all of these requirements are dependant upon lift, the first logical step therefore was to estimate a gross weight for the aircraft.

The weight buildup for the aircraft varied from each person's initial design report by a significant amount and was subject to some deliberation. The items that the team took into account included the cargo of softballs, the batteries, the motors, propellers and gears, the landing gear, and then the fuselage, wing, and empennage sections. Perhaps the most important weight consideration was the amount of softballs to be carried. This turned out to be a compromise, since, based on the score equation, it was obvious that the more softballs that could be carried the higher the score would be. However, if a smaller number was used, it reduced the weight, and allowed for a faster and smaller aircraft. Since the score was also based on the time it would take to complete the course, creating a faster airplane reduced the score. Thus, the best of both design parameters had to be considered to determine the number of balls carried on the missions.

Once the weight of the aircraft was found, it was then used to determine the lift needed based on the propulsion that could be generated from the power plant. The greatest limiting factor with the power plant was that the plane could only carry five pounds of batteries and pull 40 amps during flight. This fixed the feasible range of weights based on the preliminary thrust to weight ratio of 3:1. From these limiting variables, the aircraft was designed and optimized for maximum performance.

In addition to the weight buildups, some initial materials and propulsion testing was completed prior to the conceptual designs. This gave the team members a better idea of what they had to deal with in these two areas.

The materials testing consisted of constructing several different rectangular test samples that had the same width, length, and thickness. The rectangle specimens consisted of a core material with a composite covering on either side. Both the core materials and the composite materials were varied for each specimen. They were then point loaded until failure occurred, giving the team members a good idea of what load each composite and its accompanying core material could take.

The propulsion testing was conducted mainly to validate the data that ECALC was giving for different engine, battery, and propeller configurations. The team set up a test with an Astro 40 size engine and a 12x12 propeller that were available. The propeller efficiency actor was then determined from these tests, and was found to be slightly better than the one being used in ECALC. This new efficiency factor was then used in subsequent propulsion setups that were tested in ECALC to give a better idea of what the real results would be.

From this collected data, weight buildup, and assumed historical values for the major aircraft characteristics, each team member came up with their own conceptual design. Each design is discussed below.

3.1 Midshipman Park

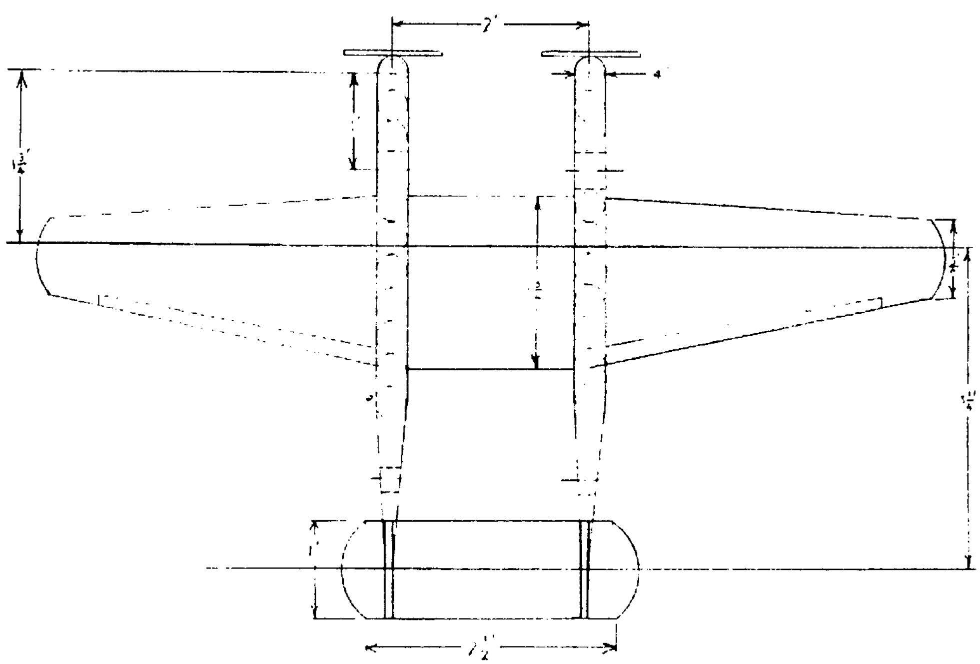
Dong Park's basic design was a twin-motor, twin-propeller airplane. By using two propellers, the design did not need a high clearance by the landing gear. Instead, the twin-prop design allowed the diameter of the propellers to be 12 inches, as compared to around 22 inches for a single engine. Since the landing gear only needed to provide a six-inch clearance, it was integrated into the fuselage shown in Figure 3. Two rows of ten softballs were then placed in the two slender fuselages. Therefore, it made the most sense to have two vertical tails, one of each fuselage, for the best yaw stability. The horizontal tail was placed crossing both of the vertical tails. The size of the horizontal tail was calculated to be 2.56 ft^2 using a tail volume of 0.5.

After the entire weight build-up, the empty weight of the plane came out to be 12.22 pounds and the weight of the 20 softballs came out to 8 pounds.

Because of the twin-propeller design, it was practical to apply two Astro-C40 motors in a parallel circuit. A 59/19 (.31) gearbox was also fitted to the motors. The propeller was chosen using ECALC. The 12-inch, 6-degree pitch propellers provided the best performance in terms of endurance and thrust based on the results that ECALC turned out.

Figure 3

Park's Twin Engine Conceptual Design



Based on the required amount of lift, stall characteristics and the desire for a realistic aspect ration, a NACA 4412 airfoil was chosen for an airfoil section, and the wingspan stretched 9 feet with a wing area of 12.25 square feet. The root chord was 1.75 feet and the tip chord was 0.75 feet, while taper was calculated to be 0.43 and the aspect ratio was 6.6. A dihedral of 4 degrees was built in for stability, as this was the best compromise between roll dynamics and lift produced. Using the wing root bending moment spreadsheet, the maximum bending moment was determined to be 1033 inch pounds, which was approximately 86 foot-pounds. Since all of the preliminary testing showed that wing failure would occur in compression on top of the wing, only a thin layer of fiberglass would be put on the bottom of the wing. On the top of the wing, 6 layers of carbon fiber would be required at the root and one layer at the tip.

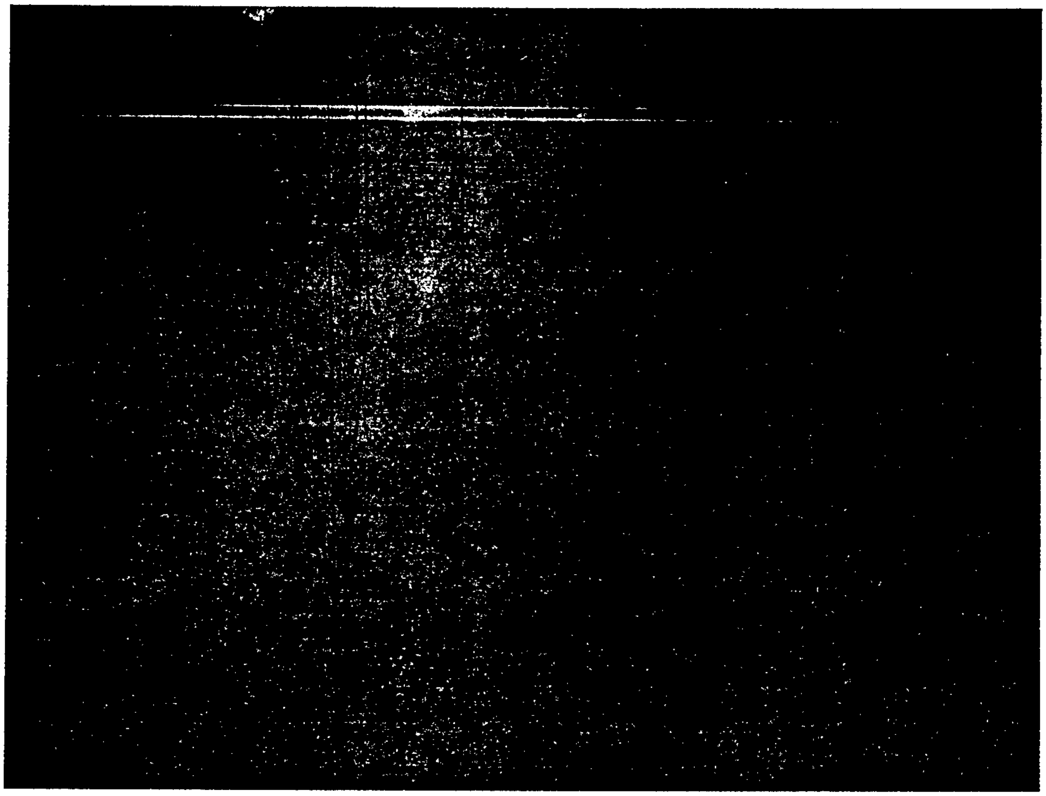
Using the DBF program, the total number of minutes to complete the mission came out to be 10 minutes and 9 seconds. Also, the motors used up 2366 mAh at the end of 10 minutes and 9 seconds. Finally, performance wise, the take-off distance was calculated to be 79.8 feet and the stall speed was near 20 mph. While this design appeared to have some good qualities, it was later found to be in violation of the rules.

3.2 Midshipman Swift

Midshipman Swift's design for the DBF competition aircraft was based on a gross weight of 31.4 pounds. This value was reached by finding out the weight of last year's plane, estimating the weight of the softballs that will be carried this year, and then estimating the total weight. A wing loading of 40 oz/ft^2. was chosen to find a value for the required wing area. The wing area was found to be 12.56 ft^2. Most general aviation aircraft have aspect ratios from 6 to 7, so an aspect ratio of 6.8 was chosen for this aircraft. From the aspect ratio and wing area, the necessary wingspan was found, and with it the mean aerodynamic chord. A straight tapered wing with a tip-to-root chord ratio of 0.5 was chosen. This resulted in a root chord of 1.81 ft and a tip chord of 0.907 ft. Cruise speed was calculated by taking the entire length of the flight course and using the time limit to find the absolute slowest speed that could be used. The resulting cruise speed was 35 mph. Combining this speed with the MAC allowed for the Reynolds number over the wing to be determined, and the result was about 442,000. The Reynolds number was then used to help chose a suitable airfoil for the wing. The airfoil that seemed to have the performance that best suited the design requirements was found to be the RAF 32. Two motors were decided on for propulsion, each turning a 22x22 propeller and drawing 40 amps max from five pounds of batteries. A low wing, tail dragger configuration was chosen for simplicity of construction. The rated aircraft cost for this design was determined to be $12,000. Figure 4 shows Midshipman Swift's conceptual design in detail.

Figure 4

Swift's Twin Engine Conceptual Design



3.3 Midshipman Scheu

Midshipman Scheu's design weighed the costs and benefits to the number of softballs carried, and determined that while 24 balls would be desirable, the aircraft would simply be too heavy for the amount of thrust possible. Therefore, 18 softballs were chosen which set the aircraft gross weight at 25 pounds. This reduction in the number of balls not only saved weight in terms of payload, but also allowed the aircraft to be much smaller, thus saving weight in materials for a larger wing area and empennage section.

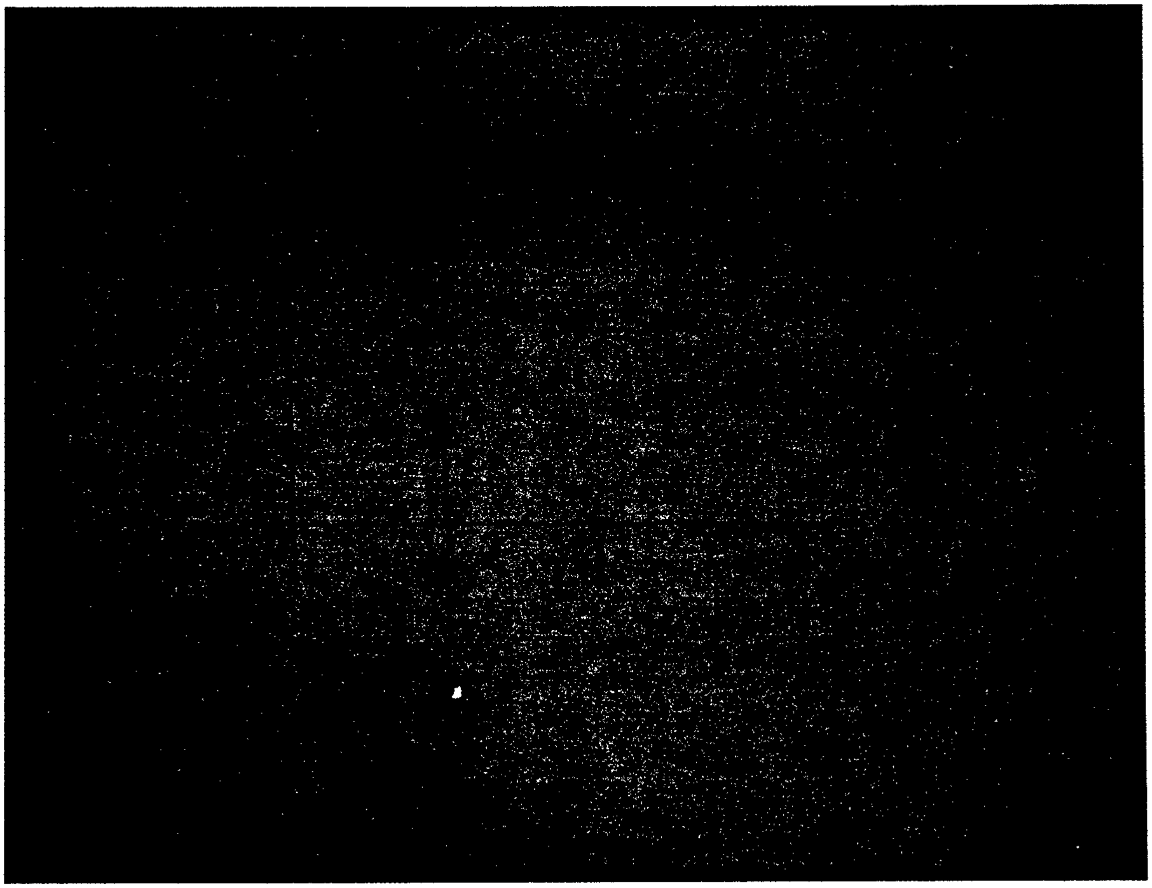
After running MFOIL and determining the lift and drag characteristics for different airfoils, it was apparent that the NACA 2412 would be a well performing airfoil given its stall characteristics. Once the NACA 2412 was chosen, data was complied on the section to include maximum lift coefficient and cruise lift coefficient. From this maximum lift coefficient, the maximum needed value for the planform wing area resulted in 1491 square inches for a 30 ft/s landing. Based on the ability to go slightly faster for landing, the wing area was then 1400 square inches

Next, several characteristics of the wing were chosen. The first was taper ratio, which was determined to be .5, since this taper ratio gives the best approximation to the most efficient lift distribution. Secondly, the aspect ratio was determined. High aspect ratio was good for cruise speed since it allows for

low induced drag, but the only drawback was structural concerns, so an aspect ratio of 11 was chosen. Because of the availability of carbon fiber and the strength that it offers, such an aspect ratio was not unreasonable at all, and the structural factor does not play in as much. The span was then found to be 124 inches with an 11.3-inch mean aerodynamic chord. Figure 5 gives the three view drawings of Scheu's conceptual design.

Figure 5

Scheu's Twin Engine Conceptual Design



The actual construction of the wing and fuselage was based on the strength properties of carbon fiber. It was apparent that little material would be needed, as the provided carbon was so strong. Slight reinforcements at the landing gear and engine placements, along with the joining of the polyhedral would support the stresses while the rest of the wing outboard could be covered simply in a single layer of thin fiberglass, as this was the lightest material possible that could be used as a covering.

Knowing the wing area and the weight for the aircraft, the power plant could then be determined. The most optimum design was determined using ECALC, and it was found to be with two Astro 40 size engines mounted on the wings. The combination of these two engines, wired in parallel with 36 cells rated at 2500 milliamp hours turning two 13x11 propellers gives a resultant static thrust of 137 oz, or 8.56 pounds, and this configuration also gave a maximum speed of 73 miles per hour at 40.6 amps. This data

was then put into the DBF program, and some performance characteristics of the plane were determined. At maximum velocity with a drain of 40 amps for most of the flight, the aircraft finished the course in 7.18 minutes using 8000 amp seconds. This also takes into account a softball loading and unloading time of 90 seconds. When these numbers are substituted into the equation used to calculate the aircraft score, the result was 3.68.

3.4 Midshipman Sword

For Midshipman Sword, the first challenge that was encountered was to figure out how much wing area to use. To determine this, a gross weight of 27 pounds was used, based on a weight buildup using the weight of many of the known components of the aircraft. This gross weight included a payload of 20 softballs. Because most planes fly at a coefficient of lift of around 0.4 to about 0.6 while in cruse, and assuming the flight velocity to be at around 40 mph, then a wing area of $1800in^2$ would be needed. However, after more consideration, because this large amount of wing area was unnecessary for the high-speed operation desired, the wing area was reduced to $1000in^2$. That not only reduces the minimum drag coefficient, it also allowed for higher speeds around the course. A slightly tapered zero swept wing was used, as this provided more efficient lift but was also simple to construct.

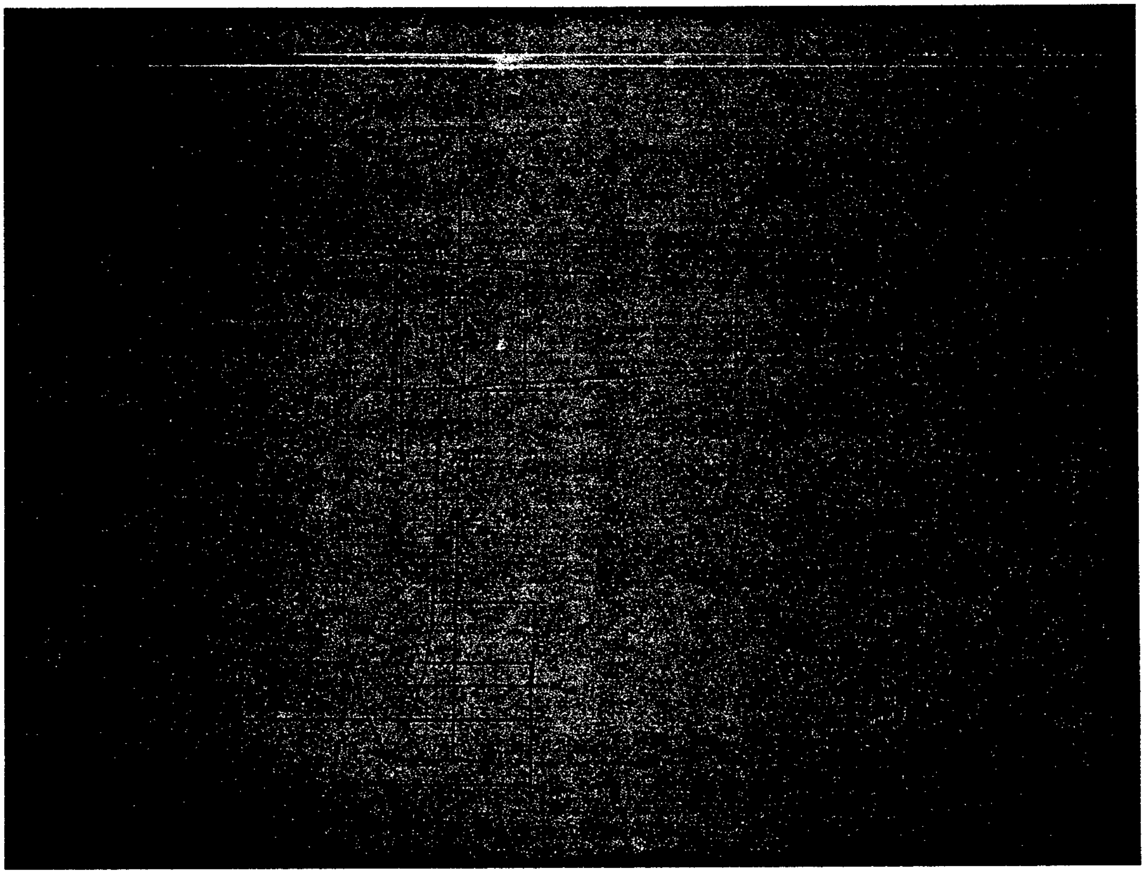
The next aspect of the design of the conceptual aircraft was choosing a power plant. While a twin engine design would allow for less ground clearance, it was thought that a single engine and propeller would give better thrust, because of its large diameter. To get the required thrust, at 28-inch diameter propeller with a pitch of 16 degrees was used.

The score from DBF was 3.74 with all of these characteristics. However, the amount of amp-seconds used was over 8,800, which meant that the aircraft would most likely not finish the course. As a result, this design was left out later on in the more design processes.

Figure 6 on the next page gives a detailed drawing of Midshipman Sword's conceptual design.

Figure 6

Sword's Single Engine Conceptual Design



3.5 Midshipman Frantz

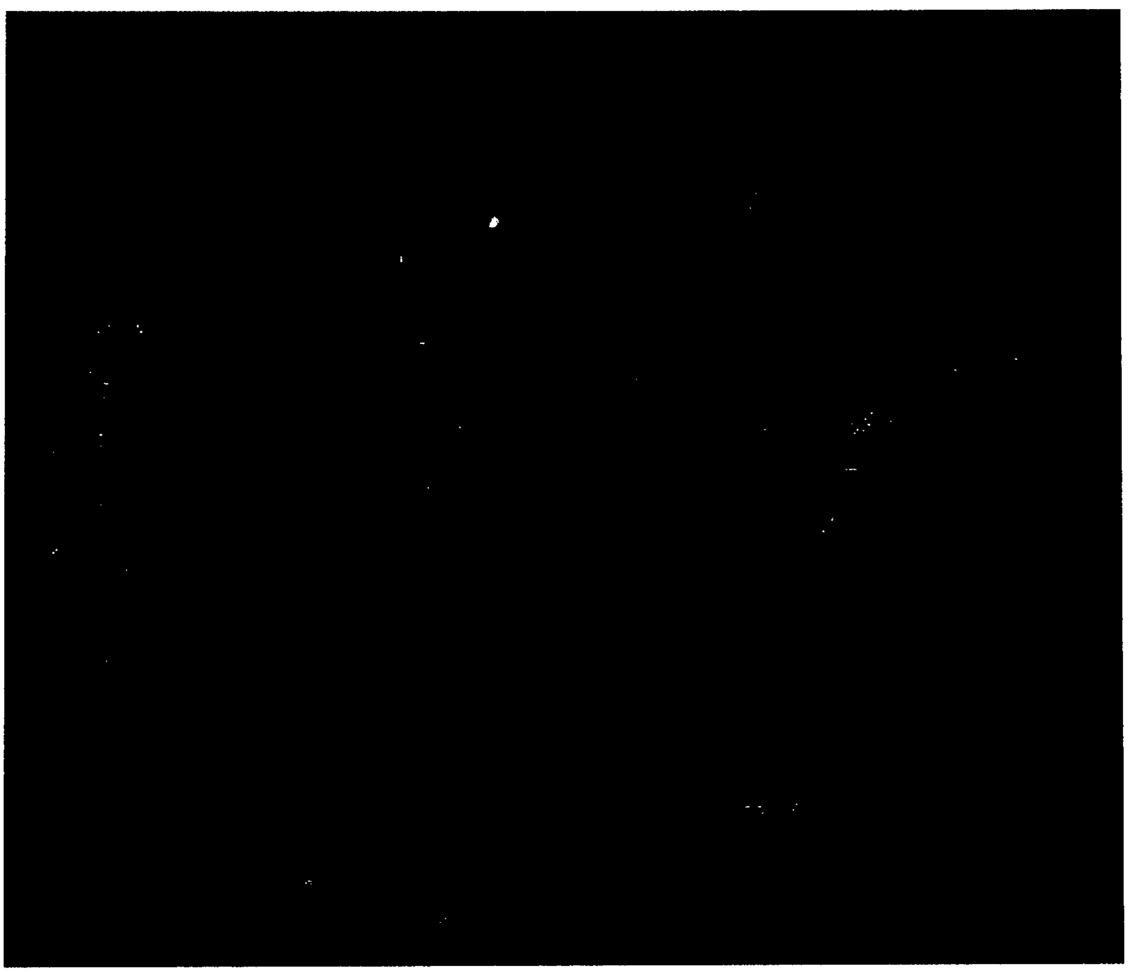
Midshipman Frantz's design for the DBF competition aircraft was based on an initial gross weight build up of 26 pounds and a twin engine propulsion plant. The weight was calculated by weighing various parts of the 2001 DBF team's aircraft, such as the fuselage and wings, and then making approximations based on the estimated size of the aircraft desired. The approximate weights where then added together to create an initial weight estimate. The wing airfoil was then chosen to be a NACA 4810 because of its high lift capability and smooth stall characteristics. The airfoil lift capability, a Cl of around 2.0 with flaps, was used, along with a desired stall speed of 25 miles per hour to calculate the required wing area. The worst-case wind speeds expected during the competition were estimated to be about 15 miles per hour, so a stall speed of 25 miles per hour allowed for forward flight even in these conditions. The required wing area under those conditions was 1171 square inches, rounded down to 8 square feet. Most general aviation aircraft have aspect ratios from 6 to 7 and most cargo or passenger aircraft have aspect ratios of

15 to 20, so an aspect ratio of 8 was chose for this aircraft in order to compromise between lower drag and structural integrity.

From the aspect ratio and wing area, the necessary wingspan was found, and with it the mean aerodynamic chord. A tapered leading edge and a straight trailing edge with a tip to root chord ratio o f.5 was chose, since this was the most efficient. Deciding on a cruise speed lead to calculating the entire length of the flight course and then using the time limit to find the absolute slowest speed that could be used. The resulting cruise speed was 65 mph. In order to fly at this speed with two motors, 10x12 inch propellers were chosen based on calculations made in ECALC. A mid-wing configuration was chosen for its maneuverability and positive combination of high and low wing characteristics. Lastly, a leading edge extension and winglets were added in order to create forward lift and add more effective wing area while keeping the same wing and aspect ratio. From the equation given in the DBF rules, the rated aircraft cost for this design was $15,400.

Figure 7

Frantz's Twin Engine Conceptual Design



3.6 Conceptual Design Summarization and Figures of Merit

Conceptual designs submitted from each member were used the DBF program for analysis. By comparing each designs, the team was able to gain a feel for which design parameters would make a better airplane. The five conceptual designs that the USNA 2002 DBF team came up with are summarized below.

Table 2

Conceptual Design Parameters

	Scheu	Swift	Park	Frantz	Sword
# of motors	2	2	2	2	1
type	c40	c40	c40	c40	c90
propeller	13x11	22x22	12x6	12x10	28x16
# of fuselage	1	1	2	1	1
wing design	polyhedral	dihedral	dihedral	mid-wing	dihedral
airfoil	2412	RAF 32	4412	Clark Y	4412
wing area(in^2)	1400	1809	1764	1000	1000
Aspect Ratio	11	6.5	6.6	7.1	9
taper	0.5	0.5	0.43	0.5	0.5
# of balls	18	16	20	18	20
total weight	25	31.4	20.22	26	27
Time	390	390	420	376	417
Amp Sec	7496	9280	6693	6534	8869
Score	3.689	3.381	3.716	3.834	3.7416
Rated Aircraft Cost	$16,105	$16,065	$16,245	$16,025	$13,950
Rank	4	5	3	1	2

3.7 Assumptions Made and Justification

The design parameters were used in the DBF program for each aircrafts performance. Before the team started using the DBF program, several assumptions were made and consistent flying parameters were defined. Consistency in the comparisons was necessary so all the planes could start at the same playing field. First, the total aircraft weight was assumed to be the same at 26 lbs because the conceptual weight estimates varied from 20-31 lbs. Because the conceptual designs looked very similar in streamlining and elliptical wing design, the team also assumed that Cd_o was 0.05 and e was 0.70. The same battery and motor configurations were also used, except for Midshipman Sword's case. Midshipman Sword had a single motor design so he used an Astro C90 motor instead of the two C40 motors for everyone else. The parameter that had a major influence in the performance was the wing area. By having a smaller wing area, the plane can cruise faster and achieve a higher score. Also, aspect ratio improved the overall score.

In terms of propulsion, the team was limited only by 5 pounds of batteries and any design they could come up with that would not exceed 40 amps. However, no limiting factor was place on the amount

of amp seconds available. This was because the team was overly optimistic when it came to thinking they could find high mAh NiCad batteries. Later, the team found that the batteries that were available for purchase did not have large amounts of amp seconds available, so some of the propulsion designs had to be streamlined.

In the end, all of these different designs resulted in different scores, and each member of the team learned both good and bad ideas from each other.

4.0 Preliminary Design

A panel made up of faculty, facilities personnel, and the student pilot reviewed all of the conceptual designs. Based on the panel's input, each student gained a better understanding of the entire design process and was able to put this knowledge to good use once the teams were assigned and each team began its preliminary design.

4.1 Design Parameters and Sizing Trades Investigated and Their Importance

Team Tango began its preliminary design process by studying the advantages and disadvantages of each team member's conceptual design. In the preliminary design, the team determined that drag was, the most important design consideration and minimizing drag was of extreme importance. Drag directly influenced the total amperage, and any reduction in drag both increased the total amount of time the aircraft was able to fly and made it capable of flying faster, each of which positively affected the overall flight score.

However, before the team began deciding on which aspects of preliminary design to pull from the conceptual designs, it was necessary to determine the maximum number of softballs that the team's aircraft would carry. Because the weight buildups varied from person to person during the conceptual design phase, the number of softballs that each aircraft could carry also differed. Team members were not in agreement on this issue, making resolution necessary before design work could continue so that there was some standardization among designs.

One great advantage available to the team after the conceptual designs had been submitted was the DBF simulation program. Almost every figure of merit was explored using the program in order to help determine the preliminary design parameters for the aircraft. The program allowed the team to design theoretical aircraft and input almost every design characteristic, including empty and gross weight, wing area, aspect ratio, zero lift drag count, wing efficiency factor, wing taper, and maximum lift coefficient. Additionally, the program included the required flight course and the aircraft's estimated velocities, along with turning speed and radius. DBF also allowed for testing different power plant configurations in tandem with ECALC. These variables included available amperage, available volts, static thrust, cruise thrust, takeoff amperage used, and the number of cruise, turn, and landing amps.

The propulsion plant could therefore be maximized in ECALC and then tested in DBF to see how these, along with the aerodynamic and flight characteristics, affected the overall aircraft performance in terms of amp seconds used and time to fly the course. Based on these outputs, the design could be maximized before the team even began looking beyond basic aircraft characteristics.

4.2 Figures of Merit Used & Mission Feature of Each Figure of Merit

When evaluating the preliminary design in DBF, several factors were constantly altered in order to create the optimal design. It was found that the aircraft's wing, weight, payload size, and cruise speed had the largest effects on the overall score and amperage used. The wing's aspect ratio was found to greatly influence the total amperage used through induced drag reduction, and other wing design considerations did not prove to have a large affect on overall score. Weight directly influenced the wing design in terms of area necessary to generate sufficient lift. Payload size had a very large impact on the overall score and determining a payload that optimized the score without excessive adverse affects was of vital importance. Cruise speed directly influenced the flight score and depended heavily on the battery run time and drag. A higher cruise speed increased the flight score but was limited by the battery amperage for a required thrust, and drag.

The aspect ratio had the greatest influence on aircraft performance and flight score, according to DBF. An aspect ratio increase decreased the number of amps used by the motors and vise versa. While the team would have liked to build an aircraft with a very high aspect ratio, the impracticality of building such an aircraft was realized once the structural aspects of the design are considered. The group finally settled on an aspect ratio of eight based on the results from DBF and the consideration that an aspect ratio of anything greater than eight simply did not provide enough benefits to outweigh any losses caused by the increased structural weight.

Another factor that changed many times was the weight of the aircraft. Since the completed weight build-up that was done earlier could not be fully relied on, it was changed in DBF to see how it would affect the overall score and, more importantly, amperage used. It was found that the weight of the airplane had little effect on the total number of amps used, as long as the wing area was kept constant. The wing area chosen was based on the lift necessary for a 2G turn, and the effect of changing the weight for this wing was minimal for both the score and battery pack. The results of the weight buildup are given in Table 3.

Table 3

Weight Buildup

Item	# of Items	Weight of Each(lbs)	Total Weight(lbs)
Softballs	18	0.4	8
Battery	35	0.14	5
Motor w/gear (C40)	2	1.06	2.12
Propeller (12inch dia.)	2	0.1	0.2
Wheels (Front)	2	0.15	0.3
Wheel (Rear)	1	0.1	0.1
Fuselage	1	3	3
Wing	1	3	3
Tail (Horizontal)	1	0.75	0.75
Tail (Vertical)	2	0.4	0.8
Total			23.27

The next factor adjusted was the number of softballs carried in the aircraft. Adjusting this number had a very large effect on the overall score, but not enough to justify putting the maximum number of balls in the aircraft. Carrying twenty-four balls was possible, but the margin for error was too small with this large number. Because the environmental conditions could potentially vary from little wind to very windy, if the plane was under built in cargo carrying capacity, the score would be made up for with speed during the competition. An increase in the number of balls required a size and weight increase that ended up outweighing the benefits of the increased payload. The number of softballs and plane size were adjusted to find the optimal number of 18 softballs. However, a small margin was designed into the plane such that if battery or motor performance were better than predicted, the plane would have enough room to carry at least two more softballs.

The last variable adjusted was the cruise speed. This was the least dependable variable within the program because the actual performance of the motors and the amount of drag generated were the least accurate constants at this point. A flight envelope was therefore tested in DBF and an optimal flight speed of 50 mph was found. Speed had a very large effect on battery usage because the faster the simulated course was flown the faster the batteries were drained. This was because the amps used were proportional to the drag on the aircraft, which was proportional to the square of the plane's velocity. Turning flight was determined to be the greatest drain on the batteries. However, the plane was designed with the ability to fly at speeds between 25 and 70 mph, so a cruise speed of 50 mph left enough battery

power to provide a safety margin. It was decided within the team that it would be better to land with some extra power left in the batteries than to run out and suffer a landing that ended the aircrafts useful life. Based on the conditions at the time of the competition, the team would decide how fast they could actually drive the airplane and still land with some amount of power remaining.

From these inputs, the team initially determined that 20 softballs would be the maximum amount that the team could possible carry and still hope to get around the course and finish with some margin. Unfortunately, the amount of amp seconds available was initially over estimated because the team did not consider actually having to find batteries like this available for purchase. When the best batteries that were available were purchased, to ensure that the team had enough margin for error as discussed, it was decided to remove two softballs from the maximum payload, leaving a new total of 18 balls.

After running DBF several times, constrained by setting the available amp seconds to 6500, it was determined that 18 balls was, indeed, the most that could be carried. The team then determined the size of the wing and fuselage needed and to find the best propulsion plant available. All decisions were based on a weight buildup that yielded a 17-pound empty aircraft. When loaded with 18 softballs, the gross weight came to 25 pounds, assuming each softball weighed 0.44 pounds.

4.3 Wing and Power Loading Requirements

The next major aspect of designing the 2002 competition aircraft was to come up with a wing that maximized every positive area of aerodynamics. However, as with all of aircraft design, many compromises had to be made in order to have the best performance characteristics where it was needed. Resultantly, the wing design was a combination of compromises between maximum lift, minimum area, and minimum drag.

For Team Tango's 2002 Design-Build-Fly aircraft, the wing needed to provide at least 25 pounds of lift while still being able to take-off within the required distance and land at a reasonable speed. Wing loading was limited by two factors. The first factor was how many G's the aircraft would be pulling in the turns of the flight pattern. The second was the strength and weight of the materials available for construction. It was determined that turning flight and wind gusts would be the main area of concern for G forces acting on the aircraft, and that the maximum it would most likely be subjected to would be four G's. The aircraft did not have enough power to pull any more than four G's. At the same time, it was felt that the wind gusts most likely would not subject the aircraft to more than four G's unless the conditions were extremely hazardous, in which case flying the aircraft would be extremely difficult in the first place. In terms of materials strength, the primary concern was saving as much weight as possible while still making the aircraft able to pull the specified number of G forces. The composite materials available were tested and showed the ability to provide strength far beyond the amount of G's the aircraft could pull based on its available power. However, using the minimal amount of composites necessary to construct the wing was very important, as extra strength was simply excess weight. So, the aircraft was designed

to have a design load of 4 Gs with a limit load of 6 Gs, as these numbers allowed the best flight performance characteristics while still staying in the feasible structural regime.

4.3.1 Wing Characteristics Chosen and Why

One of the main factors in choosing the airfoil section was to provide the necessary lift for landing and takeoff while at the same time reducing drag in cruise configuration. After initial investigation, the airfoil section chosen was the NACA 5412. The team felt that the 12% thickness in the wing was necessary to generate the required lift needed for the specified flight conditions. In addition, the added thickness gave the wing extra structure and therefore strength so it could support the loads that the mounted motors would place on it. The camber was chosen to help generate lift, and at the same time, make it easier to attach components such as the landing gear and the motor mounts. The wing included a slight over-design in terms of size and airfoil selection to allow for a margin of error that would counter any adverse weight that might have been included in the manufacture of the aircraft, or weight creep.

Looking into drag reduction, a mid-wing mounting system was chosen. In terms of construction, this added complexity, but it was deemed acceptable in that it allowed for a reduction in interference drag. The wing was also blended into the fuselage to reduce possible drag eddies.

The mid-fuselage design was more stable than a low-wing configuration and maneuverable than a high-wing. As the mid-wing design was not as laterally stable as a high wing, it was decided that a polyhedral wing would be necessary to add a restoring moment during banked flight. Polyhedral was chosen instead of dihedral because the motors were to be mounted in the wings. The area where they were to be mounted, 13 inches out from the fuselage, had to be reinforced to carry the additional loads. As a result, it became convenient to include any bend in the wing at the motor mounting points. In addition, it would have been much more difficult to attach the wings mid-fuselage with full dihedral.

Aspect ratio was an area of debate within the group. In full-scale aerospace applications, the aspect ratio was most often limited for structural reasons, as most traditional materials are not capable of sustaining the large bending moment associated with high aspect ratio wings. However, in this small-scale remote controlled application, composite materials such as fiberglass, carbon fiber, and Kevlar, allowed for the creation of extremely strong, lightweight structures. The predicted bending moment was not large enough to be a major concern, so it was structurally possible to build an aircraft with an extremely high aspect ratio. This would have reduced the number of amps pulled from the batteries but the stiffness of the wing would begin to suffer. One of the main mistakes that last year's DBF team made was that their wing was too flexible and deformed in flight, therefore making it unpredictable. So, the team chose an aspect ratio of 8 as a compromise.

To create the high lift needed for a slow landing approach, flaps were included into the design. Without the flaps, the wing section chosen did not create enough lift to allow the plane to land at a reasonably slow speed. The flaps provided the large lift coefficient of about 2.3 for the slow landing speed. They could also be used on takeoff to ensure that the aircraft was able to take off in the distance

specified and climb to altitude rapidly. While it was predicted that the headwind at the competition would be enough to get the aircraft off the ground, it was much easier to build the plane with flaps and not use them than to have to remove payload because not enough lift could be provided without headwind. The flaps were designed with a chord that was 15% of the wing chord and spanned from the fuselage to the wing joint.

The wing was also designed to include winglets with a symmetrical airfoil section. The high aspect ratio was intended to reduce induced drag to some degree and so that the team felt another reduction could be accomplished without a bending moment penalty. The winglets were six inches high, tapered from a root chord of 7.2 inches to a tip chord of 3.6 inches, and were attached at an angle of five degrees outward with reference to the fuselage.

Another important aspect of the wing was the fillet added just forward of the leading edge. The fillet was added to improve performance during the high angle of attack maneuvering that would occur in the turns. The fillet was 12 inches long and extended six inches out along the wingspan. Originally the team felt that a leading edge extension would serve the aircraft best, as it would provide increased wing area close to the fuselage that would serve to reduce wing root bending moments. In addition, the leading edge extension was hoped to create vortices, and resultantly, turbulent flow over the largest part of the wing, increasing lift at higher angles of attack. However, the leading edge extension would have effectively reduced aspect ratio, something the team had been trying hard to increase. Therefore, after conferring with the Navy's chief test pilot of the F/A-18 Super Hornet, an aircraft that made extensive use of leading edge extensions, a compromise was reached. An attempt to gain the advantages of the leading edge extension while reducing the disadvantages as much as possible seemed possible using the fillet.

Structurally, the wing was designed to disperse the bending moment as much as possible while making manufacturing easy. Several main areas of the aircraft that would have to be strengthened structurally in order to support the loads created by the motors, landing gear, and the polyhedral attachment of the wing were put in the same place so that no more additional material was needed, thus saving weight. The wing from the fuselage to the motor mount, to include the center tubular part of the fuselage, was all one straight piece for structural stability. This piece was to have a shear web and a top and bottom spar to take the large moments at the quarter chord. The outer wing section was joined to the center section at the motor mounts.

4.3.2 Power Plant

Team Tango was founded on the idea that twin motors would be the best way to provide propulsion for the 2002 Design Build Fly aircraft. The computer program ECALC was used to determine the best setup for two motors. After comparing the results of the single and twin-motor propulsions options in the conceptual design phase, it became apparent that two motors, wired in parallel, would give

more thrust than a single motor. In addition, a rather high theoretical propeller speed in ECALC - approaching 70 miles per hour - could be achieved.

The first main benefit was the number of amps that the motor pulled at a certain thrust rating. Because of the 40-amp fuse that had to be installed for the competition, any motor and gearing combination that had a high thrust and rated speed, but exceeded the 40 amps, was unacceptable. Compared to the single motor combinations, the twin 40s wired in parallel outperformed the single 60 or 90 sized Astroflight motors. In addition, the 800 watts per motor in power that would be delivered to each 40-sized engine would allow the engines to run relatively cool. The reliability of an Asto 60 sized motor with 1.5 kilowatts being delivered to it was questionable, and the team did not want to have to worry about an engine overheating and seizing up in the twin-engine configuration, as controlling and landing the aircraft would become rather difficult.

A very important consideration for using two motors was the availability of the parts needed. After looking into what Astroflight had available for gearboxes that were compatible with their motors, it was determined that the Superbox with a 3.1:1 gearing ratio would work well with the 40 size motor to get the desired thrust within the amperage limits. However, in order to get the necessary thrust from a 60 or 90 sized motor, a special gearbox was needed and would have been more difficult to find or manufacture. As a result, the team had to go with what was available. Fortunately the best setup was, in fact, the easiest to purchase.

Considering weight, two motors did not weigh much more than one 60 motor, and almost exactly the same as a single 90 with a gearbox. The weight comparison of the different motors considered is shown in Table 4.

Table 4

Engine Weights

Motor	Weight of 1 Motor (lbs)	Weight of 2 Motors (lbs)
C40	1.06	2.12
C60	1.72	3.44
C90	2.14	4.28

Table 4 shows that the choice of two forty sized engines weighed the same as a single 90 and a little more than a single 60 sized engine. However, the additional advantages of the twin-engine design in conjunction with those previously discussed outweighed the small weight saving that a 60-sized engine offered. These other advantages were first was that two, relatively small diameter propellers can be used. This allowed for shorter landing gear while still maintaining ground clearance. Since the landing gear could be close to the fuselage, it greatly reduced the moment arm at which the landing forces would act on the gear and wing. As a result, the gear could be lighter and still strong enough. This resulted in a savings in weight, both in the gear and in the reinforcements.

Next, the fuselage structural concerns are much less with two motors. Instead of having to worry about a firewall in the forward part of the plane that would add complexity and weight, the two motors were mounted on the wings. The motors' smaller size and the relative strength of the available building material made it possible to build light and strong motor mounts that were inserted into the wing. This also made use of the main spar in the wing, an already structurally sound member, as an attachment point for the motors.

Finally, the twin motors, which were mounted just forward of the wing's quarter chord, were very nearly in line with the aircraft's center of gravity. This greatly simplified ensuring that the center of gravity lay near the center of lift of the wing. If the weight of a single motor were in the nose, it would have been more difficult to balance the plane about its center of gravity. Additionally, the propellers and the gearboxes are protected from ground prop strikes, which are a real possibility in early flights.

In the conceptual design phase of the project, thrust to weight ratios from 0.3 to 0.5 were considered. While static thrust was important primarily in takeoff performance, a good deal of available thrust would be needed if the aircraft stalled or drag in the turns was excessive. Based on the weight build up discussed previously, the aircraft needed approximately nine pounds of thrust. This could be achieved with the twin-engine setup, however endurance became the next issue to be addressed.

The batteries finally chosen for use in the aircraft were 2400 mAh matched cells. For these cells, the manufacturer's data showed that the average cell voltage was 1.126 volts and the discharge time while drawing 30 amps was 288 seconds, or five minutes.

The next major step of designing the propulsion plant was determining the propeller to be used. According to ECALC, more thrust was obtained though a larger diameter propeller, and a higher speed was obtained through an increased pitch. However, the design was not complete by simply using a large diameter, high pitch propeller because efficiency of the motor-propeller combination had to be considered. For propeller size, the team was limited to an 11 to 14-inch propeller, a range that provided the best performance in terms of endurance and thrust. Using the data collected from ECALC, the best propellers were the 13x9 and the 13x10 propellers manufactured by APC. Two 13x9 propellers provided the best results with the highest maximum speed and sufficient thrust for take off. A three-bladed propeller gave better results that the two-blade, but efficiency suffered and it was difficult to find three-bladed propellers. Since the initial weight estimates were not final, different types of propellers were to be tested in the wind tunnel. To include a few, they were the 11x8 3-blade, 11x12, 12x12, 12x11, 12x10, 13x10, 13x 9, and 14x10 2-blade propellers.

While ECALC was an extremely valuable tool that could perform all of the mathematical calculations, the program's results were totally relative and could be compared only to each other. In addition, the propeller efficiency of 80% that was used in ECALC to come up with some of the propulsion data was simply an educated guess. As a result, it was not known for sure what the real world results would be, and it became necessary to test a real propeller in the Naval Academy's Eiffel wind tunnel. The

data, taken for one motor at half the amperage (as both motors were wired in parallel), is given in the following table.

Table 5

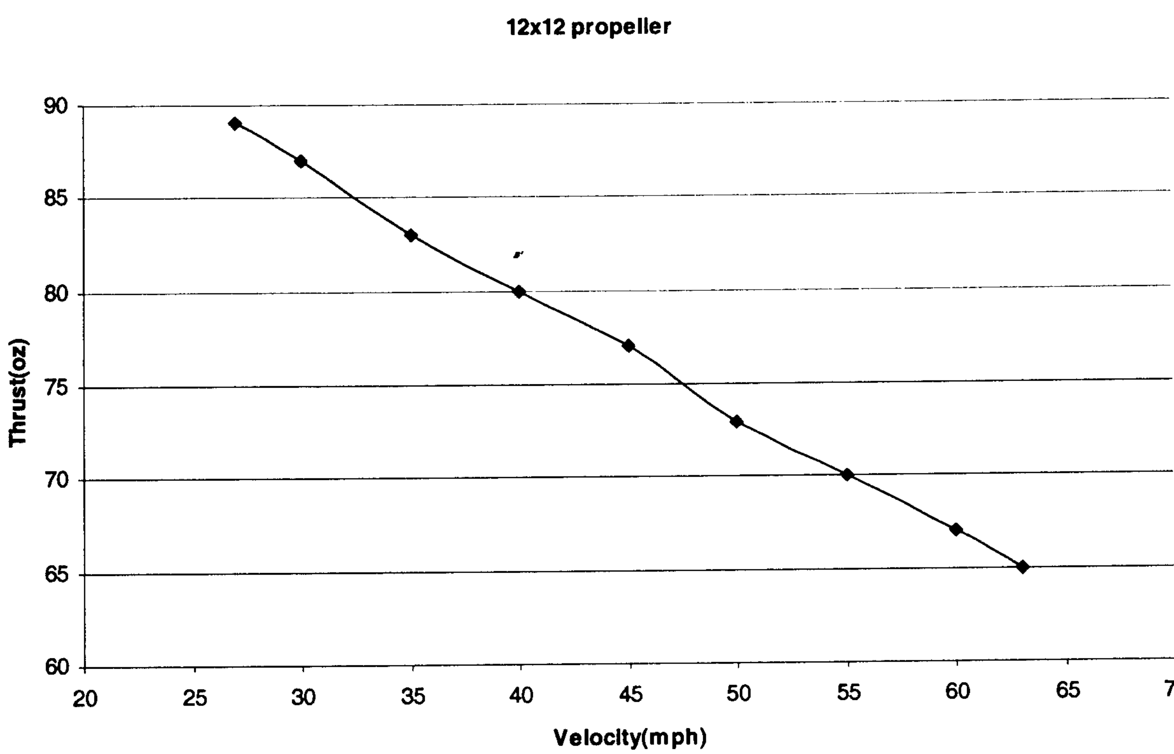
Resultant Wind Tunnel Test Data for 12 x 12 Propeller

	Velocity = 0		Velocity = 20mph		Velocity = 40mph		Velocity = 60mph	
Amps	*Volts*	*Thrust*	*Volts*	*Thrust*	*Volts*	*Thrust*	*Volts*	*Thrust*
20	30	4.3	29.7	4	28.9	3.1	29.3	2.4
18	27	3.6	28	3.5	26.5	2.7		
14	22	2.48	23	2.6	22.5	1.8		
12	19.9	2	20.2	1.94	20	1		

Once this data had been obtained, it was compared to what ECALC predicted. A graph of the thrust that the 12x12 propeller provided versus the velocity in the wind tunnel is given below.

Figure 8

Test Thrust and Velocity Comparison



In addition, a comparison table for the output thrust from ECALC and the output thrust obtained from the test follows in Table 6.

Table 6

Theoretical & Test Data

Amps	Actual Velocity	Actual Thrust	ECALC Thrust
20	30mph	4.3lbs	3.9lbs
18	27mph	3.6lbs	3.6lbs

From this data, it can be seen that the initial estimation of 0.8 for propeller efficiency was slightly lower than the actual result. The first test showed that 0.84 was a more reasonable efficiency to use. This result was, of course, rather fortunate, as it gave more thrust at all velocities than was originally planned for. While the amount of thrust was not a great deal, it allowed for more margin of error during flight in case the aircraft was put into a stall situation, in which it can help the pilot a great deal.

4.4 Assumptions Made and Justifications for Their Validity

The assumptions made in this section of the design project were focused largely in one area—the validity of the computer programs that the team had at its disposal. As previously discussed, it was apparent that Team Tango relied heavily on computer programs written for model applications to determine many of the preliminary design characteristics. The three main programs used were the DBF aircraft performance program, ECALC, and MFOIL. In using these programs, however, was assumed that they were, in fact, accurate, and that they provided a good approximate model to the actual expected aircraft performance. While computer models can provide great insight into how well conceptual designs can function in the real world, there are two major issues with relying so heavily on them. The first was that a computer program could never model the real world, since there are too many variables that can be altered in too many ways to hope that a single program could accurately depict reality. In addition, the team had limited experience with the validity of these programs, especially the DBF program, which had never been tested before. So, if there were any large error in any of these programs the team would be designing a plane that would not perform as assumed.

However, in the design phase of an aircraft, there were so many variables that had to be considered that the use of a computer program to perform all of the long and tedious mathematical calculations was more than justified. In addition, the programs were used simply as models for performance and were not relied on as the "end all – be all" of the aircraft, but rather as an approximate solution. This solution was then used to make other decisions as to how to proceed with the preliminary design when compared with factual and other historical data. The team also employed the "common sense check" to ensure that the solutions did, in fact, make good sense based on the inputs.

There were also two other aerodynamic and propulsion assumptions that were made based on available and historical information. First, the Oswald efficiency factor, which the team estimated to be around 0.65, was an input into the original design program. While this efficiency factor was merely an assumption, it was based on past data and recommendations given by the advisor. Next, the zero lift

drag coefficient, which the team said was 0.055, was another approximation based on these as well, along with an initial drag buildup. While both of these were assumptions, they were educated guesses which the team felt closely approximated the real world solution.

4.5 Comparisons of Assumptions Used With Those in the Conceptual Phase

While the weight estimation, along with other important performance characteristics, were narrowed down to a more estimate in the Preliminary Design Section from the Conceptual design, there were still assumptions that were similar to each section. The main assumption was the standard drag building giving Cd_o as 0.05 and also the assumption that e was 0.65. The team also narrowed their approach and began to look at wing loadings for better handling qualities in windy conditions, assuming that the weather in Kansas would not include calm winds. In addition, the team used the same battery and motor assumptions based on the initial wind tunnel testing in ECALC for determining the propeller efficiency, and simply left more precise testing until the final detailed design was completed.

5.0 Detailed Design

Upon completing the preliminary design for the aircraft, the group then began to look at the detailed characteristics of the aircraft. These included performance characteristics, stability and control issues, and looking into how the team would construct the aircraft.

5.1 Performance Data

In order to determine the theoretical performance of the aircraft, the team updated its design and made several basic aerodynamic performance assessments. The first, and most important of these was the re-evaluation of the predicted stall velocity using the equation shown below. During the conceptual and preliminary design, the stall speed was assumed in order to determine a necessary wing size, based on a specific C_L. During the detailed design, the wing size was adjusted to account for design changes, max C_L was adjusted to account for the fact that the flaps do not extend across the entire wing, and the stall velocity was found to be 36 ft/s, or 24.5 mph full and 30.5 ft/s empty, or 21 mph. As expected, these should not have changed much from our original predictions since S was based on max C_L.

$$V_{Stall} = \sqrt{\frac{2 \cdot W}{\rho \cdot S \cdot C_{L\max}}}$$

Next, the takeoff distance had to be calculated to ensure that the aircraft would be able to take-off within the required 200 ft. Based on an estimated $C_{L_{max}}$ of 2.0 during take-off, it was found that the ground roll of the aircraft was 32 ft empty and 63.5 ft full, using the ground roll equation for Sg.

$$Sg = \frac{1.44 \cdot \left(\frac{W}{S}\right)}{g \cdot \rho \cdot C_{L_{max}} \cdot \left(\frac{T}{W}\right)}$$

The estimated $C_{L_{max}}$ of 2.0 was found by the using MFOIL. This really showed how small changes in weight could make a significant difference in performance. Should the aircraft's actual weight be 25.2 pounds, the ground roll increased by 5 feet. If $C_{L_{max}}$ was only 1.8, the aircraft's roll distance increased to 71 ft. Given this margin for error, it was apparent that take off distance was not an active constraint within the thrust to weight and wing loading ranges.

Landing velocity was another important performance characteristic because it determined the amount of strength needed in the gear, and thus the wing, in order to handle the impact on landing. It also determined how much runway would be needed to stop the aircraft and the pilot's ability to react to wind gusts on approach. A faster landing caused longer ground rolls, and harder impacts, thus creating increased wing weight to handle higher impacts. Landing velocity was a simple function of stall velocity, as shown below, so the real goal was to minimize the stall velocity, by either over sizing the wings, lowering the aircraft weight, or increasing the max C_L of the wings.

$$V_{Landing} = V_{Stall} \cdot 1.2$$

Cruise velocity was easily calculated using the same equation as stall velocity, except that the cruise C_L was used instead of max C_L. For the aircraft design, the cruise velocity was chosen based on optimizing battery usage. Cruise velocity was chosen to be 55 mph based on the DBF computer program. This fixed the lift coefficient and the team then chose an airfoil with a high lift to drag ratio at that C_L based on the following equation.

$$V = \sqrt{\frac{2 \cdot W \cdot n}{\rho \cdot S \cdot C_L}}$$

Because there were a large number of turns in the course, it was important to minimize the size of the turn. The turn radius equation helped minimize the total length of the course by decreasing the

radius of the turns. Velocity and turn Gs were the important factors in decreasing the size of a turn. Minimum velocity and maximum turn Gs were optimal. However, doing this increased the C_L required of the aircraft in the turn. A minimum turn velocity, or corner speed, must be determined using the above velocity, including the number of Gs expected in the turn and max C_L. A minimum turn radius could then be determined using those numbers. Using a standard 2 G, 60-degree turn, the minimum turn velocity was found to be 45 ft/s, or 31 mph, empty and 54 ft/s, or 35 mph, full, giving turn radii of 36.5 ft and 52 ft, respectively without the use of flaps. This was based on

$$R = \frac{V^2}{g \cdot \sqrt{n^2 - 1}}.$$

The use of flaps in the turn was still being debated because although they increased the max C_L, they also add much more drag, which became a problem when looking at the amount of amp seconds available. The drag would pull more amps, and the team needed to make sure that they would have enough power to land.

In summary, the basic design parameters are given below:

Table 7

Design Parameters

W empty	17 lbs
W full	24.2 lbs
Cl max	2
S	7 ft^2
V cruise	73.33 ft/s

Using this data, and the equations from above, the following information was found:

Table 8

Design Performance

Parameter	*We*	*Wf*
Ground Roll	32 ft	63.5 ft
Stall Velocity	30.5 ft/s	36 ft/s
Landing Velocity	36.6 ft/s	44.4 ft/s
Load Factor	5.77	3.94
Turning Radius	36.5 ft	52 ft

These performance parameters were well within the desired aircraft design limits. Takeoff distance was well within the specified 200 feet, even with the full load. The stall velocity was low enough that the landing velocity was slow enough to allow comfortable landings for the pilot and a minimal ground roll

upon touchdown. In addition, the turning radius appeared to be suitable and comparable to the numbers that were run originally in the DBF program to give the desired course completion times and scores.

5.1.1. Stability and Control

The next task was to figure out the horizontal and vertical tail areas. The horizontal tail volume, Vh, was calculated using the distance from the aerodynamic center of the tail to the aircraft center of the wing body, the mean aerodynamic chord of the wing, the area of the wing, and the Vh value. Based on data collected from historical figures, Vh was targeted at 0.5 based on a wing area of 1000in^2. The mean aerodynamic chord was calculated using the equation

$$\bar{c} = \frac{2}{3} C_r \frac{1+\lambda+\lambda^2}{1+\lambda} = \frac{2}{3} \cdot 11.2 \cdot \frac{\left(1+.45+.45^2\right)}{1.45}$$

which gave 11.2 inches. Putting all of these numbers into the equation

$$V_H = \frac{l_t \cdot S_t}{\bar{c} \cdot S} = .5 = \frac{45 \cdot S_t}{11.2 \cdot 1008},$$

the surface area was determined to be 124.4in^2. From that point it was decided that an aspect ratio of 3 be used, and from the equation,

$$AR = \frac{b^2}{S} = 3 = \frac{b^2}{144}$$

An aspect ratio of 3 was used for the same reason that a lower aspect ratio was chosen for the wing—stiffness. Because the horizontal tail was so thin, a high aspect ratio would make the tail lose its rigidity, thus making the aircraft harder to control. The team determined that a tail span of 19.3 inches would be used.

In order to find the vertical tail area the equation

$$V_{VT} = \frac{l_{vt} \cdot S_{vt}}{b \cdot S} = .05 = \frac{45 \cdot S_{vt}}{90 \cdot 1008}$$

was used. The vertical tail volume was chosen to be 0.05 based on historical trends in light aircraft. Using this vertical tail volume ratio, and the distance from the center of gravity to the aerodynamic center of the vertical tail being 45 inches, the wing span at 90 inches, and the wing surface area being 1000in^2 gave a tail area of 100in^2. For ease of construction, it was decided that the chord at the root of the vertical tail would be about the same as the chord of the horizontal tail, which was 6 inches. By having the same chord length at the root, the vertical tail and the horizontal tail could be attached at the root. Using the taper equation, the tip chord was 2.7 inches.

5.1.2 Sizing the Tail

The most important aspect of the tail was its size. The team needed to have enough horizontal tail to counter the effects of nose down pitching moment, enough vertical tail to counter any yaw due to

engine failure, and to make the airplane yaw and bank when there was an aileron failure. Also, the size of the horizontal tail would determine where on the airplane the stick fixed neutral point would be. Once the stick fixed neutral point was determined, the center of gravity was found so that the desired maneuver requirements could be achieved.

The downward pitching moment was caused by two different yet very similar factors. There will always be a downward pitching moment that was caused by the camber in the wing, and must be countered by the horizontal tail. However, an extra moment due to flap deflection would be added and must also be countered by the tail. These are similar in that adding extra camber into the wing causes them both, except that the pilot has control over the flap deflection while in flight. Aircraft maneuverability was also affected by the horizontal tail. The fixed stick neutral point was set by the size of the tail, but the distance from the fixed stick neutral point to the center of gravity could be controlled by the placement of the softballs or batteries.

Shifting from pitch to yaw, the vertical tail must be considered. In a single engine airplane the size of the rudder was a function of the wingspan as well as the area of the wing. This was also a factor when considering a twin-engine aircraft, except that there was additional yawing moment that must be countered when dealing with a failed engine. This was caused by increased drag due to the failed engine, as well as p-factor and asymmetric thrust from the operating engine.

For sizing both the horizontal and vertical tails, the team looked at historical data. The team first started with the equation for the horizontal tail area:

$$V_h = \frac{l_t \cdot S_t}{\overline{c} \cdot S}$$

V_H for light, general aviation aircraft is typically 0.6, and was the value used. From equation 5.10, l_t was calculated to as 43.5 inches. For this design, the distance from the center of gravity to the aerodynamic center of the tail, $\overline{c}$, was the mean aerodynamic chord of the airfoil and was 11.2 inches. S was the surface area of the wing, and was 1008 square inches. Substituting these numbers into the following equation, the result was

$$.6 = \frac{43.5in}{11.2in} \cdot \frac{S_t}{1008} \therefore S_t = 156in^2 .$$

Because of the interference cause by the fuselage boundary layer, and additional 10% was added to these numbers, giving the final value of the surface area of the horizontal tail as 170 square inches.

For flight test purposes, however, the team wanted to make sure that they had more than enough tail area to ensure the aircraft could be controlled. When the original tail size of 156 square inches was laid down it appeared that this size, when compared to historical sizes from other aircraft, was not large enough. As a result, for initial flight test purposes, a tail volume of .85 was used. This value was derived from looking at typical historical values for tail volumes of twin engine and cargo aircraft. The aircraft fit into both of these categories, so the team felt this measure of safety was justified, as it would be rather simple to cut off the additional tail area if desired later. However, it would be much more difficult to

rebuild the aircraft if the tail was too small and the aircraft crashed during initial flight-testing. So, the team gave itself a large margin for error and could trim the tail area down if necessary after initial flight tests.

Sizing the vertical tail was almost exactly the same as sizing the horizontal tail, except that a slightly different equation was used. This equation used is

$$V_{v.t.} = \frac{l_t \cdot S_{v.t.}}{b \cdot S}.$$

A value for $V_{v.t.}$ of 0.05 was arrived by considering general aviation aircraft that also have two engines. l_t was the distance of the aerodynamic center of the vertical tail to the center of gravity, and was 47in. b is the wing span of the aircraft and was 90in. S is the wing area of the aircraft and was 1008in. From these numbers, the vertical tail area was:

$$V_{v.t.} = .05 = \frac{47in \cdot S_{v.t.}}{90in \cdot 1008in^2} \therefore S_{v.t.} = 115.8in^2$$

The only other factor that impacted the sizing of the tail was how well the vertical tail was able to counter the yawing moment caused by an engine failure of the critical engine. What was done was to calculate how much of a force the tail could produce at certain airspeeds, and calculate the moments associated with that force. If the moments created exceeded the moments produced by the failed engine below stall speed, then the rudder would be effective at all airspeeds that the plane could fly. In this case it was assumed the thrust would be constant (static thrust) and that all of the thrust acted where p-factor acts. It was determined that in the worst case, the vertical tail would be able to counter the dead engine right at about the stall speed of the airplane.

5.1.3 Static Stability & Control

The team felt that it would be very beneficial to solve for the basic equations of static stability in order to understand better how the aircraft would perform during preliminary flight-testing.

The first step was to find the tail volume ratio, which was based on the following equation:

$$V_h' = \frac{l_t' \cdot S_t}{\bar{c} \cdot S} = .655$$

Next, the lift curve slope of the wing body combination must be determined using equation 5.14.

$$a_{w.b.} = \frac{a_o}{\left(1 + \left(\frac{57.3 \cdot a_o}{\pi \cdot e \cdot AR}\right)\right)} = .08069\, per \deg$$

Now that this was known, the lift curve slope for the airplane, a, could be determine using,

$$a = a_{w.b} \cdot \left[1 + \left(\frac{a_t \cdot S_t}{a_{w.b} \cdot S}\right) \cdot \left(1 - \frac{d\varepsilon}{d\alpha}\right)\right] = .0902 \, per \deg$$

Now, using the same equation used to find the lift curve slope of the wing body, the lift curve slope of the tail could be found, giving a as .081 per degree. Thus, knowing this, the stick fixed neutral point, h_n, could be determined by

$$h_n = h_{n_{w.b}} + V_h' \cdot \frac{a_t}{a} \cdot \left(1 - \frac{d\varepsilon}{d\alpha}\right) = .66\,.$$

The team felt it was important to determine this position, as it allowed the team to know where on the aircraft the center of gravity will fall so that it makes the aircraft most stable, yet at the same time maneuverable enough to do turns and climb out after takeoff. This value is called static margin, and should be negative in order for the aircraft to be stable. In this group's case, the static margin was at

$$.0895 \cdot (.25 - .655) = -.0362\,,$$

which was an acceptable value.

5.1.4 Nose Down Pitching Moment

The next stability area for the aircraft that needed to be addressed was the range of pitching moments created by the aircraft that the horizontal tail would be able to counter. Calculations for nose down pitching moments were done for the aircraft in two configurations—flaps up and flaps down.

To solve for the flaps up condition, the stability values were first calculated so that a matrix of equations could be set up. C_{m_o}, or the nose down pitching moment at zero angle of attack, was calculated first. The incidence angle in this case was the zero lift angle of attack, and $C_{m_{ac}}$ was the pitching moment about the quarter chord of the wing; there was also no downwash value for this problem. It was determined that the plane's value for $C_{m_{ac}}$ was .45 in this configuration.

The calculations for the airplane with flaps were almost the same, except for a few subtle differences. With flaps, downwash must be considered, and the total lift coefficient has changed. However, these differences were so minimal that the aircraft was considered to be just as stable with the flaps up as compared to when the flaps were dropped.

5.1.5 Elevator Deflection Per G

5.1.5 Elevator Deflection Per G

The elevator angle for stall was not the only elevator deflection that needed to be known. The elevator deflection per G was also necessary because it would show how well the plane could maintain a turn at a given G loading. It also revealed if the aircraft had enough elevator deflection to perform a sudden unexpected maneuver. To find the elevator deflection per G, the following equation was used

$$\frac{\Delta\delta e}{n-1}=\frac{-aC_L}{C_{L\alpha}C_{m_{\delta e}}-C_{L_{\delta e}}C_{m_\alpha}}\left[\left(h-h_n\right)+\frac{1}{2\mu}*\frac{l_{th}}{cbar}m_q\right]$$

with the known values

$$C_L=0.25;$$

$$C_{m\alpha}=-0.0362/°;$$

a=0.0895/deg;

$$C_{m_{\delta e}}=-0.0486/°;$$

$$C_{L_{\delta e}}=0.0137/°$$

$$m_q=-2*a_t*V_H=-2*0.081/°*0.6=-2*\left(0.081/°*\frac{180}{\pi}\right)rad^{-1}=-5.57rad^{-1}$$

$$\mu=\frac{2*m}{\rho Scbar}=\frac{2*0.807slugs}{2.3769E-3\frac{slugs}{ft^3}*7ft^2*0.93ft}=104$$

and substituting to solve gave

$$\frac{\Delta\delta e}{n-1}=-3° \qquad \text{at 4 G's => } \Delta\delta e=3*-3°=9°$$

The answer that came from this equation was that it takes 9 degrees of deflection in order for the plane to pull four g's. This was not all that much, but this was only for a steady rate climb, and it could be assumed that a bit more elevator would be required for a steady bank. The answer did show, however, that the plane had enough elevator deflection to perform drastic, high-g maneuvers at higher speeds.

5.1.6 Cn Beta – Weathercock Stability

The next stability derivative to be addressed was the aircraft's directional stability, also known as weathercock stability. This stability was important when considering the pilot's ability to fly the aircraft, with three different parts of the aircraft contributing to the stability. Each part was analyzed independently and then combined into the total stability derivative Cn beta. The first was from the vertical tail, the second was from the wing, and the last was a result of the fuselage. In order to calculate these values, a combination of DATCOM and simple equations was used. The first element to be analyzed was the vertical tail. Its contribution is given by

$$\left(C_{n_\beta}\right)_F = \frac{a_F S_F l_F}{Sb} = \frac{0.0806/{}^\circ * 120in^2 * 45in}{1008in^2 * 90in} = 0.0096\deg^{-1} = 0.550rad^{-1}.$$

The wing contribution was determined by

$$\frac{C_{n_\beta}}{C_L^{\ 2}} = 0.02 \Rightarrow \left(C_{n_\beta}\right)_w = 0.02*(1.1)^2 = 0.024rad^{-1}.$$

The final part analyzed was the fuselage, and the following equation was used

$$C_{n_\beta} = \frac{-1.3(fuse.volume)}{Sb} * \frac{h}{w}$$

with known values

h = 4.25in; w = 9in,
and V = $1500in^2$.

This gave

$$C_{n_\beta} = \frac{-1.3*1500in^3}{1008in^2*90in}\frac{4.25in}{9in} = -0.0102$$

The overall derivative was found by adding the contributions of the vertical tail, wing, and fuselage, resulting in

C_{n_β}=0.584/rad.

5.1.7 Cl Beta – Dihedral Effect

The size of this derivative indicated how much roll could be induced by a rudder deflection. All that was needed in this situation was to ensure the rudder was adequate for lateral stability and control. For this project, the only importance of this value was to ensure the pilot could bank the plane in case of aileron failure. Also, roll stability reduced pilot workload and was important in RC flight for moments when the pilot had to take his eyes off the plane for a few seconds. There were two sources that contributed to the size of C_l beta, and those were due to the wing and amount of dihedral. The multipliers were all taken from DATCOM, and were simply table look up values of wind tunnel test results. Using DATCOM, the contribution from the wing planform was given by

$$\frac{-(C_{l_\beta})_w}{C_L} = 0.06 \Rightarrow (C_{l_\beta})_w = -0.066\, rad^{-1}$$

with $C_L = 1.1$. The dihedral contribution comes from

$$C_{l_{\beta\Gamma}} = -0.014$$

$$(C_{l_\beta})_\Gamma = C_{l_{\beta\Gamma}} * \Gamma = -0.014 * 4° = -0.056 rad$$

For this solution, the dihedral was four degrees.

$$C_{l\beta} = (C_{l_\beta})_w + (C_{l_\beta})_\Gamma = -0.066 rad^{-1} + -0.056 rad^{-1} = -0.122 rad^{-1} = -0.00213 \deg^{-1}$$

5.1.8 Roll Rate Derivatives – $C_{l_{\delta_a}}$

With the lift curve slope, root chord, wingspan, surface area, and taper ratio of the wing known, the change in lift for aileron deflection $C_{l_{\delta_a}}$ was calculated using the equation

$$C_{l_{\delta_a}} = \frac{2C_{L\alpha w}\tau C_r}{Sb}\left[\frac{y^2}{2} + \left(\frac{\lambda - 1}{b/2}\right) * \frac{y^3}{3}\right]_{y_1}^{y_2},$$

where

$$y_1 = 2.3; y_2 = 3.75; C_{L_{\alpha w}} = 0.08068/° = 4.61 rad^{-1}$$

and $C_r = 1.42 ft; S = 7 ft^2; b = 7.5 ft$.

This then yields

$$C_{l_{\delta_a}} = 0.129 rad^{-1}$$

5.1.9 Roll Rate Derivatives

The roll damping coefficient, C_{l_p}, was solved for giving

$$C_{l_p} = \left(\frac{\beta C_{l_\beta}}{K}\right)_{C_L=0}\left(\frac{K}{\beta}\right)\frac{C_{l_\beta}}{(C_{l_\beta})_{\Gamma=0}} + (\Delta C_{l_\beta})_{drag} = -0.47$$

To find the steady state roll rate, the following equation was solved for varying aileron deflections

$$\frac{p_{ss}*b}{2*u_o}=-\frac{C_{l_{\delta a}}}{C_{l_\beta}}\Delta\delta a => p_{ss}=\frac{2u_o-C_{l_{\delta a}}\Delta\delta a}{bC_{l_p}}$$

The above equation was then solved for 20 degrees aileron deflection below

$$\frac{2*50ft/s*-0.129rad^{-1}0.349rad}{7.5ft*-0.47rad^{-1}}=1.28rad/s=73.2\deg/s$$

This same equation also was solved for with 5 degrees aileron deflection, giving

$$\frac{2*50ft/s*-0.129rad^{-1}*0.0873rad}{7.5ft*-0.47rad^{-1}}=0.319rad/s=18.3\deg/s\cdot$$

These values of the roll rate showed there was enough aileron control near stall speed for the airplane to fly under control even in cases of extreme aileron deflection.

5.1.10 Maximum Crosswind Component

The static stability derivatives were also used to find out the maximum crosswind limitations. The values of beta and aileron deflection came out to be –16 and –15 degrees respectively. Using a speed triangle, the maximum crosswind component was 11mph, which was acceptable for the team's airplane since a crosswind of that magnitude was not anitcipated

5.1.11 Elevator Deflection Required for Trimmed Flight – No Flaps

To find the required elevator deflection that would allow control to be maintained up to the stall point without flaps extended, the following equation was used.

$$C_{m_o}=C_{m_{ac_{wb}}}+a_tV'_H\left(\varepsilon_o+i_t\right)\left[1-\frac{a_t}{a}\frac{S_t}{S}\left(1-\frac{\partial\varepsilon}{\partial\alpha}\right)\right]$$

Substituting the following values into the above equation,

$$\alpha_{L=0}=-6°;\ C_{m_{ac}}=-0.121;\ i_t=6°;\ \varepsilon_o=0$$

and solving for C_{m_o}, the following result was obtained:

$$= -0.121 + 0.081/° * 0.655(0+6°)\left[1 - \frac{0.081/°}{0.0895/°}\frac{170in^2}{1008in^2}(1-0.3)\right] = 0.163$$

A matrix was set up to solve for a range of elevator deflections over a range of angles of attack and the subsequent lift coefficient. The matrix is shown below.

$$\begin{bmatrix} C_{L_\alpha},,,C_{L_{\delta e}} \\ C_{m_\alpha},,,C_{m_{\delta e}} \end{bmatrix}\begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} C_L \\ -C_{m_o} \end{bmatrix}$$

Most of the other quantities in the matrix were known, to include

$$C_{L\alpha} = 0.0895/°,\ C_{L_{\delta e}} = \frac{S_t}{S}a_t = \frac{170in^2}{1008in^2} * 0.081/°,$$

$$C_{m\alpha} = C_{L\alpha}(h_{ac} - h_n) = 0.0895/°(0.25 - 0.655) = -0.0362/°,$$

and $C_{m_{\delta e}} = -a_e * V_H = -0.081/° * 0.6 = -0.0486/°$.

Substituting them into the matrix then yielded

$$\begin{bmatrix} 0.0895,,,0.0137 \\ -0.0362,,,-0.0486 \end{bmatrix}\begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} C_L \\ -0.163 \end{bmatrix}$$

The following solution for the elevator deflection resulted, which was positive for a trailing edge down deflection.

$C_L =$	0.25	0.4	1.1
$\alpha =$	3	5	13
$\delta =$	2	0	-7

5.1.12 Elevator Deflection Required for Control – Flaps

Flaps were included in the design of the aircraft and so it was necessary to also solve for the required elevator deflection with the flaps deployed. Deflecting the flaps altered some of the previously known variables, giving

$$\alpha_{L=0} = -9°,\ \Delta C_L = 0.43,\ C_{m_{ac}} = -0.15,\ i_t = 6°,\ \varepsilon_o = \frac{\partial \varepsilon}{\partial \alpha}, \text{ and } \varepsilon_o = 0 + 0.3 * 6° = 1.8°.$$

Solving again yielded

$$C_{m_o} = -0.15 + 0.081/°(1.8+6)°*0.655\left[1 - \frac{0.081/°}{0.0895/°}\frac{170in^2}{1008in^2}(1-0.3)\right] = 0.363,$$

$$\begin{bmatrix} 0.0895,,,0.0137 \\ -0.0362,,,-0.0486 \end{bmatrix}\begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} C_L - 0.45 \\ -0.163 \end{bmatrix}, \text{ and}$$

$C_L =$	0.5	1.5	2	2.4
$\alpha =$	0	13	19	24
$\delta_e =$	3	-6	-11	-15

which showed that the maximum required elevator deflection was in the flaps down configuration, when 15 degrees of up elevator was needed to maintain control.

5.2. Estimated Mission Performance

After determining all of the major performance characteristics for the aircraft, the team was then able to determine the theoretical scoring capability of the aircraft. The final design parameters, including battery characteristics, were put into the DBF program and the following results were determined. These design parameters are given in Table 9.

Table 9

Final Design Parameters

Empty Weight	17 lbs	**Wing Area**	7.5 sq ft
Full Weight	25 lbs	**Cdo**	0.05
Turn G's	2	**Turn Velocity**	32 mph
Time to Load	60 sec	**Voltage per Cell**	1.2 V
Wind	0 mph	**AR**	8
Efficiency Factor	0.7	**# of Balls**	18
Cruise Velocity	57 mph	**Clmax**	1.8
Number of Cells	35	**Cell Rating**	2500 mAh

The score that the final design achieved was 3.815, which was higher than the score achieved by the conceptual designs. The time it took for the aircraft to complete the course, with 60 seconds allowed for loading and unloading balls, was 377 seconds. This was well within the time limits set forth by the competition and was even under the amount of time that the batteries would last by about 2 minutes, which meant that the team had a comfortable margin of safety for power.

5.3. Weight and Balance Sheet

The aircraft center of gravity was perhaps the most important aspect of stability that the team needed to calculate for two reasons. First, the team needed to ensure that the wing was attached to the fuselage with the quarter chord located right on the center of gravity, since it gives the pilot the most amount of control possible when flying the course. Second, the team wanted to make sure that the payload of softballs was centered around the center of gravity, so that when they were removed and put in, they would not alter the cg and make the plane impossible for the pilot to fly. To establish the center of gravity, the nose of the aircraft was selected as the datum. Then, all moments due to each of the parts of the airplane were summed up to establish where the zero moment location of the plane was. The softballs would then be loaded symmetrically, fore and aft of this position so as to not shift the center of gravity when they were loaded or unloaded. The following MATLAB code was used to compute the zero moment point on the aircraft.

Figure 9

MATLAB Code Used to Determine CG

```
clear
% All of these are either distances or weights. If there is a W in the
% word then that is the weight of that item. However, if there is an A in the
% item, then that is the moment arm from the nose of the aircraft.
fuse_W=3;              fuse_A=35;
wing_W=3;              wing_A=30;
tail_W=2;              tail_A=75;
motor_W=2.12;          motor_A=25;
battery_W=5;           battery_A=30;
smallbat_W=.26;        smallbat_A=27;
wattind_W=.24;         wattind_A=27;
speedcon_W=.12;        speedcon_A=27;
forsoftball_W=4;       forsoftball_A=12;
aftsoftball_W=3.2;     aftsoftball_A=42;
gear_W=.5;             gear_A=28;
servo_W=1;             servo_A=35;
prop_W=.3;             prop_A=27;

% This is simply the moment of each of the pieces of equipment.
Moment=fuse_W*fuse_A+wing_W*wing_A+tail_W*tail_A+motor_W*motor_A+battery_W*battery_A+small
bat_W*smallbat_A+wattind_W*wattind_A+speedcon_W*speedcon_A+forsoftball_W*forsoftball_A+aftsoft
ball_A*aftsoftball_W+gear_W*gear_A+servo_W*servo_A+prop_W*prop_A;

% Below is the total weight of the aircraft.
Weight=fuse_W+wing_W+tail_W+motor_W+battery_W+smallbat_W+wattind_W+speedcon_W+forsoftball
_W+aftsoftball_W+gear_W+servo_W+prop_W;

% This is the distance back in inches of the center of gravity.
dis_cog=Moment/Weight

%dis_cog =
%  32.5077
```

Whenever a component weight estimate changed, this code determined the new center of gravity.

5.4. Component Selection and System Architecture

While there are a great number of aircraft components involved in the construction and building of a fully functional remote control aircraft, the major parts of the airplane should be discussed. Each component and it's structure and design are discussed below. Figures accompany several of the aircraft parts to give a better idea of how the items were put together. Figure 10 gives a picture of the aircraft in its final leg of construction to show what the combined product looked like.

Figure 10

Picture of Nearly Complete Aircraft

5.4.1 Wing

The wing structure was polystyrene core covered with fiberglass and a carbon-Kevlar weave, and reinforced with two carbon-fiber shear webs with spar caps. The spars ran the full span of the wing with a joint at 15 inches from the root, where the polyhedral begins. The outer wing section was joined to the inner wing section by secondary bonding unidirectional carbon along the quarter chord span and strips of fiberglass along the chord. The two wing halves were then joined in a similar manner. Part of the leading edge of the wing was dug out at the polyhedral so that the motor mounts could be bonded to the quarter chord shear web and spar. Tubes were run along the inside of the wing so that wires for the servo control

systems could be run internally to allow for less drag. Figure 11 gives a good representation of the wing with the motor mounts installed.

Figure 11

Picture of Finished Wing

5.4.2. Aerodynamic Control Surface Hinges

The traditional way to hinge the ailerons and flaps was to add more material for an attachment assembly. However, to cut weight and complexity, a strip of Kevlar was laid down along the hinge line so that after the wing was laid up, a strip of the wing along the hinge line would be cut and the control surfaces would swing freely in this hinge. The Kevlar was used because of its flexibility and its elasticity. The Kevlar hinge lines are visible in Figure 11 near the trailing edge. They appear as orange strips along

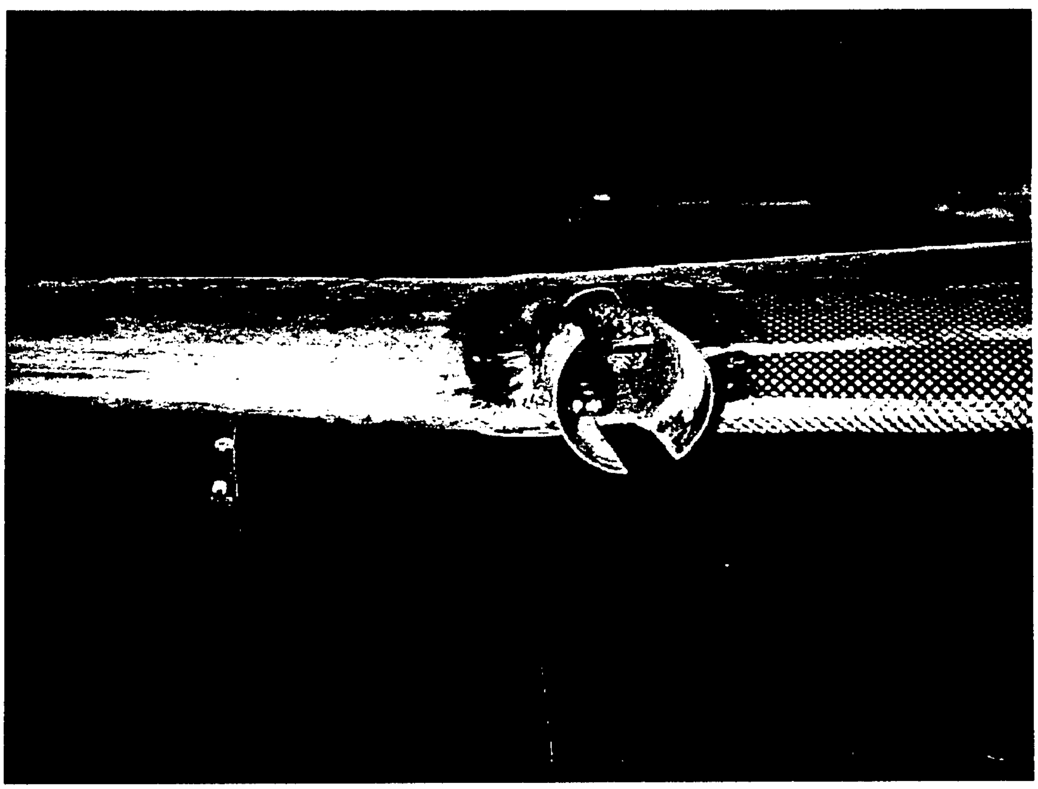
the outer wing sections. Because the inner wing section was covered in a carbon-Kevlar weave, it was unnecessary to lay down additional Kevlar for a hinge line, since this weave had all the required flexibility.

5.4.3. Motor Mounts

The two motors were mounted on a carbon-fiber tube mounted into the wing. The tube was custom made from a mold specific to the Astro 40 size engines. Hose clamps were the used to hold the Astro 40's onto the mounts. While this left most of the engine in the free stream, which produces a bit of drag, the team felt that cooling the engine was a more important issue in this case. The smaller engines with the large amount of wattage flowing through them tended to heat up during initial testing, and the team did not want to risk having an engine seize up. Figure 12 shows a close up frontal view of the engine mounts and the protruding connection wires, which were run through the inside of the wing in a tube.

Figure 12

Picture of Engine Mount



5.4.4. Gear

After considering the advantages and disadvantages of both tricycle and tail dragger landing gear configurations, the team decided to go with a tail dragging configuration because of the complexity of a

nose gear design along with drag considerations. The landing gear was originally going to be a simple coiled wire design located at the motor mount but that turned out to be structurally risky, as the gear on a hard landing could punch through the wing. In addition, it was thought this gear might be too flimsy for an aircraft of this weight. Instead, the team came up with a structurally solid, yet aerodynamic, design. The result was a symmetrically shaped airfoil that attached to the wing along most of the chord to spread the load. Additionally, the wheel was almost totally encased within the cowl to reduce drag. The tail wheel was attached to the rudder in a small cowling of its own after the rudder hinge had been substantially strengthened to take the increased load. Figure 13 shows one of the main landing gear under the wing.

Figure 13

Picture of Landing Gear Strut

5.4.5. Servos

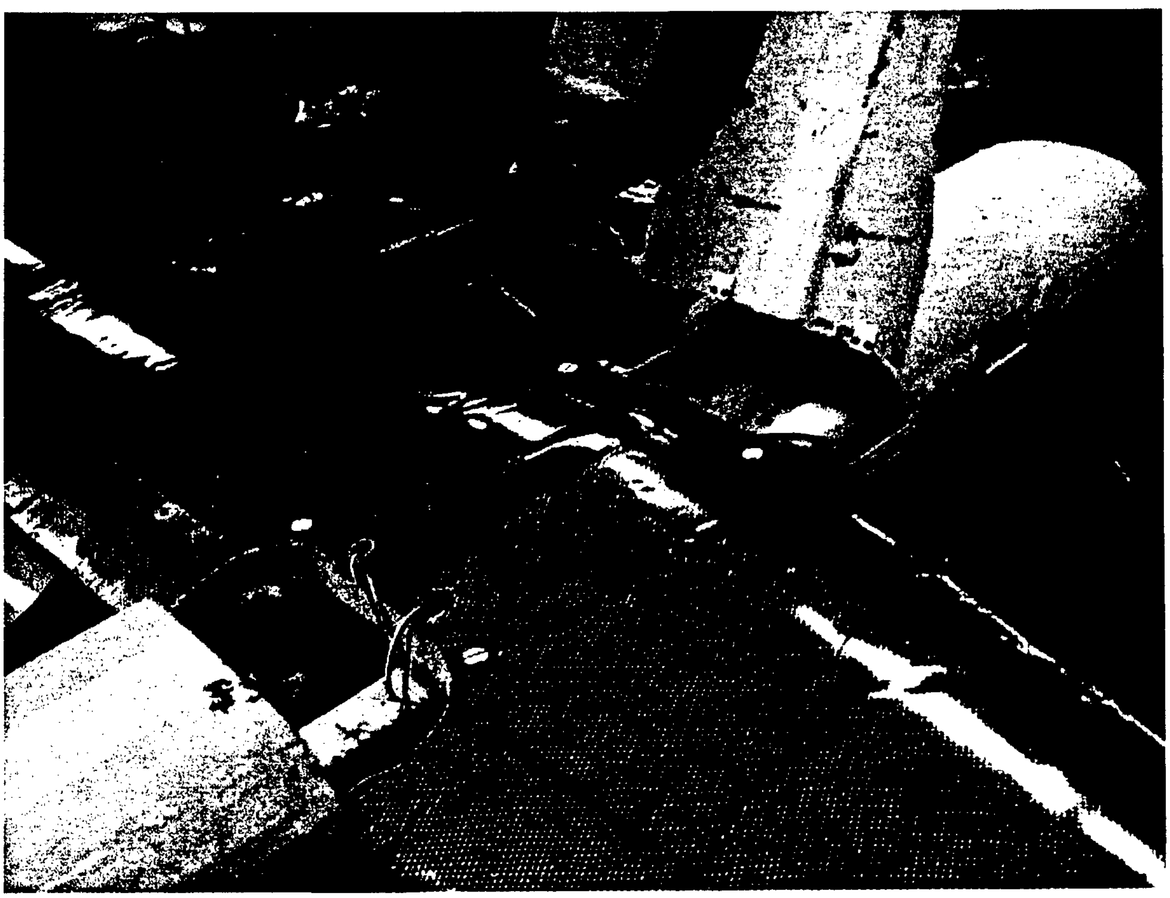
Three different types of servos provided the control systems for the aircraft. The team decided to go with micro servos for the ailerons and elevator because they provided the as much torque as needed at a minimal weight. Because of the large size of the rudder for the twin-engine configuration, a special steel geared ball bearing servo was used for the rudder. Additionally, larger servos were used for the flap surfaces so that the team did not have to worry about the gears stripping when the flaps were lowered at higher speeds.

5.4.6. Wing Attachment

The structure that changed several times during detailed design was the wing joining method. The original idea for joining the wing and fuselage was to have a continuous spar and shear web wing section that would be attached to the fuselage. It was then decided that the fuselage and inner wing panels out to the motors should be built as a single piece with a continuous shear web. This presented a very difficult structure to build all at once and so another idea was developed. The wing spar (also acting as fuselage bulkheads) would be used to build a wing-central fuselage combination with the forward and aft fuselage sections bolting to the bulkheads. This idea was also quickly dropped because of structural problems associated with multi-sectioned fuselages. A wing plug-in system was then developed in which the shear webs would plug into slots built into the fuselage and held in place by aluminum bolts. This system was chosen in the end because of its strength, simplicity, and ease of manufacture. Figure 14 shows a top down picture of the wing attachment to the fuselage, where the heads of the four aluminum bolts can be seen.

Figure 14

Picture of Wing Attachment to Fuselage



5.4.7. Fuselage

The fuselage was to be based on a full-length keel of a glassed honeycomb material. The batteries were mounted between the two bulkheads built into fuselage at the wing joint and cooled by a large, ducted vent that opens just in front of the wing and exits just behind it. The radio gear was to be mounted just aft of the rear payload bay. The payload bay doors were cut from the upper half of the fuselage before the two halves are joined and hinged to allow for quick access to the payload.

Three fuselage halves (two bottom and one upper) were built, and one of the bottom halves had the air scoop built in. Stiffening torsion rings were added when the two fuselage halves were joined because the team felt the thin layer of fiberglass would not quite be stiff enough.

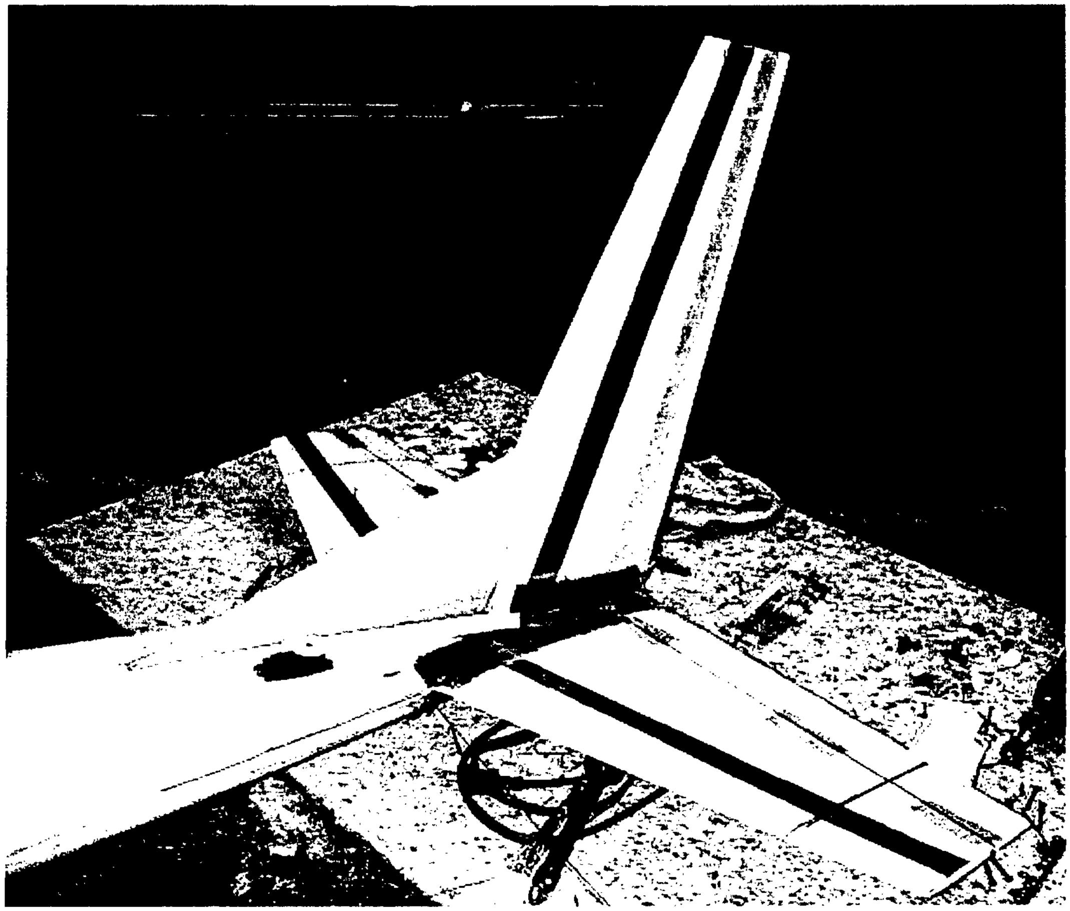
The air scoop was originally going to be built into the lower fuselage half but after a test build and some questioning of whether or not our CG location was accurate, the team decided to build the battery compartment later and attach it where necessary towards the end of construction.

5.4.8. Tail

The tail surfaces were glassed polystyrene epoxied to the tapering fuselage and firmly secured in order to prevent their separation from the aircraft under flight loading. The team looked at different tail volumes, and after initially selecting a tail volume of 0.5, it was determined that this was too small based on historical aircraft trends. After looking at what cargo aircraft and twin-engine aircraft had, which was around 8-9, the team instead went with 8.5, giving a Vt of 220 inches. Then, a 10% margin of error was given and a final area of 250 square inches was reached. The tail design can be seen in Figure 15 on the next page.

Figure 15

Constructed Tail Attached to Fuselage



5.5. Rated Aircraft Cost Worksheet

The rated aircraft cost was determined using the rated aircraft cost given in the DBF 2002 rules. The overall formula for the cost was given as,

$$\text{Rated Aircraft Cost} = (A*MEW + B*REP + C*MFHR) = (100*MEW + 1500*REP + 20*MFHR)$$

In this equation, MEW, or the Multiple Empty Weight, was estimated to be 17 pounds according to the weight build-up. Next, the rated engine power was calculated, where,

$$REP = (1 + 0.25*(N - 1))*X \text{ pounds} = (1 + 0.25*(2-1))*5\text{pounds} = 6.25$$

In this equation, N was the number of engines and X was the weight of the batteries in pounds. Putting in the numbers, it was determined that REP was equal to 6.25.

Next, the determined the total hours required for manufacture. This was the sum of 5 different work breakdown structures, or WBS's, and was given as,

$$MFHR = WBS1 + WBS2 + WBS3 + WBS4 + WBS5$$

The first work breakdown structure was based on the wing size and dimensions, and was equal to,

$$WBS1 = 8hr/ft(7.5ft) + 8hr/ft(1.34ft) + 3hr/cs(6cs) = 88.72$$

The measurements are the span and the chord of the wing and there are six control surfaces on the wing.

The second work breakdown structure was based on the fuselage length, and was determined by,

$$WBS2 = 10hr/ft(6.75ft) = 67.50$$

The third work breakdown structure was found using the numbers of vertical and horizontal tail surfaces, and was,

$$WBS3 = 5hr/vs(2) + 10hr/cvs(1) + 10hr/hs(2) = 40$$

The fourth WBS addressed the contribution of the flight systems, which, for the remote controlled aircraft, are servos and electronic speed controls. This was equal to,

$$WBS4 = 5hr/servo(7) = 35$$

Finally, the fifth work breakdown structure was based on the propulsion systems, to include the engines and the propellers, and was equal to

$$WBS5 = 5hr/engine(2) + 5hr/prop(2) = 20$$

When all this was summed to total manufacturing hours comes out to 251.22 hours to build the aircraft according to this formula. Putting this and the other parts into the rated aircraft cost equation, it was determined that,

$$\text{Rated Aircraft Cost} = (100*17 + 1500*6.25 + 20*251.22) = \$16,099.$$

What was interesting to note was that the weight of the batteries determined 2/3 of the total Rated Aircraft Cost; the five pounds of batteries cost $9375. What this shows was that most of the aircraft cost was already accounted for before design of the aircraft even begins, and that it would be rather difficult to design an aircraft which flew the competition well but was much lower priced than any other aircraft.

5.6 Assumptions Made and Justifications

The assumptions made in the detailed design of Team Tango's aircraft were much fewer than those made in the previous conceptual and preliminary design sections. The team had gotten to the point in the design where facts needed to replace most of the assumptions made in order for the design to be accurate with little margin.

However, since the computer programs were still used to optimize the detailed design, the team still had to make some assumptions to meet all the variable input requirements. For example, with the DBF program, variables such as the weight and aspect ratio were now known because the plane had been mostly built. However, other variables, such as the Oswald efficiency factor and the parasite drag, still had to be assumed as 0.7 and 0.05, respectively.

In addition, the detailed design section included a good deal of stability and control calculations, in which a lot of historical data was used to approximate some variables. Even though many of the variables used in this section were picked from this historical data, the team had to assume that a specific variable would work well for the aircraft. For example, while tail volume was based on those of cargo and twin engine aircraft, it did not necessarily mean that this tail volume would work for a scale aircraft operating at low Reynolds number. However, because of the limited manpower the team had, a lot of testing could not be done, so in some areas of the design making these educated assumptions was required.

5.7 Drawing Package

All drawings of Team Tango's aircraft were drafted by hand to scale. The final drawings for manufacturing were made at full scale. Figures 16 and 17 give the side, front, and top down views of the aircraft.

Figure 16

Side and Front View Final Design

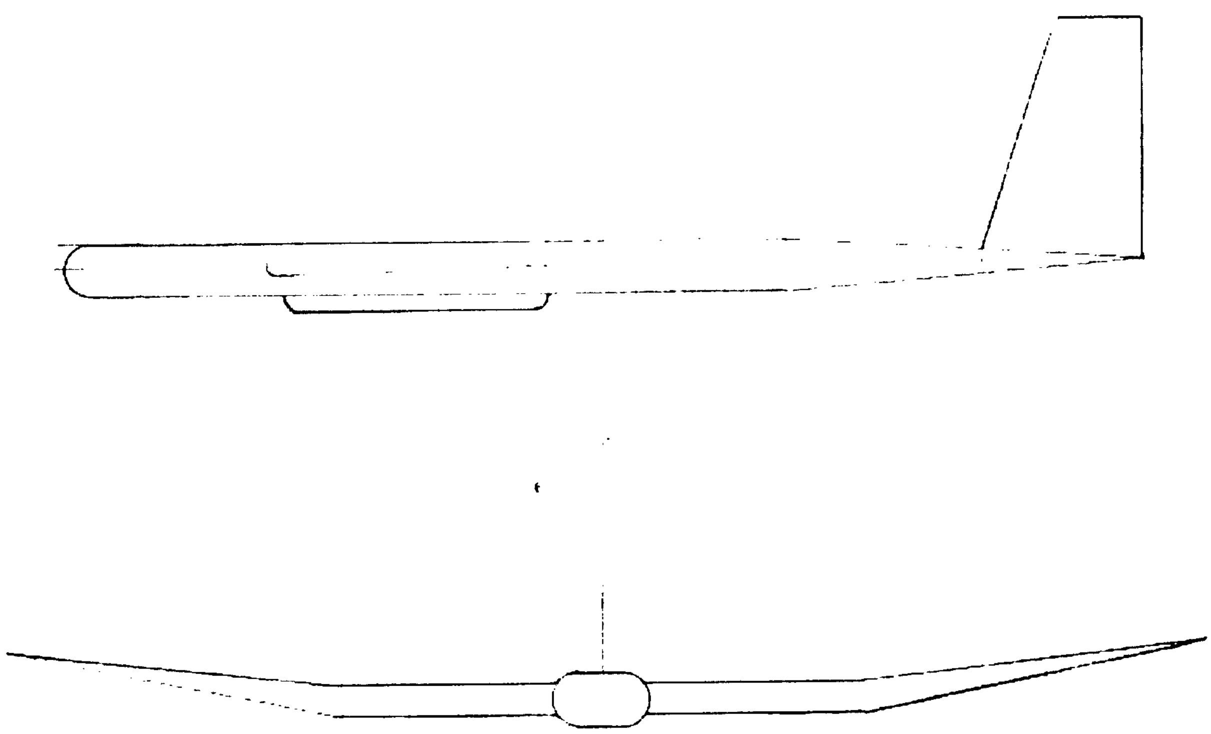
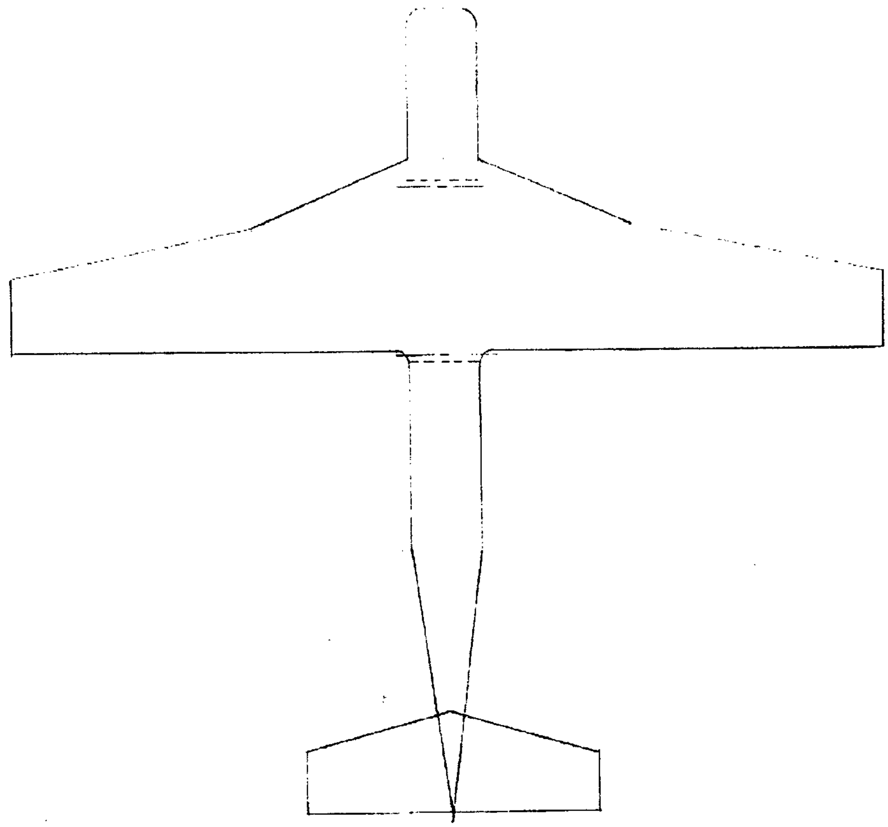


Figure 17

Top Down View of Final Design



6.0 Manufacturing Process

After finalizing all of the detailed design parameters for the aircraft, the methods for constructing the aircraft was investigated.

6.1 Processes for Component Manufacture

The design team decided very early in the process to build the aircraft entirely of composites. The fuselage would be formed on a mold made of foam and the lifting surfaces built from a combination of foam and composite material. Composites were chosen based on the simplicity allowed in manufacturing complicated geometries and the high strength to weight ratio inherent in their structures, especially when combined with lightweight foam cores.

The process for building the aircraft was fairly simple and straightforward. The wing was built in four sections. The inboard sections included the motors, landing gear, fuselage attachments, and were to

encounter the greatest loads. A lightweight foam core for each section was made from polystyrene and cut at the quarter-chord in order to build in a shear web. The shear web was made from carbon fiber and was laid up in the initial wing lay-up and attached directly to the foam in order to increase the strength of the bond between the skin and the shear web. This configuration was decided on after conducting experiments with different foam types and shear web arrangements. The results of the experiments gave a two-webbed, glassed polystyrene foam section a load-to-weight ratio of 1517 lb/lb. The bending moment at this configuration was 110 ft-lbs, enough to withstand a 4-G flight load for the design weight of the aircraft. After the shear webs were glassed in place, each of the four wing sections was glassed with all hinge elements, servo supports, and wire routing in place.

The foam for the wings was cut first, using the hot-wire method to create a precise outline of the wing for future glassing. This process used a single electrically heated wire following the outline of two airfoil sections out of polystyrene foam. This allowed the aircraft wing to have very complex geometries including tapered airfoil thickness, tapered wings, and both geometric and aerodynamic twist without any complicated tools. After creating all four airfoil sections, the first item to go on the inner wing sections was a carbon fiber spar cap at the quarter-chord along the upper surface. At the same time, a small strip of Kevlar fiber was glassed to the upper wing aileron and flap surfaces along the intended cut line to add durability and remove the necessity of having actual plastic hinges. Three extra layers of carbon were also placed on the upper and lower surfaces aft of the spar to add strength to the servo attachment area. The entire wing section was then covered with a layer of a Carbon-Kevlar mix fiber. The outer wing panels were built in the same manner, but were instead covered with fiberglass rather than the Carbon-Kevlar mix fiber.

The tail surfaces were built in a similar manner, using a small variation in the hot-wiring method to create smaller airfoil sections. Both the horizontal and vertical tails were made completely out of fiberglass, as they should not be taking as high a load as the wing.

The fuselage was built as two pieces, an upper and lower half. Each piece was built on the same mold to decrease manufacturing time, and then a keel was added to each half to add strength along the length of the fuselage. After the two fuselage halves were joined, bulkheads were placed into the tail section to increase the torsional rigidity of the tail section while support the weights and loads of the tail surfaces. A single section for the inner wing panels and softball ingress/egress was then cut around the center of gravity on the upper half of the fuselage. Two inserts for bolting the wings to the fuselage were then added under the intended placement of the wings in the fuselage.

Dihedral was added to the two outer wing section and all four sections were then attached using carbon fiber strips. The wing was then attached to the fuselage using four lightweight steel bolts. This allowed for easy removal of the wings from the fuselage, should any section need replacement during the test flights or the competition. After attaching the wing, the tail surfaces were attached directly to the fuselage using strips of carbon.

Landing gear were manufactured using Carbon fiber with a lightweight moldable foam core. These were manufactured rather than purchased because it allowed a more streamlined, efficient, and complex design. They were then attached directly to the inner wing sections using Carbon-Kevlar fiber to aide in their flexibility under loads encountered during landing.

6.2 Manufacturing Processes Investigated

As this was only the Naval Academy's second year entering the Design-Build-Fly competition, there was little experience with construction techniques for the aircraft. Therefore, for each part of the aircraft that required a different manufacturing technique, the team relied heavily on the experience of our team members with large amounts of R/C experience, such as 1/C Sorensen, and the manufacturing knowledge of our shop assistant, Bill Beaver. Because we had a great deal of experience between these people, and as the team had limited manpower with which to explore alternate manufacturing techniques, the team did not explore a wide range of alternatives. There was preliminary material testing completed early on during the conceptual design process that gave the team some figures of merit for material selection.

Analysis of the structures would not be very accurate mathematically due to the varying quality of the composite material but physical testing would provide a general idea of the required amount of material required to meet the structural loading. This data is given in the Table 9 on the next page.

Table 9

Material Selection Figures of Merit

Core Material	Top Composite	Bottom Composite	Weight (g)	Failure Load (lbs)	Load/Weight	Rank
Glass Alone	N/A	N/A	18.4	N/A	N/A	N/A
Polystyrene	4oz & 9oz Glass	4oz & 9oz Glass	26.3	30	517	12
Nomex	4oz & 9oz Glass	4oz & 9oz Glass	34.4	55	725	10
Green Foam	4oz & 9oz Glass	4oz & 9oz Glass	36.8	75	924	3
Core Cell	4oz & 9oz Glass	4oz & 9oz Glass	37.8	80	960	2
Westcore	4oz & 9oz Glass	4oz & 9oz Glass	40	65	737	9
Airex	4oz & 9oz Glass	4oz & 9oz Glass	41.8	60	651	10
White Foam	4oz & 9oz Glass	4oz & 9oz Glass	42.4	30	321	15
Balsa Wood	4oz & 9oz Glass	4oz & 9oz Glass	45.9	82	810	8
Lastafoam	4oz & 9oz Glass	4oz & 9oz Glass	61.2	65	482	14
Renwood	4oz & 9oz Glass	4oz & 9oz Glass	133.3	150	510	13
Polystyrene	2 x 9oz Glass	1 x 9oz Glass	31	60	878	6
Pstyrn w/ Shear Webs	2 x 9oz Glass	2 x 9oz Glass	32.9	110	1517	1
Polystyrene	10oz Carbon	5.7oz Carbon	22.8	45	895	4
Polystyrene	5.7oz Carbon	5.7oz Carbon	18.9	35	840	7
Polystyrene w/ Double Thick Core	10oz Carbon	5.7oz Carbon	38.3	75	888	5
Polystyrene w/ Double Thick Core	4oz & 9 oz Glass	4oz & 9oz Glass	32.6	40	556	11

After considering this data for materials selection and discussing different manufacturing processes with our shop technician, several different manufacturing ideas were approached before deciding on the final process. Since the shop was building two similar planes simultaneously, there was a strong argument between building custom-made parts that would fit the aircraft better, while taking longer to build, or modifying the design slightly to use more common parts for both aircraft. For example, it was decided that the top and bottom sections of the fuselage were to be built from one universal mold for both planes, saving time, money, and materials. The manufacturing of the wing had several different possibilities. The first one was to build the entire half-span wing out of one piece of foam rather than two, but this was discarded due to the requirement of dihedral in the wing complicating the manufacturing of a single half-span wing. Another consideration in building the wing was to cut out sections of blue foam along the span of the wing and adding reinforcing carbon strips. This idea was discarded due to the perceived structural requirements of the landing gear, engines, and winglets. The last consideration for the manufacturing of the wing was how to mount it to the fuselage.

6.3 Analytical Methods

The required skill level for construction of this aircraft was not at such a level that a person with minimal amount of training could not accomplish the building process. While the shop technician supervised many

of the more difficult aspects of construction, the point was for the team members to be able to construct the aircraft on their own with minimal help. The hardest part of the construction was the construction of the molds and templates. Laying up the glass and cutting the wing cores were relatively simple tasks and the simple planform eased construction. After most of this was accomplished, the team members were rather comfortable working with the composite materials, and had no problems joining the tail and wing to the fuselage, along with wiring the control systems in the aircraft.

When looking at the construction of specific aircraft parts, experience was the driving factor when analyzing the manufacturing process and the skills required. Each team member was more familiar with a specific section of the aircraft, and therefore was first assigned to build that section. As each team member gained experience in more specific manufacturing techniques under the supervision of our shop technician, the member was given an aircraft section more in line with that member's manufacturing experience. For example, MIDN Swift was the structures specialist, so he was assigned the fuselage and motor mounts. As skill increased, smaller and more complicated parts were manufactured.

Cost was minor a driving factor in manufacturing the various parts of the aircraft. The availability of materials and required weight and strength were balanced in order to create a design that was both strong and cost effective. Parts that required large amounts of materials or little strength, such as the fuselage, were generally manufactured from relatively inexpensive fiberglass. Parts requiring high strength, such as the landing gear, were made out of carbon fiber or Carbon-Kevlar. This allowed for maximum cost to strength efficiency.

Even with this efficiency, however, construction costs were moderately high with the inclusion of carbon fiber and Kevlar into the construction. The carbon fiber spar caps cost $80 per pair, the Kevlar-carbon weave costs $20 per square yard, and the fiberglass costs $8 per square yard. The estimated cost for the entire aircraft, composite material only, was $250. Including the cost of the foam from which the molds were made and the cores for the lifting surfaces, the airframe cost increased to approximately $450. The motors cost $200 each and the battery pack for the motors cost another $350 for the 35 match cells. Radio equipment added another $350 to the construction cost. The total manufacturing cost for one aircraft came to $1,600.

Scheduling was based on the perceived amount of time required to manufacture a certain part. Larger, more complicated parts required more time. Certain parts, such as the motor mounts, could not be manufactured until after both wing sections were manufactured. A complete analysis of the manufacturing schedule was done on Microsoft Project beforehand in order to create required part delivery dates. The parts requiring more time had to be put first in the manufacturing process so that the process could move in a more efficient fashion and to ensure that enough time would be left over in the end to allow for test flights before the competition.

6.4 Manufacturing Milestones

After seeing how the 2000-2001 USNA DBF team did not allow themselves much time for flight testing before the actual competition, and how much it hurt them not to be able maximize their design, the team determined that the earlier aircraft could be built, the better the team could compete at the competition. In addition, the earlier the plane was built, the more time it would give 1/C Sorensen to practice flying it. It was felt that the level of comfort at which the pilot could fly the plane would be a large factor in the success of the design. As a Result, the team set forth a series of milestones and goals to be accomplished so that the aircraft would be completed in good time. The milestones chart, included below, shows all the projected completion dates.

Figure 18
Milestones Table

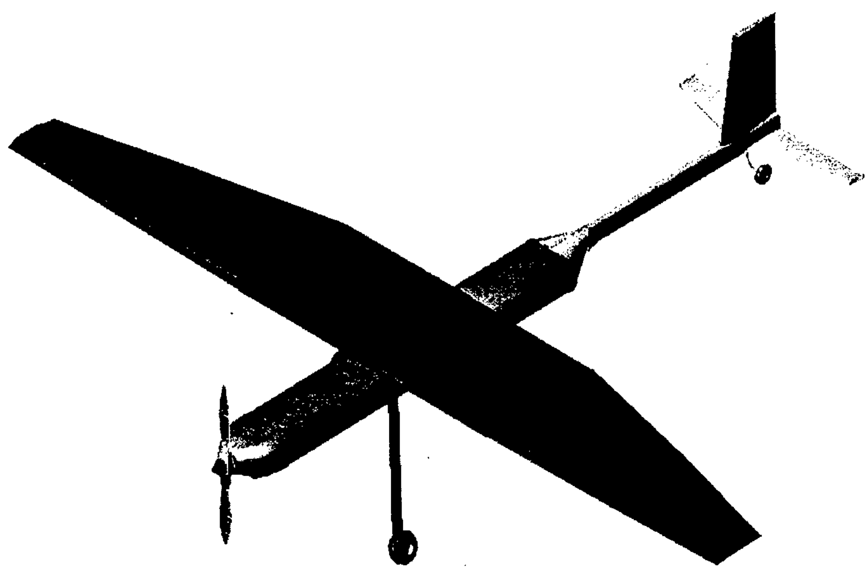
ID	Task Name	Duration	Start	Finish
1	**Wing**	29 days	Tue 1/15/02	Fri 2/22/02
2	**Complete Construction of Winglets**	7 days	Tue 1/15/02	Wed 1/23/02
3	Wind Tunnel Test Winglets	7 days	Tue 1/15/02	Wed 1/23/02
4	Redesign or Correct Winglets as Necessary	2 days	Tue 1/15/02	Wed 1/16/02
5	Bond Winglets to Wing	1 day	Tue 1/15/02	Tue 1/15/02
6	Cut out areas for servo mountings, flap hinges, etc on main wing	4 days	Fri 1/25/02	Wed 1/30/02
7	Bond Inner & Outer Wing Sections	4 days	Wed 1/30/02	Mon 2/4/02
8	Bond two wing halves	1 day	Mon 2/4/02	Mon 2/4/02
9	Cut out and prepare motor mount	4 days	Wed 2/6/02	Mon 2/11/02
10	Construct Wing Mounting System	4 days	Tue 2/5/02	Fri 2/8/02
11	**Cut out and shape LEX to proper dimensions**	5 days	Mon 2/11/02	Fri 2/15/02
12	Attach LEX, mold & fare	5 days	Mon 2/11/02	Fri 2/15/02
13	Install Servos, Wires, Control Systems	6 days	Fri 2/15/02	Fri 2/22/02
14	**Fuselage**	35 days	Tue 1/15/02	Mon 3/4/02
15	Lay up Lower Half of Fuselage	3 days	Wed 1/23/02	Fri 1/25/02
16	**Determine if Rings are necessary**	4 days	Mon 1/28/02	Thu 1/31/02
17	Build and Install Rings	4 days	Mon 1/28/02	Thu 1/31/02
18	Bond Top & Bottom Halves	4 days	Tue 1/15/02	Fri 1/18/02
19	Install Mounting Blocks for Wings	2 days	Mon 1/28/02	Tue 1/29/02
20	Cut & Mold Fuselage to Fit Wing	2 days	Fri 2/8/02	Mon 2/11/02
21	**Install Wing & Trim To Proper Fit**	5 days	Tue 2/12/02	Mon 2/18/02
22	Fillet Top & Bottom of Wing	5 days	Tue 2/12/02	Mon 2/18/02
23	**Landing Gear Construction**	4 days	Wed 2/27/02	Mon 3/4/02
24	Install Gear	4 days	Wed 2/27/02	Mon 3/4/02
25	**Empennage**	11 days	Wed 1/23/02	Wed 2/6/02
26	Make Templates for Airfoil Sections	1 day	Wed 1/23/02	Wed 1/23/02

27	Hotwire Sections out of Foam	2 days	Fri 1/25/02	Mon 1/28/02
28	Lay Up Sections	3 days	Wed 1/30/02	Fri 2/1/02
29	Attach to fuselage	3 days	Mon 2/4/02	Wed 2/6/02
30	**Flight Systems**	15 days	Fri 2/1/02	Thu 2/21/02
31	Mount Servos for Empennage in Fuselage	15 days	Fri 2/1/02	Thu 2/21/02
32	Install Radio & other Systems	7 days	Fri 2/1/02	Mon 2/11/02
33	**Propulsion**	27 days	Wed 1/30/02	Thu 3/7/02
34	Construct Motor Mount	21 days	Wed 1/30/02	Wed 2/27/02
35	Wind Tunnel Test Propellers	6 days	Fri 2/1/02	Fri 2/8/02
36	Install Motor Mount & Run Wires	3 days	Thu 2/28/02	Mon 3/4/02
37	Install Motors & Batteries	3 days	Tue 3/5/02	Thu 3/7/02
38	**Systems Engineering and Integration Testing**	5 days	Mon 3/18/02	Fri 3/22/02
39	Preliminary Flight Control Testing	3 days	Mon 3/18/02	Wed 3/20/02
40	Preliminary Ground Trim of Control Surfaces	1 day	Fri 3/22/02	Fri 3/22/02
41	Preliminary Power Run Up & Engine Test	1 day	Fri 3/22/02	Fri 3/22/02
42	**Flight Test**	16 days	Sat 3/23/02	Wed 4/10/02
43	Test I	1 day	Sat 3/23/02	Sat 3/23/02
44	Test II	1 day	Sat 3/30/02	Sat 3/30/02
45	Test III	1 day	Sat 4/6/02	Sat 4/6/02
46	Fly off	1 day	Wed 4/10/02	Wed 4/10/02
47	Competition	1 day	Fri 4/26/02	Fri 4/26/02

ISTANBUL TECHNICAL UNIVERSITY
AERONAUTICAL & AEROSPACE ENGINEERING
DEPARTMENTS

ATA-4

COMPETITION TEAM



AIAA STUDENT

DESIGN/BUILD/FLY COMPETITION 2001/2002

1. EXECUTIVE SUMMARY

In the beginning ATA-4 group consisted of only two people when decision was taken to participate in the Design/Build/Fly competition as ITU students in the past mid-summer. However, as the semester started at the end of September, the pioneers of the group caught the opportunity to come together with other students, and expanded the group to twenty people. Later on, in a short period of time many intensive meetings has been arranged so as to have an image of the aircraft in our minds that we would design.

According to the rules of the Design/Build/Fly competition, the aircraft that will be designed and manufactured has to stay airborne for different task profiles such as empty and loaded flight. The aircraft has to complete the full task within ten minutes. In addition the aircraft has to be thrusted by a propeller, powered electrically and controlled by radio.

As traditional, our organization consists of three main phases: conceptual design, preliminary design and detail design.

1.1. *Conceptual Design*

In order to get full productivity and the best execution, the design group was divided into five groups: wing, fuselage, landing gear, empennage and power groups. First duty of the groups was to think of alternative designs and to do researches for them. At the same time, the groups were to come together to discuss the adaptation of each alternative. These first designs have been evaluated as to some criteria like grading of contest, weight, drag, manufacturability, availability and firmness. The aim of this process was to decide on a short list of alternatives for detailed design.

1.1.1. *Alternatives of Design*

During the conception of design alternatives, groups had considered the following: wing planform, wing configuration, selection of profile, selection of landing gear, vertical positioning of the wing, fuselage arrangement, payload and power supply arrangement, empennage arrangement, arrangement of landing gear and other components.

Rectangular, elliptical, single tapezoid, double trapezoid , triple trapezoid, forward swept, and delta-wing shapes have been discussed for wing platforms. Bi-plane configuration has not been considered; instead we focused on the single wing. As for profile selection, the flight conditions and our laboratory sources have been contemplated. For vertical positioning of the wing all three possibilities (low-wing, mid-wing, high-wing) have been discussed. However, due to payload arrangement and dimension restrictions there were not many alternatives for fuselage design. Thus, the only alternative that could be thought was single fuselage. The different geometries and mounting locations of landing gear have been discussed later on. Power supply works were mostly inspired from the establishments of last year's ITU participants. Alternative empennage

arrangements were; classical tail, T-tail, inverted-V tail, twin tail, horizontal stabilizer with no rudder and canard have been discussed.

1.1.2. *Design Tools*

Fortran, Drag Estimator, AutoCAD 2000, Ansys, Microsoft Excel are a part of the softwares used for design. While Drag Estimator was being used effectively for evaluating different fuselage types, AutoCAD have imaged these types to provide a clear vision about the design. Microsoft Excel has been used especially for providing the grading results.

1.1.3. *Results of Conceptual Design*

Honestly, the path of long researches in order to design the best alternative could not be followed due to limited time. Instead, the results of short and rather empirical researches have been collected, evaluated and improved via the design tools.

In this stage, single trapezoid, double trapezoid and triple trapezoid wing planforms has been considered suitable and were examined in detail. Bi-plane alternative has been eliminated at the beginning. High-wing has been selected for the vertical wing positioning. Payload capacity has been decided to be 24 balls according to calculations done via Excel tables. A tractor type electrically powered engine has been accepted to provide thrust with a twenty-two inch propeller.

1.2. *Preliminary Design*

At this stage, the choices made in the preceding step have been intensively examined. Especially, aerodynamic and structural calculations, and weight estimations have been settled.

1.2.1. *Design Alternatives*

Double trapezoid form has been chosen for wing planform. Also, assembly detail 2x12 arrangement of the balls has been examined. Landing gear and wing mounting has been discussed. Location and the style of load-unload alternatives (front top gate, rear top gate, rear end gate) and gate opening-closing problems discussed.

Three alternatives for landing gear were lattice structure, composite and hollow pipe.

1.2.2. *Design tools*

In addition to the design tools used during conceptual design stage, Ansys, Catia and MotoCalc were used. Catia was used for volume-weight calculations, while Ansys was being used for structural analysis.

1.2.3. *Preliminary Design Results*

At the end of this stage, two cylindrical packets of batteries have been decided to be located one on top of the other, between the two series of balls, through the fuselage. Front top gate was decided for loading-unloading the balls. For landing gear, a hollow pipe, traildragger type that is assembled to the wing, has been selected.

1.3. *Detail Design*

In this stage, the performance of the aircraft has been tried to improve as much as possible. Also, solutions to some structural problems have been reexamined. One of these problems was the center of gravity problem. After dynamic and static stability and control analysis, the manufacturing details would be determined.

1.3.1. *Detail Design Alternatives*

The topics of this stage was the position of the center of gravity relative to center of lift (fore or aft) and the positioning angle of the tail according to those two alternatives.

1.3.2. *Design Tools*

Excel, AutoCAD 2000, Catia and Ansys were used as the design tools of this stage.

1.3.3. *Detail Design Results*

Double tapered high wing was decided for wing configuration. Fuselage will carry batteries and softballs like 2 x 12 configuration. Empennage would go a little aft position according to preliminary design. Manufacturing methods were considered towards the end of this phase.

1.4. *Manufacturing Phase*

Materials and processes are investigated for construction of our aircraft. Composite material was decided to be the main manufacturing material. Manufacturing exercises have been done prior to starting manufacture process for about two weeks.

2. MANAGEMENT SUMMARY

Architecture of the Design Team

The ATA-4 group was founded by two ITU Aeronautical Engineering seniors on September 24, 2001 to join AIAA's Design/Build/Fly-2002 competition. Number of participating students reached six in the following days. The ATA-4 group is divided into 3 main subgroup: Design, production and logistics. In general seniors make up the design group because of their academic background while juniors and sophomores take part in the manufacturing and logistics groups. But this organization was not strictly applied because of not having experienced members and also enough members in each subgroup.

The design group consists of five subdivisions that are supposed to design various systems and components. These subdivisions work in coordination with each other. Meanwhile the design group works on the design process, it communicates with the production group to confirm the manufacturability of the design. Some group members take part both in the design and production groups in order to maintain this control.

Before construction starts, the production group is busy with building construction skills, seeking for materials, purchase of materials, and seeking for sponsors. Determining which materials to use, suppliance of tools, strength calculations and different production techniques are to be realized by this group.

The logistics group is responsible for all kinds of official and unofficial communications, writing of reports, struggling to find sponsorship, meeting possible sponsoring firms, organizational planning, liaison, coordination, keeping a record of bills, technical drawings and necessary research. The group also searches for ways of transportation and hotel information for the competition. All these things can be seen in Table 2.1.

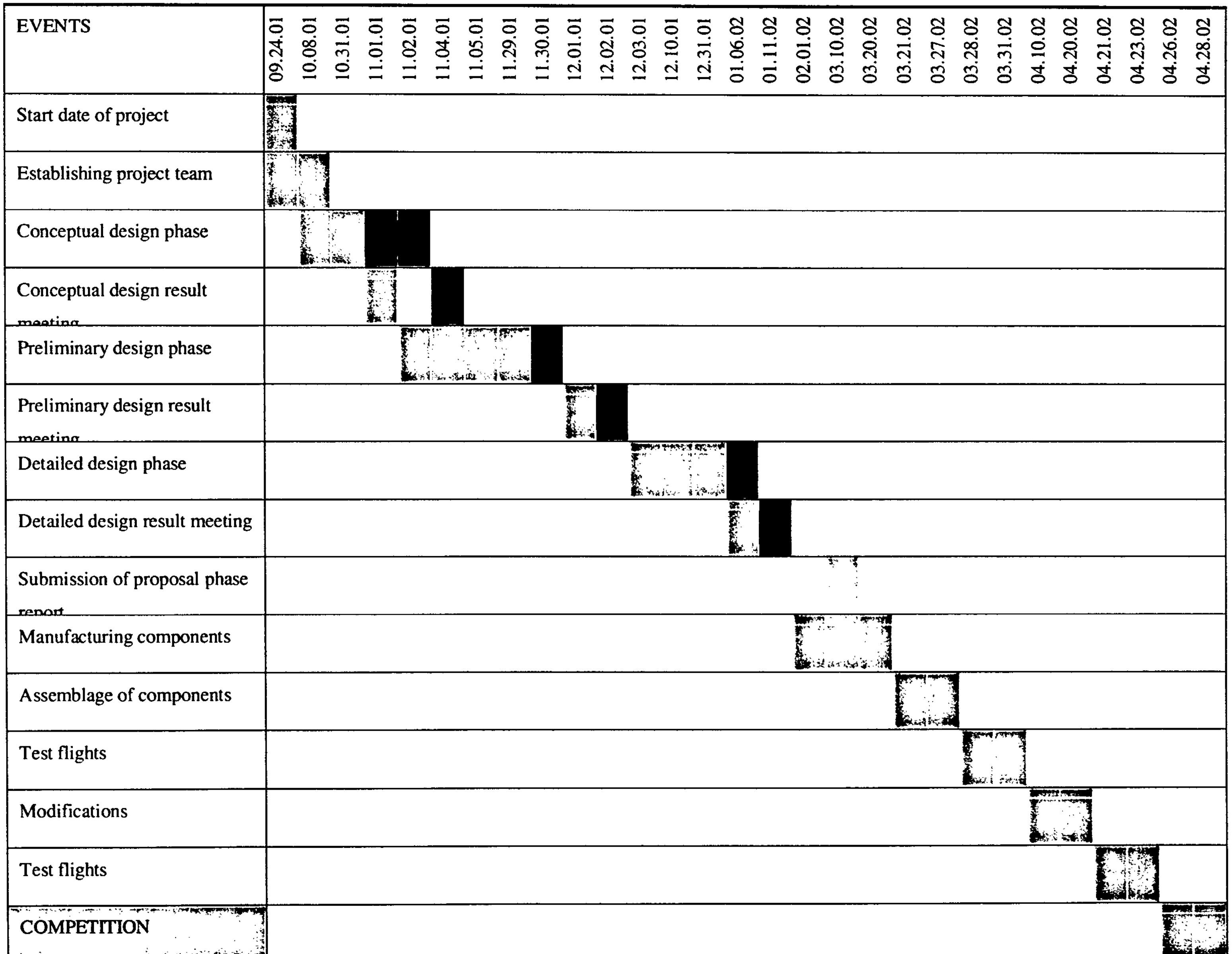
Table 2.1. Architecture of the ATA4 Competition Team

ATA- COMPETITION TEAM		
DESIGN GROUP	MANUFACTURING GROUP	LOJISTIC GROUP
WING DESIGN SUBGROUP		COORDINATION
FUSELAGE DESIGN SUBGROUP		COMMUNICATION
EMPENAGE DESIGN SUBGROUP		FINANCIAL
LANDING GEAR DESIGN SUBGROUP		SPONSOR RELATIONS
ELECTRICAL AND PROPULSION SYSTEMS DESIGN SUBGROUP		REPORT PREPARATION

Table 2.2. Personnel assignment areas table

ATA4 TEAM SUBGROUPS	*PERSONAL ASSIGNMENT AREAS*	HUSEYIN MUTLU	ILYAS TOPRAK	CIHAN KAN	ILHAN YILDIRIM	ONUR SISMANGIL	ALI GUNDOGDU
DESIGN	WING	0	5	5	0	0	0
	FUSELAGE	0	0	0	5	0	0
	EMPENAGE	5	0	0	0	0	0
	LANDING GEAR	1	0	2	0	0	0
	ELECTRICAL AND PROPULSION SYSTEM	4	0	0	0	0	0
MANUFACTORING		2	4	2	2	2	4
LOGISTIC	COORDINATION	5	0	0	0	1	0
	COMMUNICATION	5	0	0	0	1	0
	FINANCIAL	5	0	0	0	1	0
	SPONSOR RELATIONS	4	3	0	0	5	1
	REPORT PREPARATION	5	1	3	2	5	0

Table 2.3. Time table of DBF2002 ATA4 project



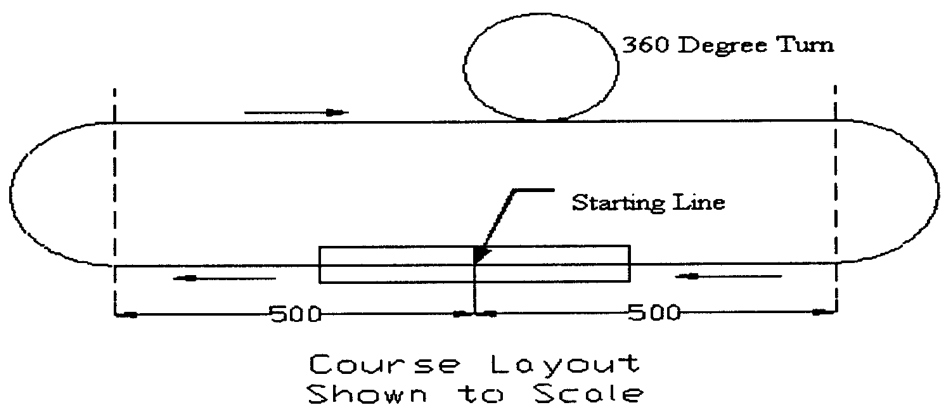
Blue colored boxes indicate the planned time. red boxes achieved time

3. CONCEPTUAL DESIGN

Although we went out in the same order of design phases we did not apply the classical rules for design phases. In the concepptual design phase every component of the aircraft have been investigated and analysed individually. Then the results of these studies were explained in a design group meeting, and the matter of how these components alternatives could be used in harmony was discussed. We set some design requirements for our design and fitting with competition rules. These are:

1. 130ft maximum take-off distance at sea level
2. 98ft/s cruising speed at sea level
3. 130ft maximum landing distance at sea level
4. Accomplish the mission profile (Fig 3.1.) for 6 times
5. To be able to carry 11lb payload
6. Bearing 26 ft/s side wind
7. To be able to exhibit the performance mentioned above with minimum Rated Aircraft Cost
8. To have the minimum weight
9. To be able to carry easily, so aircraft's parts must be separable and carried individually

Fig 3.1. Mission Profile:



The main components are wing, fuselage, empennage, landing gear and propulsion system.

3.1. *WING*

3.1.1. *Airfoil Selection*

Wing airfoil is the most important element of an aircraft because of many reasons.

Airfoil affects the performance parameters such as cruising speed, take-off and landing distances, stall speed, handling qualities and aerodynamic efficiency.

Instead of designing a new airfoil during the conceptual design of the wing, we preferred to search for airfoil with small Reynolds numbers that are especially designed for model airplanes.

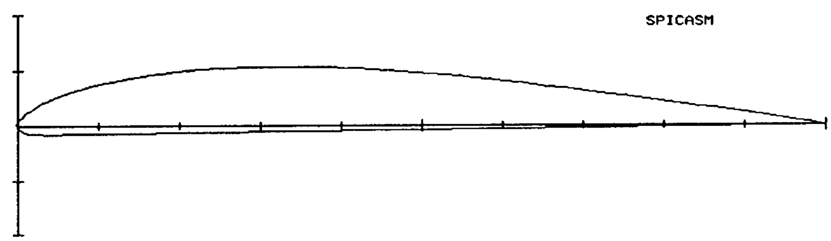
Aircraft's maximum lift coefficient at probable Reynolds numbers, aerodynamic efficiency (especially (L/D)max) and moment coefficient have been the most important parameters for determining the airfoil. The declining characteristics of moment coefficient and lift coefficient have been investigated for each airfoil after

the maximum angle of attack that gives the maximum lift. Only the airfoils with wind tunnel results have been considered for evaluation. The following airfoils were taken into consideration: Anderson, Cody-model, Gemini, Margarita Model, EPPLER 193, EPPLER 331, EPPLER 193 MOD. These airfoils are shown on Fig 3.2. All these airfoils were taken from Internet.

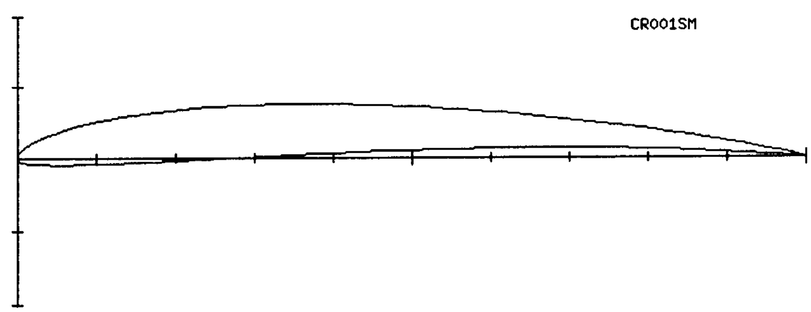
Although Margarita airfoil suited our design the best, we did not have the high precision tools to produce a wing with such a airfoil. We chose the most suitable airfoil for
us among other airfoils we investigated which was Eppler 193 MOD. The airfoils' C_l-C_d and C_l-alpha, C_m-alpha graphics are shown in graphic 3.1.

Fig 3.2. Investigated airfoils during conceptual design

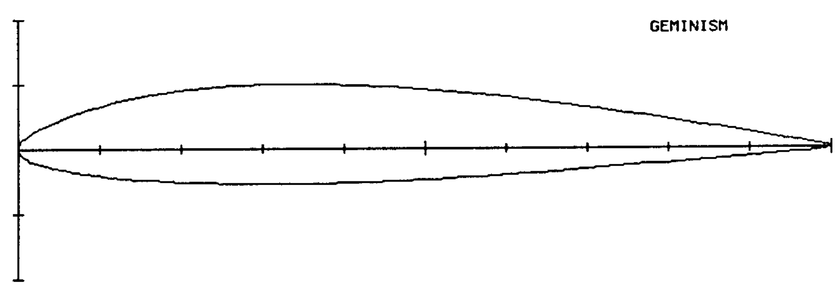
Anderson airfoil:



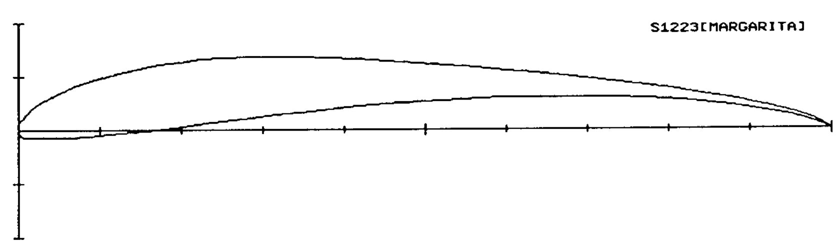
Cody-model airfoil:



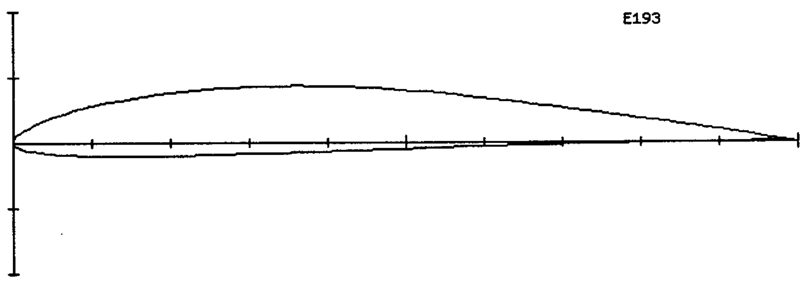
Gemini airfoil:



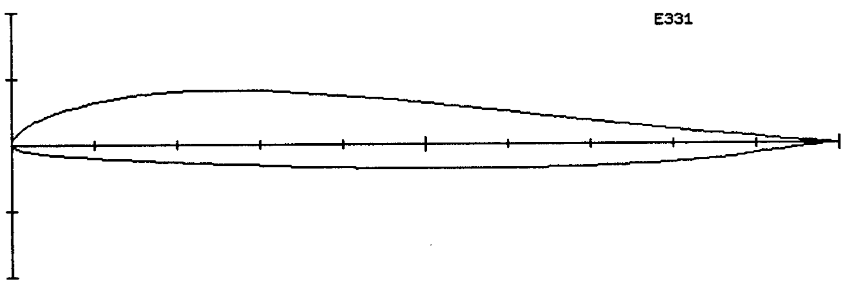
Margarita airfoil.



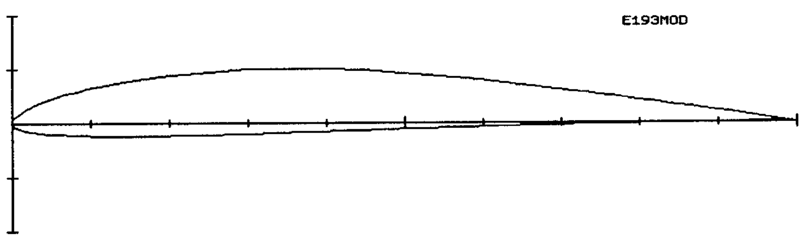
Eppler 193 airfoil:



Eppler 331 airfoil:

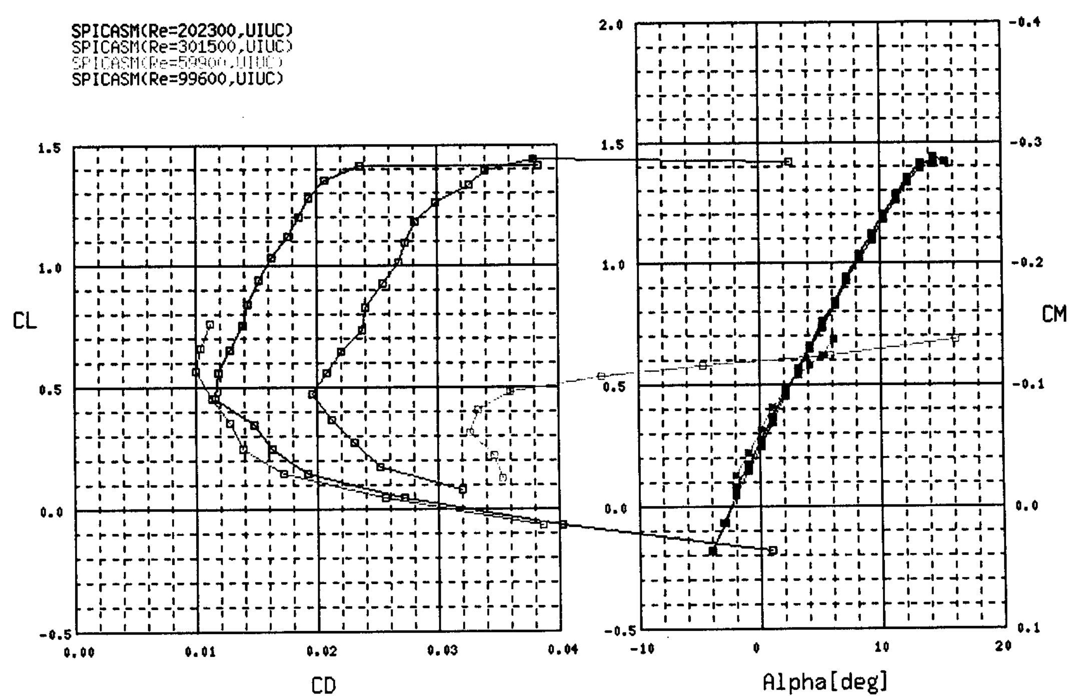


Eppler 193MOD airfoil:

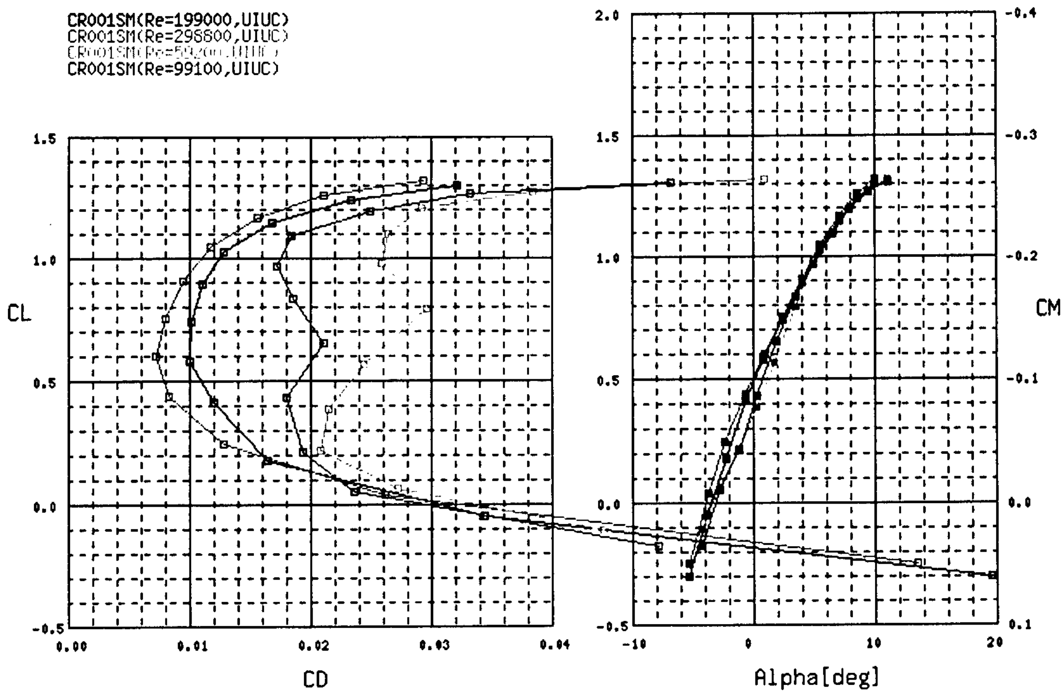


Graphic 3.1.

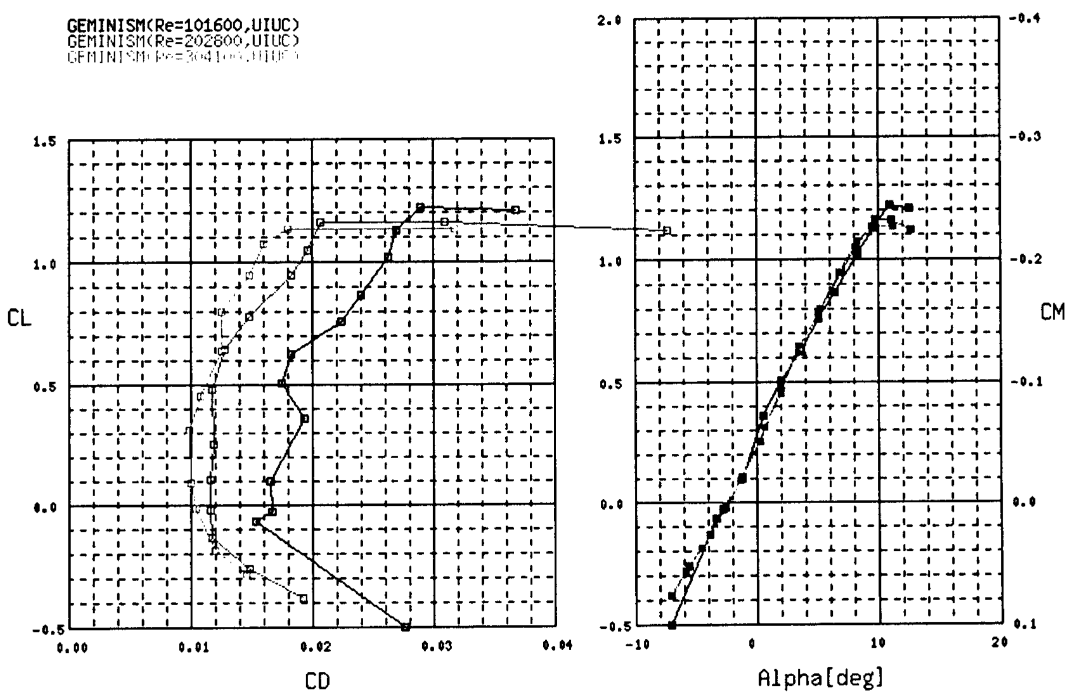
Anderson for four different Reynolds numbers;



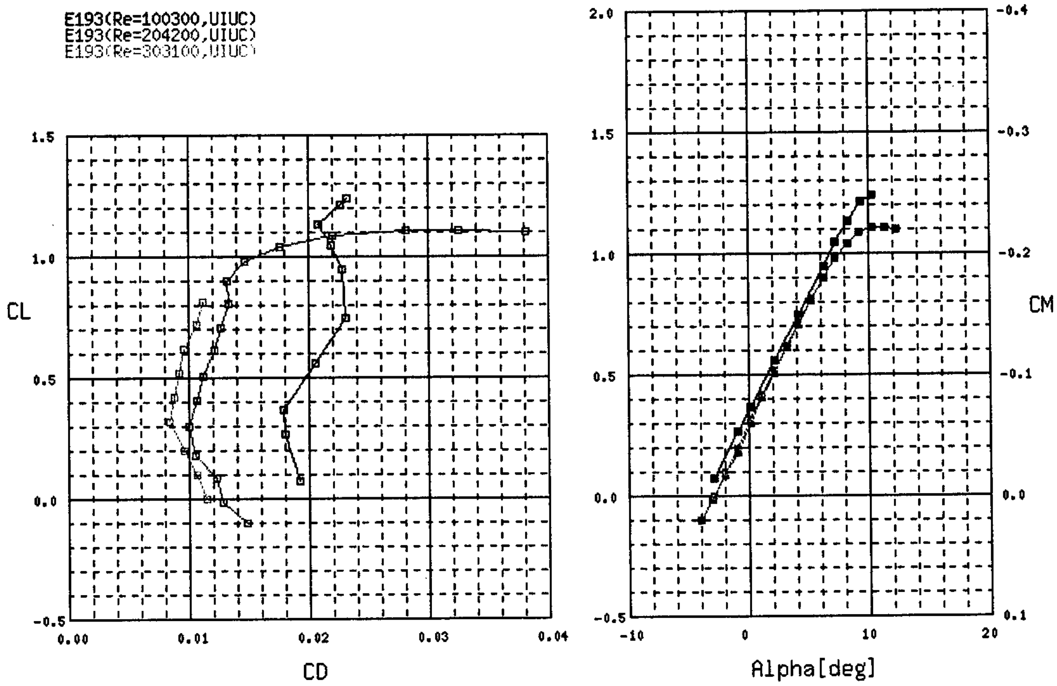
Cody for four different Reynolds number;



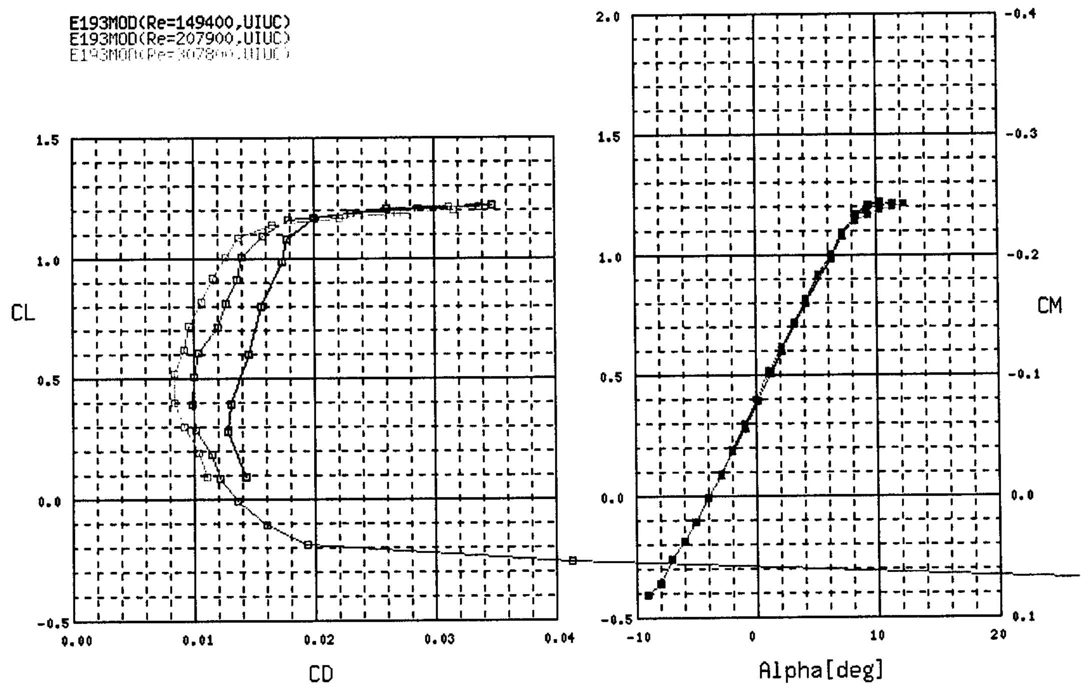
Gemini for four different Reynolds number;



Eppler193 for four different Reynolds number;

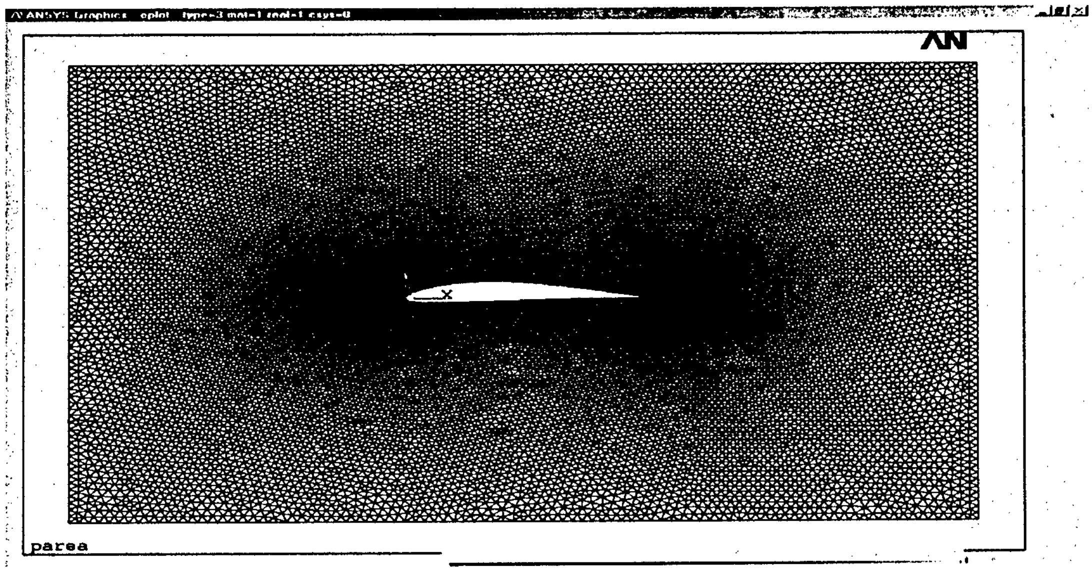


Eppler193MOD for three different Reynolds number;

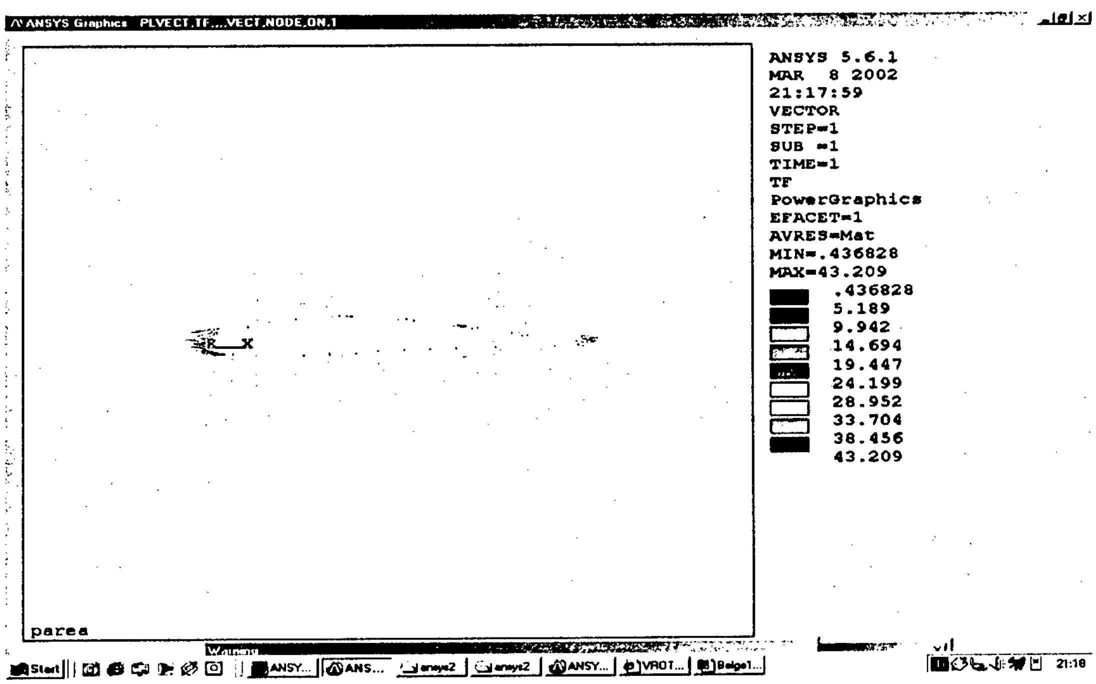


We analyzed the velocity field around E193MOD airfoil to estimate Cp distribution so have an idea about theoretical lift and drag. Ansys results of the velocity field on E193 MOD airfoil can be seen below on Fig 3.3.

Fig 3.3.



The meshing process of E193MOD by ANSYS



The velocity field around E193MOD airfoil

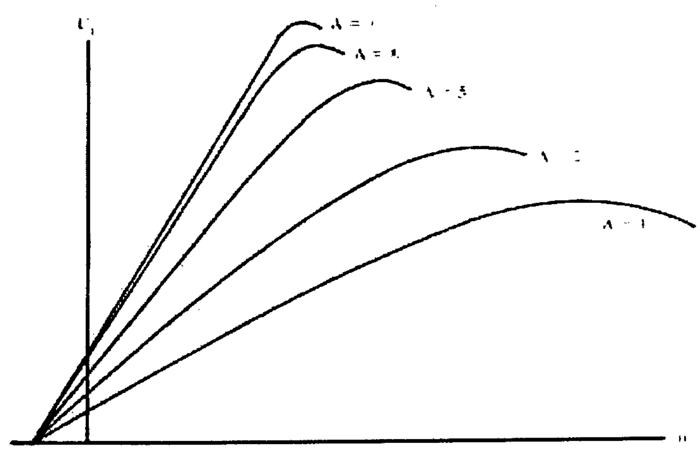
The Ansys program does its calculations according to the potential two dimensional incompressible flow principle. We assume incompressible flow because our aircraft's speed is very low. The free stream velocity has been taken 30 m/s (98 ft/s). On the Fig 3.3 the values of speed (in SI) can be seen by colors.

3.1.2. Wing Geometry

3.1.2.1. *Aspect Ratio*

During conceptual design, a wing planform with low Rated Aircraft Cost for a particular aspect ratio has been investigated to satisfy the design expectations. Empty weight of the aircraft was approximately calculated depending on the experiences of former design teams of our faculty. We had an idea about the wing area by looking at the performance equations.

Instead of the wing airfoil, the wing planform determines the stall characteristics of the aircraft for low aspect ratio values. This is because of the vortexes caused by the three-dimensional flow around the wing, which are efficient on longer parts of wings at low aspect ratios. For this reason, in order to decrease the vortex effect of the three-dimensional flow and to make the two-dimensional airfoil more efficient for stall characteristics, wing planforms with aspect ratios higher than 8 were focused on. For values of aspect ratio greater than 8, as can be seen in Graphic 3.2, aspect ratio does not have a considerable effect on wing performance and lift constant.

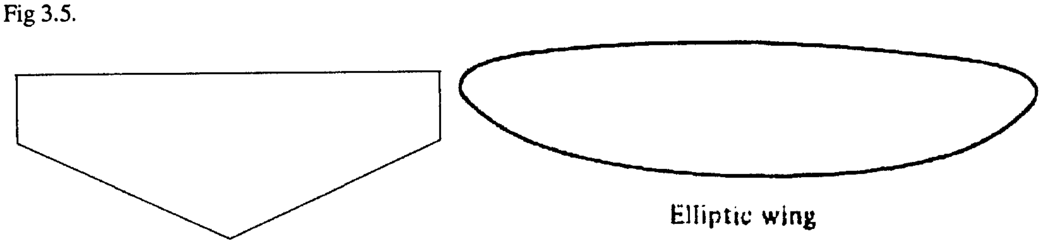


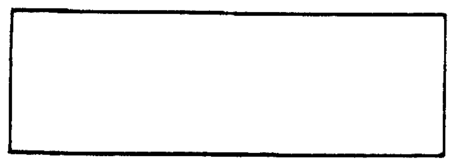
Graphic 3.4. Effect of aspect ratio on lift

3.1.2.2. *Wing Planform*

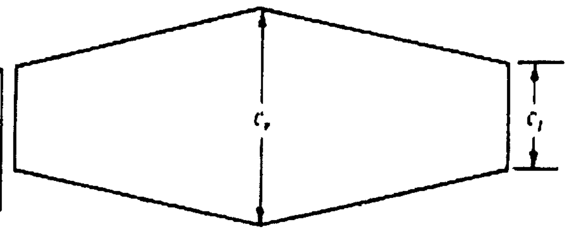
Wing planforms that have been evaluated during conceptual design can be listed as follows: Deltawing, Eliptical, Rectangular, Single trapezoid, Double trapezoid, Triple Trapezoid, Forward swept with canard, Sweptback. Some of these wing planforms were shown below in Fig 3.5.

Fig 3.5.

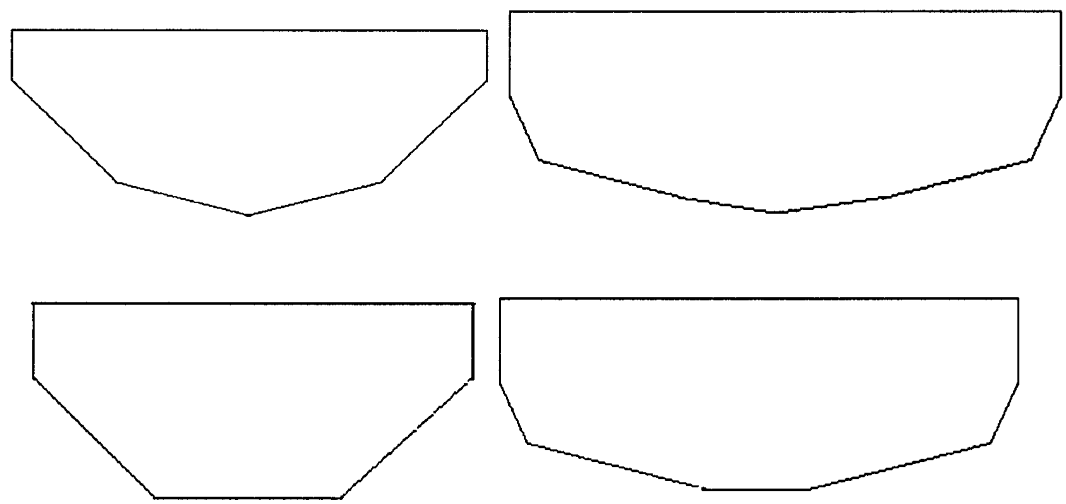




Rectangular wing



Tapered wing



Deltawing was eliminated at the very beginning for its low aspect ratio. Forward swept wing with canard configuration was eliminated because of the difficulty of controlling the pitching moment caused by the canard.

Theoretically elliptical wing planform gives the best performances for all aspect ratio and wingspan values. The lift curve is elliptical on such a wing and this minimizes the induced drag. Rated Aircraft Cost calculated for elliptical planform was low enough. Unfortunately we lack the production ability for a wing with such a precise geometry.

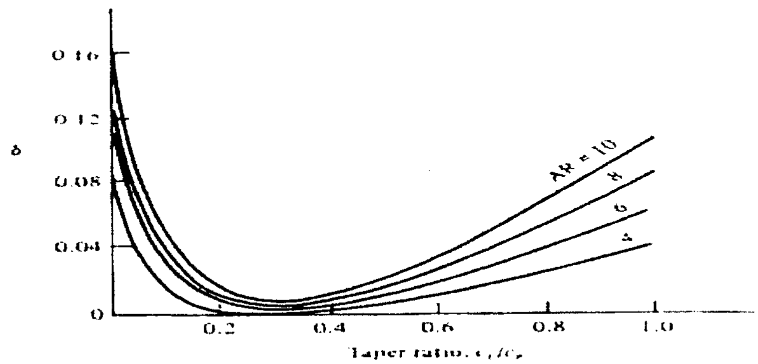
Rectangular planform was considered for its simplicity, reliability and good stability. This planform alternative has theoretically the best Manufacturing Man Hours (MFHR) according to the rules set by contest among other considered planform alternatives for a specific aspect ratio and wing area. However, it deviated considerably from the elliptical force distribution for a particular aspect ratio and caused intolerable drag. Even though not much force acted on the wingtips, they were unnecessarily built strong which caused the wing to be heavier. Rectangular planform was eliminated for the mentioned reasons.

Tapered wing planform was chosen as the second best solution. Chord length of a wing with Tapered planform was changed by giving different taper ratio values to the planform. The best taper ratio values have been found for aspect ratio values higher than 8 by using the theoretical results for approaching an elliptical lift distribution, Table3.1. Theoretical results show that a single tapered planform can be designed to create %2-3 more theoretical induced drag according to the elliptical force distribution. We used Graphic 3.3.a,b for preparing Table 3.1 and analyzing trapezoid wing planforms of different aspect ratios.

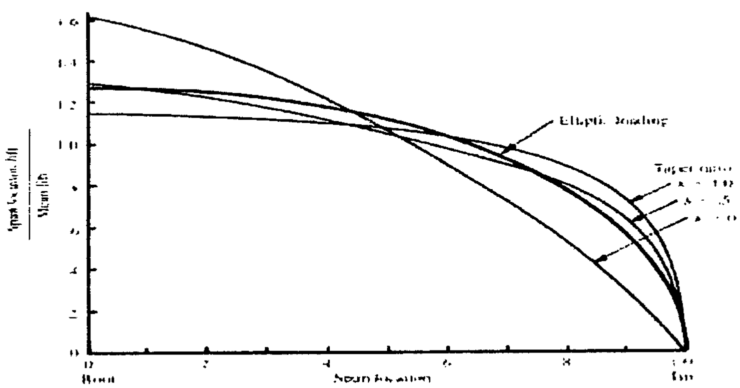
Table 3.1 Single tapered wing planform characteristics

Aspect Ratio	Best Taper Ratio	Deviation from Elliptical Form
8	0.308	0.004
8.5	0.309	0.0045
9	0.31	0.005
9.5	0.311	0.0055
10	0.312	0.006

Graphic 3.3.a. Taper ratio-deviation from elliptical lift distribution relationship for different aspect ratios



Graphic 3.3.b. Taper ratio-lift distribution relationship



It can be seen from Graphics 3.3.a and b that by selecting appropriate taper ratios between 0.3 and 0.4 we can approach elliptical lift distribution on tapered wing planforms for different aspect ratios. It was theoretically possible to reach a better elliptical force distribution and reduce drag for double and triple combinations. As a result, tapered wing planform was decided to be chosen.

Comparing of investigated wing planforms can be seen from Table 3.2.

Table 3.2.

WING PLANFORM	ASPECT RATIO	TAPER RATIO	MFHR	REFERANCE WING AREA	MAX CHORD
RECTANGULAR	9	1	98.82	10.76	1.08
	8	1	89.66	10.76	1.22
ELIPTICAL	9	0	96.2	10.76	1.378
	8	0	92.36	10.76	1.46
TAPERED	9	0.375	97.77	10.76	1.574
	8	0.375	94.06	10.76	1.673

Comparing of some planforms lift distribution with elliptical lift distribution visually by Fortran2002 program, provided by the advisor, can be seen Fig 3.5.a, b and c. red line is elliptical lift distribution, blue one is specified planform's lift distribution.

Fig 3.5.a. Rectanular planform comparing

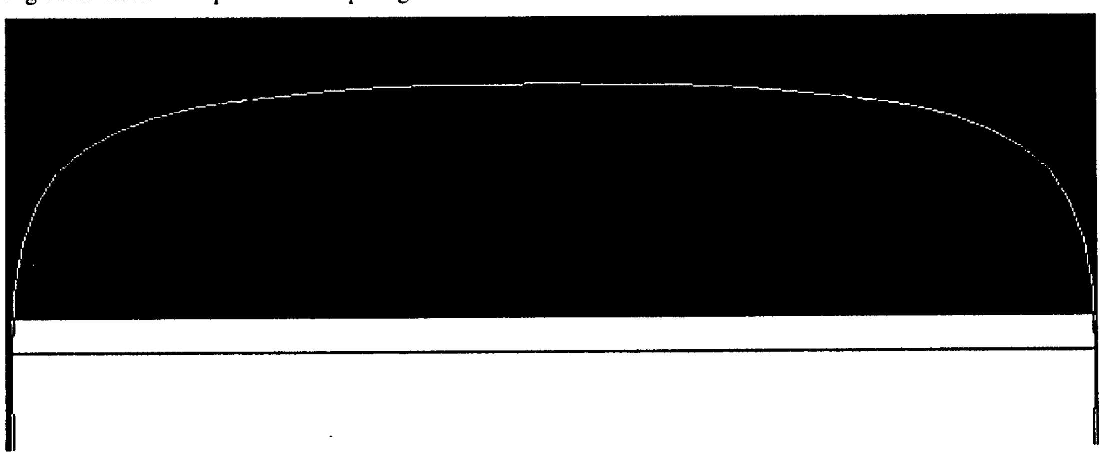


Fig 3.5.b. Comparing single tapered planform (type 1)

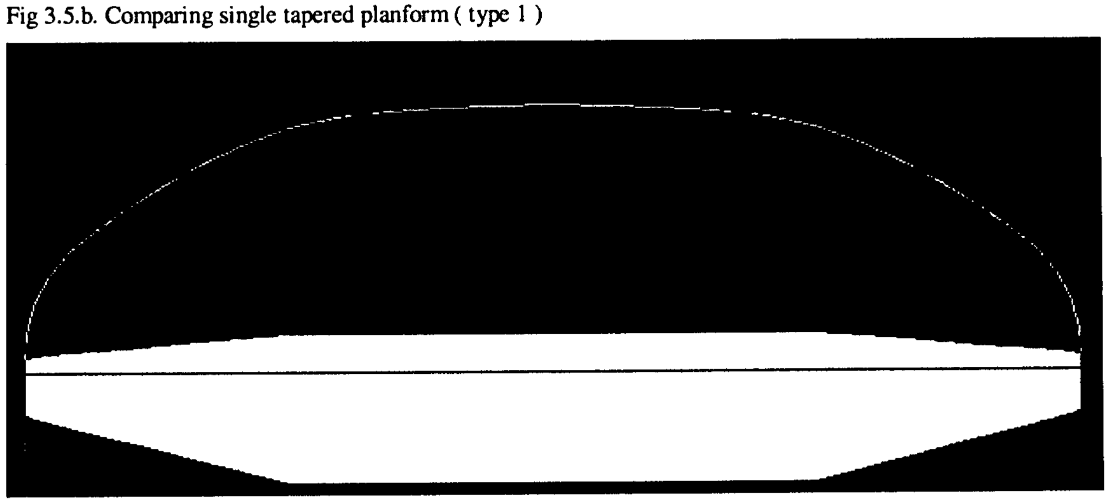
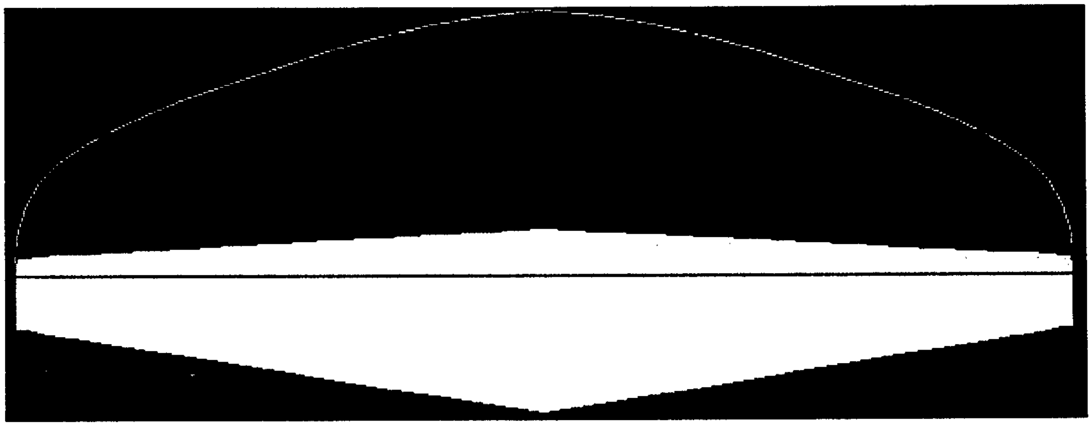


Fig 3.5.c. Comparing single tapered planform (type 2)



As can be seen from the above figure, tapered wing planform has much more better results according to rectangular planform.

3.1.3. *Wing Configuration*

At this stage of the design process, the optimum configuration that would be used most efficiently with the chosen planform was researched. Conditions about structure and stability helped us make our choice.

Never evaluated by former design teams of our institution, the biplane configuration gave better results than a monoplane. However, a monoplane with the same wing area provided more lift than the biplane configuration. This can be explained by the negative interaction inbetween the two wings that decreases the lift. Also biplane configuration was difficult to construct. Rated Aircraft Cost of the biplane was another handicap. Three dimensional flow effect was not very efficient since aspect ratio was high. Because of this and the concerns about the Rated Aircraft Cost, wingtip was not used. The vertical placement of the wing on the fuselage was another important point for stability and structural problems. High wing (full cantilever) configuration was selected. High wing was more stable because of dihedral effect and it did not create structural problems as the mid wing does.

The same airfoil was used at the root and at the tip of the wing. In order to reach elliptical lift distribution, the idea of geometric twisting was discussed. It has been discarded for its difficulty of construction.

In addition to the dihedral effect of the high wing configuration, a decision was taken to give dihedral angle to the wing for improving structural stability. The dihedral angle was decided to be low. Former years' test results show that high dihedral angles affected maneuverability negatively; therefore the dihedral angle was not taken a big number.

Because we did not possess the production capability to give direct dihedral angle, we decided to give natural dihedral angle to the wing. The upper face of the wing would be straight while the thickness of the wing would decrease from the root to the tip on the lower face. As indicated above, the dihedral angle would not be allowed to be more than 3°, otherwise the aircraft would become excessively stable and would lose maneuverability.

When the performance calculations were compared with last year's aircraft Bosphorus Blue's documents, it was understood that the wing area had to be greater than 1 m² to satisfy the design requirements of a 33 lb aircraft.

Because of the production difficulties, idea of twisting the wing was abandoned.

Instead of calculating the required reference wing area according to level flight conditions we optimized the area to make aircraft to have the capability of taking-off within 130 ft without using flap mechanisms. These calculations results can be seen in Table 3.3. The incidence angle of wing was adjusted to be four degrees so that at this angle it is possible to optimize the wing aerodynamic efficiency (L/D) at level flight.

Table 3.3

Velocity (m/s)	cl-best	Srequired (sq-ft)
25	0.8	13
30	0.8	10.8
35	0.8	8.6

3.1.4. Figure of Merit

CONSIDERATION	ELIPTICAL	DELTA	RECTANGULAR	TAPERED
MANUFACTURABILITY	1	2	5	5
DRAG	5	1	2	4
MFHR	3	2	4	3
WEIGHT	3	2	1	3
STABILITY	3	1	4	2
RESULT	E	E	E	SELECTED

During preparing FOM for wing, we assumed that a consideration would take 5 points if it has good and desired properties. It will take 1 as a grade if it is not desired. According to the FOM, we could also choose rectangular wing. But for weight consideration reason we have chosen tapered planform.

3.1.5. *Analytic Tools*

The design team used Fortran and Ansys during the conceptual design phase. Ansys program was used by the design team. It has been used for visualizing the flow vectors around the two dimensional airfoil.

3.2. *FUSELAGE*

3.2.1. *Fuselage Alternatives*

Since the contest rules set a limit to possible fuselage types, there were not many alternatives for arranging the balls. There were three choices as to ball arrangements of 2xk, 3xk and 4xk, where k is the number of the balls seen at front view. In the preliminary design phase we made a payload optimization by a MS Excel sheet.

3.2.2. *Payload Optimization*

We have prepared an MS Excel sheet according to Aircraft Cost Model that is announced this year in order to estimate the Rated Aircraft Cost (RAC) and the Score of the aircraft that we are going to design. We

have made some assumptions as we prepared this sheet. First of all, we designed the wing span of our aircraft for carrying 24 softballs. We calculated the RAC according to this design. Then, we decreased the number of softballs that we would have to carry two by two and we expressed the changes on the aircraft with a suitable formula. We hypothesized that the wing span shortened 2 inches with each pair of softballs we took out. We did not take into consideration the decrease in weight as the wing shortens; we considered the weight of the wing constant. This consideration was taken into account within the factor of safety for calculating Rated Aircraft Cost. The length of the fuselage also decreased by the diameter of softball because when we took two softballs out, a line of softballs would be unloaded. Also when we designed the fuselage in order to carry 24 softballs, we reduced the weight we assumed for the 12 softballs diameter long fuselage, by the diameter of softball in each step. Besides this reduction in weight, we also shortened the fuselage length by the diameter of softball. When we calculated the Single Flight Score used for calculating the Total Flight Score and the Score, we took the Total Mission time as 10 minutes, as mentioned in the rules of the competition. We decreased 10 seconds for a pair of softballs we took out. It was inevitable to have other alterations on the aircraft as we made the changes mentioned above. However, we assumed components such as wing control surface, total battery weight, maximum chord length, vertical surface, control vertical surface, control horizontal surface, servo, motor, propeller were constant and we calculated according to this assumption.

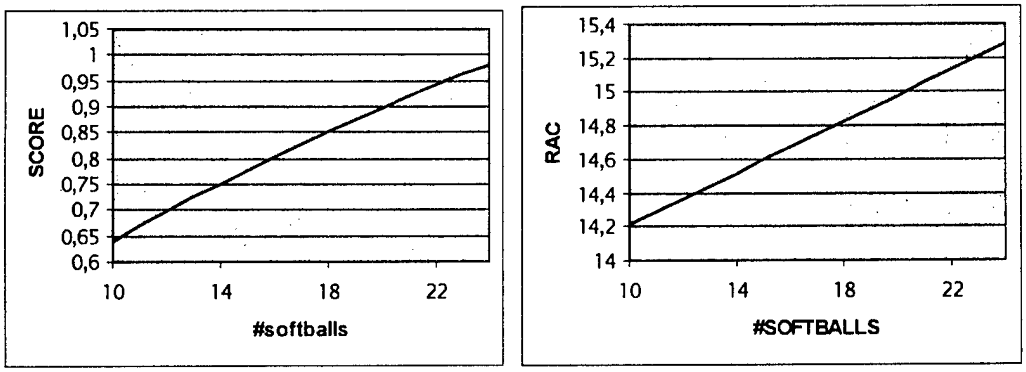
From the graphics that were drawn by the results obtained from this sheet, we observed that RAC decreases as the number of balls decreases. The decrease in RAC seems to be advantageous for us while calculating the Score. But meanwhile calculating the Single Flight Score that is used for calculating Total Flight Score, the increasing number of softballs that contributes to the numerator and the decreasing of the Total Mission Time that contributes to the denominator cause the Score to increase. These results can be seen in the graphics. After these observations in the conceptual design phase, we decided to carry 24 softballs.

Table 3.4. Payload optimization table

[illegible] balls	#	24	22	20	18	16	14	12	10
[illegible] wingspan)	ft	9,6	9,43333	9,26667	9,1	8,93333333	8,76667	8,6	8,43333
[illegible]CS (wing control surface)	#	2	2	2	2	2	2	2	2
[illegible]	ft	7,874	7,56	7,24	6,92	6,60	6,28	5,96	5,65
[illegible]	lb	4,8722	4,8722	4,8722	4,8722	4,8722	4,8722	4,8722	4,8722
[illegible]	ft	1,476	1,476	1,476	1,476	1,476	1,476	1,476	1,476
[illegible]	#	1	1	1	1	1	1	1	1
[illegible]	#	1	1	1	1	1	1	1	1
[illegible] (horizontal surface)	#	1	1	1	1	1	1	1	1
[illegible]	#	0	0	0	0	0	0	0	0
[illegible] (servo)	#	5	5	5	5	5	5	5	5
[illegible] (motor)	#	1	1	1	1	1	1	1	1
[illegible] (propeller)	#	1	1	1	1	1	1	1	1
EW (aircraft empty weight)	lb	33,0693	32,4433	31,8173	31,1913	30,5653	29,9393	29,3133	28,6873
RAC		15,2822	15,1293	14,98	14,8234	14,6704904	14,5176	14,3646	14,2117

[illegible] time	sec	600	590	580	570	560	550	540	530
SCORE [illegible]		0,98153	0,94104	0,89797	0,85214	0,80336198	0,75144	0,69615	0,63726

Graphic 3.4. The variation of score and RAC with number of softball



3.2.3. *Fuselage Type Selection*

Considering the weight and drag forces as comparison criteria, two of those designs had to be eliminated. Weight is the most important problem of an aircraft; thus, it is impossible to ignore weight properties during the elimination process. During the calculation of weight, thickness of the fuselage wall considered to be ½ inch and covered by a layer of fibreglass, which has a density of 0.02765 lbm/sqft. The foam density is thought to be 0.937 lbm/cubicfeet. Calculations were according to empty weight. Last year's competitors provided all of this data.

3.2.4. *Design programs*

Aero Design Drag Estimator Trial Version has been used for drag force calculations. Drag forces have been calculated in Newton scale, at standard atmosphere conditions and at the average speed of 82 ft/s. Tables of Microsoft Excel has been used for weight calculations and payload optimization. In addition, the aircraft has been decided to carry 24 balls.

Fuselage Type	Drag (N)	Weight (N)
Double	0,9674	0,936
Triple	1,1375	0,853
Fourfold	1,2026	1,081

Fourfold fuselage has been immediately eliminated due to its high weight and drag forces. Also, fourfold fuselage's effect on the distribution of load was the greatest. During transition from double form to triple form, there was a drag force increment of 17.6 % while the weight decrement was of 8.9 %. Considering this data and the tail design, fuselage of the double form, which is longer, has been selected.

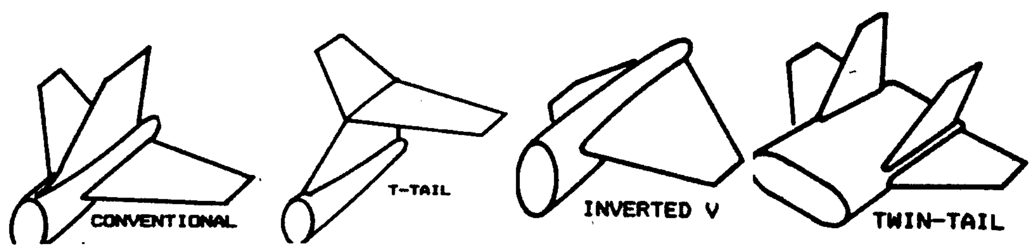
3.3. *EMPENAGE*

Some high priority goals we aimed for during empennage design were:

- To maintain minimum weight contribution of the empennage to the aircraft's weight (empenage weight caused center of gravity problems during the previously designed ITU aircraft) and to overcome the center of gravity problem
- To produce an empennage structure for the lowest drag characteristics
- To maintain the stability and control of the aircraft while minimizing the Rated Aircraft Cost contribution of the empennage

3.3.1. *Design Alternatives*

Six alternative empennage arrangements that would fit the other components of the aircraft were considered while pursuing these goals: conventional tail, T-tail, inverted V-tail, twin tail, only horizontal stabilizer and canard configuration. Some of these configurations can be seen below.



The reason we considered conventional empennage arrangement was that we would be able to maintain stability and control with the desired minimum weight. In addition to that, a conventional tail would be most adaptable to most of the airframe structures we have been considering. Conventional tail has had a value ahead of other arrangements for its reliability and positive reputation.

Possessing all the characteristics of a conventional tail, the T-tail also presents some advantages. In spite of its advantages, the T-tail requires more support for the horizontal stabilizer, which creates an extra weight problem. This would make a T-tail heavier than a conventional tail.

Inverted V-tail was thought for a twin fuselage alternative. It supplies the proverse roll-yaw coupling needed for the aircraft's control. Although a reduction in surface area was aimed, this goal was not adequately achieved. It would be helpful to indicate that inverted V-tail arrangement is not commonly used in UAVs and that there can be problems in controlling an aircraft with such an empennage arrangement. Also there were no twin fuselage design alternatives that this tail would be used in harmony.

Twin tail configuration can be considered for cases like 3x8 and 4x6 softball arrangements. Vertical surfaces are diminished in this structure when compared to conventional tail. For this type empennage configuration there would be no blanketing problems at high angle of attacks. Although more efficient than the conventional tail, the twin tail arrangement is relatively heavy.

Only horizontal stabilizer tail configuration was considered for a twin engine aircraft. This configuration reduces empennage weight considerably and reduces drag by decreasing wet surface area. Despite the other designs that have been considered, twin engine increases weight, which is a disadvantage. Extra engine and propeller mean extra money. The costs would increase therefore this configuration was not preferred either.

Canard configuration was one of the options considered at the beginning of the design period. Production of the canard represented no problem for the production team but considering the problems that might occur with stability and control, we eliminated the canard configuration.

As a result in the conceptual design phase conventional tail was chosen for further analysis in the preliminary design.

3.4. *LANDING GEAR*

Landing gear is a component for decreasing shocks during landing which damps the kinetic energy of an aircraft by bending. There is no need for complex damping structures because the aircraft we designed is a light, low-speed aircraft. Complexity of a landing gear increases costs and weight although it contributes well to the damping. Therefore a light landing gear that is able to carry structural loads as well as doing the damping, is needed. Solid Spring type landing gear would be enough for these design requirements.

Another design parameter for the landing gear is being retractable or non-retractable. Retractable landing gear is advantageous for reducing drag. On the contrary, retractable landing gear components are relatively expensive, complex to manufacture, heavier and probability of failure is higher. For these reasons retractable landing gear loses attraction for a low speed aircraft. Non-retractable landing gear presents more drag than retractable landing gear but it can be reduced with correct geometry. Another problem presented by the non-retractable landing gear is that the manoeuvring ability of the aircraft decreases. It was decided that non-retractable landing gear would be more suitable for our aircraft when the two types were compared.

There are two landing gear location options that we have considered: Tricycle and taildragger. Tricycle type landing gear would be heavier than taildragger. The engine and the propeller are at the front of our design and taildragger is mostly used for such configurations. We decided on a taidragger configuration because the main landing gear would keep the propeller high enough.

3.5. *PROPULSION SYSTEMS*

We decided to use single motor for propulsion system. For this decision, our budget availability plays a very important role. Also the motor we have is probably one of the best motors that we could find. We would use tractor type for propeller location. This location is also suitable for our empenage type. The motor and propeller will be placed at the nose of our aircraft.

The batteries would be just behind them. This arrangement will be very efficient when it is considered that the current is too high. The connection wire should be as short as possible. So this will cause the least loss for propulsion system.

<u>*CONCEPTUAL DESIGN RESULTS*</u>

WING AIRFOIL:	EPPLER 193 MOD
ASPECT RATIO:	>8
WING PLANFORM:	TAPERED
WING VERTICAL LOCATION:	HIGH WING
FUSELAGE:	FILLETED CORNERED RECTANGULAR TWO SOFTBALLS SIDE BY SIDE
EMPENAGE:	CONVENTIONAL TAIL
LANDING GEAR:	TAILDRAGGER
PROPULSION SYSTEM:	TRACTOR TYPE, SINGLE MOTOR

4. PRELIMINARY DESIGN

4.1. *WING*

We aimed to get an idea about the definite shape of the tapered wing at the preliminary design phase. We spent time on single, double and triple tapered wings for different aspect ratios (especially 8, 8.5 and 9) and for changing values of the wing area depending on cruising speed.

It would be very easy to manufacture a single tapered wing in three pieces and mount it to the fuselage. Drag characteristics got close to elliptical planform when a fine taper ratio was taken (Fig.4.1.a and Fig.4.1.b). But it was possible to get more efficiency from a double (Fig.4.2.) or triple tapered wing. Two configurations were available for double tapered wing. It would not be very hard to produce these wings in 4 or 5 pieces. Also, it was possible to get close to elliptical lift distribution with a double tapered wing.

Fig.4.1.a

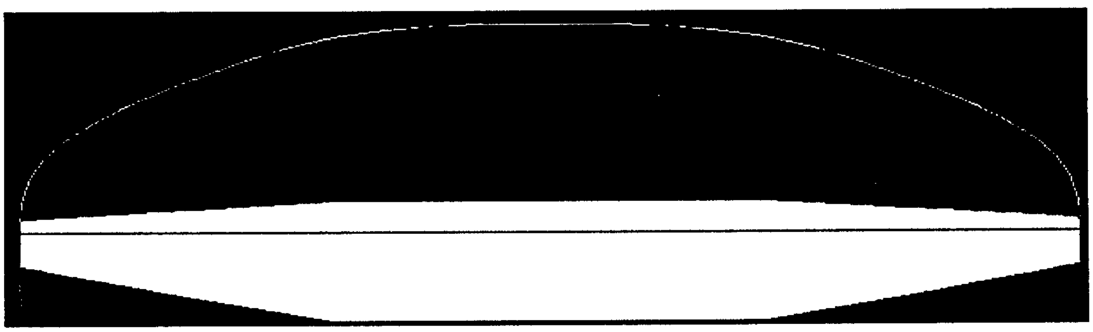


Fig.4.1.b

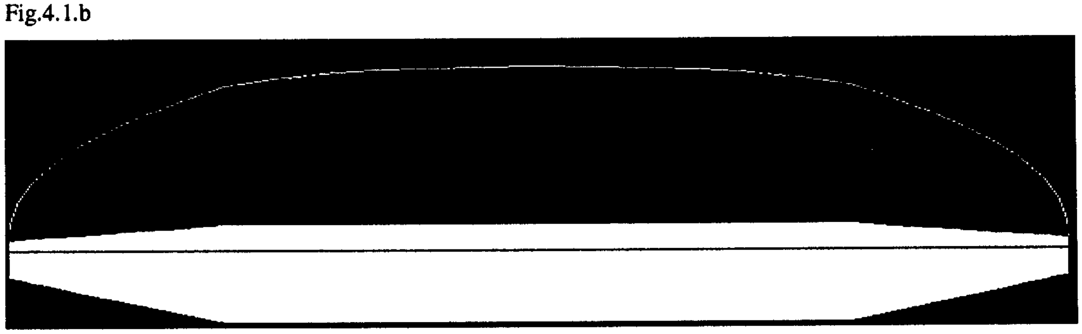
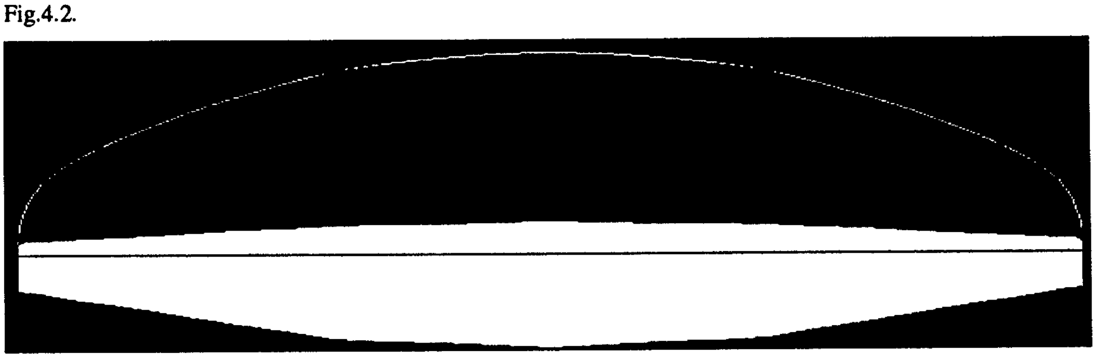


Fig.4.2.



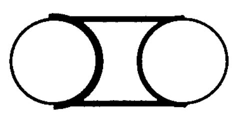
The best efficiency was taken from the triple tapered wing. Elliptical lift distribution was almost reached and the excess induced drag was reduced to less than 0.5% when the taper angles of the triple taper planform were adjusted right. But 6 pieces had to be assembled to build a triple taper wing. One of the

disadvantages of triple tapered wing was the higher MFHR than double tapered wing. When lift distributions, amount of deviation from elliptical form and drag were investigated we saw that double and triple tapered planforms give very good results around an aspect ratio of 8.5. Minimization of Rated Aircraft Cost for limited area and aspect ratio was another important parameter. The decision was taken to compare the calculation results of each planform for an aspect ratio of 8.5 and $10.76ft^2$ wing area. It was observed that double and triple tapered planforms gave fine results by means of Rated Aircraft Cost. Thus, single and triple tapered planform was eliminated.

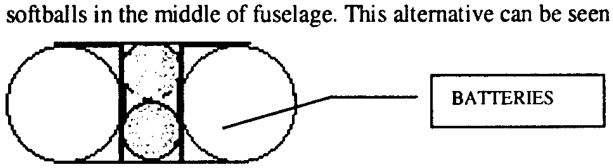
4.2. *FUSELAGE*

The size of the fuselage is limited by the rules, as no balls are allowed to be put on each other. They have to be is a single horizontal row, when looked from front. In the concept design phase the fuselage is thought to be 2x1 fuselage, where there are two balls side by side when looked forward. In the preliminary design phase we go on further design and improvements. As design parameters the shear stresses and the weight are chosen. In the materials that is supposed to be used, which is foam covered with glass-fiber, the normal stresses are small enough compared to the ultimate stress of the material.

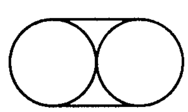
At the first case it is thought to put an I profile between the balls in order to increase the strength. This I profile is made of the same material with the whole fuselage and wing, which is foam with 300MPa ultimate strength. The figure has been exaggerated to show that the balls were separated from each other by a foam wall covered by fibreglass. The figure of this type is shown below:



In the second case the I profile is changed with a single rectangular profile. For this case it is thought to decrease the weight by using less material. For this fuselage alternative we can put batteries between the softballs in the middle of fuselage. This alternative can be seen below;



In the last, 3rd, case all the structures between the balls are removed. The softballs are touching each other. The front view of fuselage would be least one among all fuselage alternatives that has come from conceptual design phase. The last alternative is shown below;



For calculations of the stresses and weights a Microsoft Excel table is used. In this table (Table 4.1) the forces and moments are taken as unit force (1N) and unit moment (1 Nm). From the experiences of the last year's group and our instructors a thin-walled structure made of foam covered with glassfiber can meet the stress requirements. From the calculations the stresses are found in the secure region. The second choice is at the

second place according to weight and stress analysis. As a result, also available to put batteries, the most suitable choice that is the second one, with thin walled single rectangular profile is chosen.

Table 4.1. Fuselage cross section selection table

Number	Max. Shear Stress (Pa)	Max. Normal Stress (Pa)	Possible weight (N)
1	580,3	7930,6	3,355
2	523,4	10073,6	2,699
3	773,2	17370,5	1,872

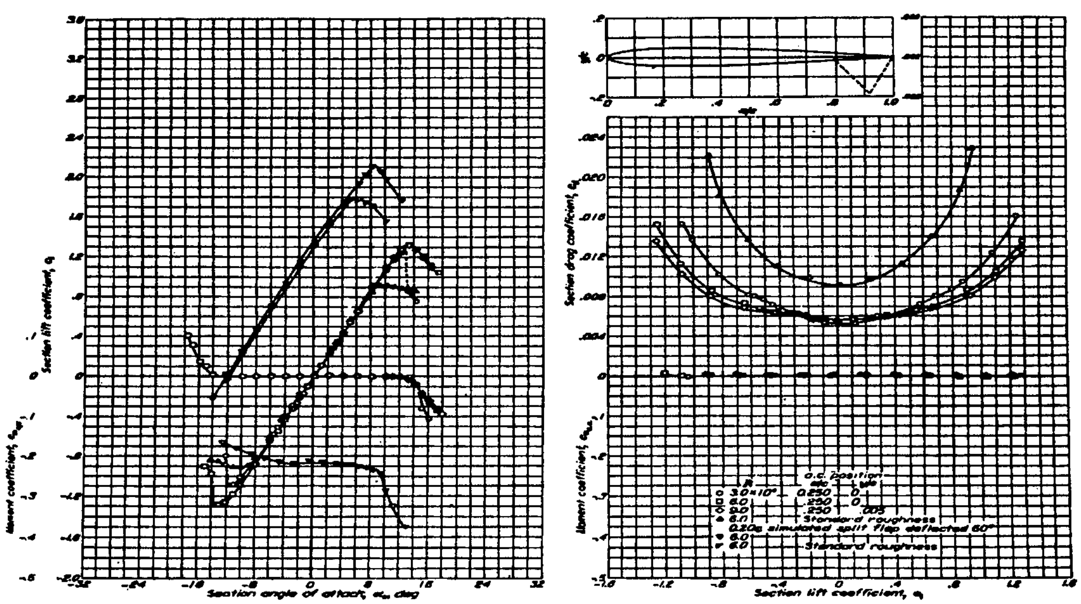
4.3. *EMPENAGE*

In many situations for determining the tail surfaces, although experience and academic knowledge are vital, common sense of the designer team plays much more important role than well known methods. At this phase instead of proceeding with detail procedures, simple methods were used to make our first size estimations. In the beginning, choices were made for parameters like aspect ratio and taper ratio. It was investigated whether the stabilizer would be all-moving or only elevator moving.

4.3.1. *Airfoil Shape*

The basic requirements are that the airfoil section should have a high lift-curve slope and a large range of usable angles of attack. As traditional, symmetrical profiles were agreed to be used for tail surfaces. NACA 0015, NACA 0012, NACA 0009 alternatives were considered for these profiles. Frequent use is made of approximately symmetrical airfoils with a thickness ratio of 9 to 12 percent and a large nose radius. We have chosen a 9 percent thickness ratio for both horizontal and vertical tail surfaces. That is NACA0009, Fig 4.3.

Fig. 4.3. NACA0009 airfoil's lift coefficient, moment coefficient- angle of attack graphics



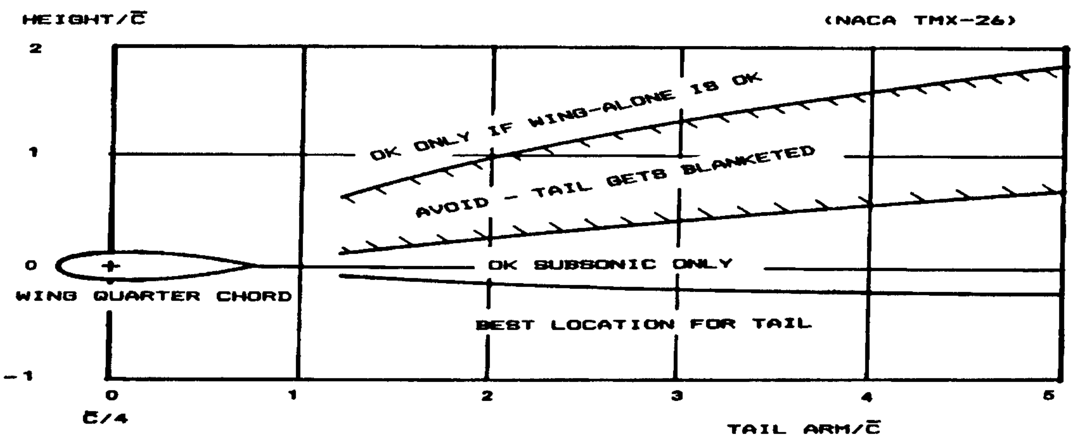
4.3.2. *Horizontal Tailplane Shape and Geometry*

Aspect ratio is of direct influence because of its effect on the lift-curve slope. Tailplane taper has a slightly favorable influence on the aerodynamic characteristics. A moderate taper was chosen to save structural weight. Positive sweepback is occasionally used on low-speed aircraft to increase the tail plane moment arm and the stalling angle of attack, although the result is a decrease in the lift-curve slope. Up to about 25 degrees of sweepback there is still an advantage. We have chosen ~16 degrees sweepback for horizontal tail surfaces.

The position of the tailplane relative to the propeller slipstream may make it desirable to shift it slightly in an upward direction. This may be achieved by using a certain degree of dihedral. But we have ignored this effect since the speed of our aircraft would not be very high.

The objective of preliminary tailplane design is not only to arrive at an acceptable guess with regard to shape and size, but also to ensure a good balance between the wing locations in longitudinal direction, the CG required, the CG range available and the tailplane design.

Fig 4.4. Horizontal location of horizontal tail surfaces



4.3.3. *Design of the Vertical Tailplane*

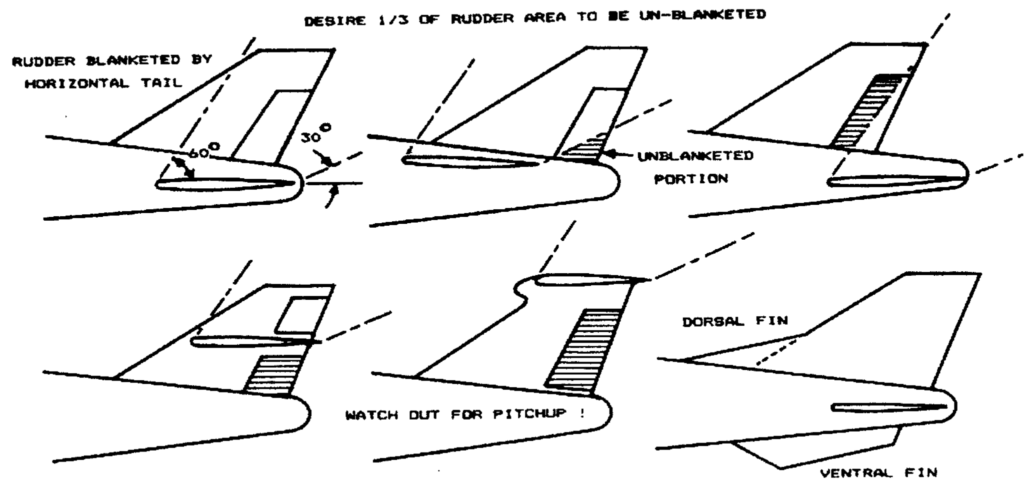
The design of the vertical tailplane is more complicated than that of the horizontal tailplane. It is generally quite difficult to calculate the lateral-directional aerodynamic characteristics, since they are closely connected with a complicated asymmetrical flow field behind the wing/fuselage combination which meets the oncoming air at an angle of sideslip.

4.3.3.1. *The Shielding Effect*

The shielding effect of the horizontal tail is responsible for a dead air region over most of the vertical tail. As a rule, it can be said that at least one-third of the rudder area should remain outside this wake, assuming that for an angle of attack of 45 degrees the forward and aft boundaries of the wake form an angle of 60° and 30° respectively with the horizontal tail. In addition, there should be a substantial amount of fixed area beneath the

horizontal tail to provide damping of the spinning motion. In our aircraft almost half of the rudder area would be out of wake.

Fig 4.5. Shielding effect



4.3.3.2. *Vertical Tailplane Shape and Geometry*

A high aspect ratio tailplane is effective at small angles of attack of side slip, but it has a small stalling angle of attack. But we did not prefer a high aspect ratio because of not having enough control area at vertical tail. The aspect ratio of vertical tail is about 1.8. The tailplane contribution to lateral stability is only slightly affected by taper, which is applied mainly to reduce weight and to increase rigidity of the fin. Taper ratio for vertical tail is 0.67. Design considerations about angle of sweep are similar to those for horizontal tailplane. A 10 degrees of sweep was chosen for vertical tail. The usual airfoil sections for a vertical tailplane are symmetrical. So we did.

4.4. *LANDING GEAR*

In this phase we discussed which material we can use. Which material to use for manufacturing gains great importance for solid spring type landing gear. The structural elements, in this case, should be elastic enough to damp the impact shock and should be rigid enough to carry the loads that would occur during landing. There seems to be two possible material options. They are various types of composite materials whose elasticity and yield strength can be adjusted and metal alloys whose elasticity and yield strength can be changed by physical processes. The greatest advantage that the composite materials have to offer is the adjustability of elasticity and yield strength. Another point that the composites are advantageous over metal alloys is that they are definitely lighter. But the cost of manufacturing the landing gear with composites that would give the desired elasticity and strength would be quite expensive. When this aspect is considered, metal alloys stand aside as a more rational choice. When we evaluated the metals according to their cost, weight and shearing stresses, two metals drew our attention. One of them is steel, which is very cheap and has high yield strength. Steels give very good results at structural applications and they are cheap. Much lighter than steel yet nearly as strong, aluminium alloys are widely being used in today's vehicles, thanks to the advancements in material science. Even though aluminium alloys have low modulus of elasticity, they have high yield strength values. Negative

aspect of using steel for Solid Spring type landing gear would be its heaviness and also not having the low modulus of elasticity, thus the flexibility we would expect from a shock-damping component. For these reasons, aluminum alloys whose density and modulus of elasticity are approximately 1/3 those of steel, would be more appropriate for use in the manufacturing of the solid spring type landing gear.

4.5. *PROPULSION SYSTEM*

For propulsion system we considered SANYO and SR batteries. Because these batteries are the most available ones, that we could find in our city. Also we saw from the previous team that SR batteries have best properties among all batteries we could buy. We calculated (Current Capacity/Cell Weight) parameter in order to help us to chose the best suitable battery type. The results can be seen in Table 4.2.

Table 4.2. Battery Selection Table

Battery Type	Current Capacity (mAh)	Cell Weight (gr)	Internal Resistance MOhm	Available Cell #	(Curent Capacity/Cell Weight) mAh/gr	Decision
SR 1300 MAX	1300	40	<4	57	32,7	E
SR 1600 MAX	1600	51	<4	44	31,3	E
SR 2000 MAX	2000	53	<4	43	37,9	E
SR 2400 MAX	2400	54	<4	42	44,5	FA
SR 3000 MAX	3000	70	<4	32	42,8	E
SANYO N-1400SCR	1400	53	4	42	26,4	E
SANYO N-1800SCR	1800	56	4	40	32,1	E
SANYO N-1700SCR	1700	54	4	42	31,5	E
SANYO N-1700SCK	1700	56	4,1	40	30,4	E
SANYO N-1900CR	1900	54	4	42	35,2	E
SANYO N-2000 CK	2000	80	4,1	48	25	E

We didn't investigated electrics motors a lot. But all the motors that we searched doesn't posses better properties compared to Cobalt 90 10T#22 which is our motor. We prepared a motor selection table (Table 4.3) for choosing most efficient motor we could find. We saw from this table our motor is the most efficient one. We didn't considered to use twin motors in conceptual design phase. But we searched Cobalt 40 8T#20 motor to see if there would be a clear advantage of using twin motors.

	SINGLE MOTOR			
Motor Types	Efficiency @ 36 Cells&40Amp	Torque (N.m)	Motor Constant (Kt/Ro^2)	Decision
Cobalt 90 10T#22 Kt:5,35 Ro:0,111 Lo.3	%81,4	1,5	434	FA
Cobalt 90 11T#23 Kt:5,89 Ro.0,55 Lo:2,5	%40	1,65	19,5	E
Cobalt 60 10T#23 Kt:4,26 Ro:0,15 Lo.2,5	%78,5	1,3	205	E

	TWIN MOTOR			
Motor Types	Efficiency @ 18 Cells for ea. & 40 Amp	Torque (total) (N.m)	Motor Constant	DECISION
Cobalt 40 8T#20 Kt:1,99 Ro.0,121 Lo.2,0	%70,5	1,37	136	E

The electric powered engine is put into a block made of composite material. This block would be assembled to the forepart of the fuselage. Upper part of the block is made removable in order to put the engine and other components easily. For preventing the deformation of the foam by the engine heat, a wood block that will hold motor would be placed inner part of the composite block. At the back part of the fuselage there is a cone where the tail boom was thought to be attached.

PRELIMINARY DESIGN RESULTS

WING AREA	10.76ft^2
WING ASPECT RATIO	8.5
WING PLANFORM	Double tapered
WING AREA	
WING SPAN	9.613ft
WING TAPER RATIO	0.425
MAX CHORD	1.444ft
MIN CHORD	0.59
AREA OF FLAPERONS	0.2 X WING AREA
Clmax	0.9
(Clmax)flapped	1.8
MAX DEFLECTION ANGLE OF FLAPERONS	20 degrees
FUSELAGE	Softball loading: 2 x 12 Batteries between softballs
EMPENAGE	Airfoil for vertical and horizontal surfaces: NACA0009 Horizontal: 16^0 sweepback Taper ratio: 0.72 Aspect ratio: 1.78 Vertical: AR: 1.8 Taper ratio: 0.67 10^0 sweepback
LANDING GEAR	Attached below wing
PROPULSION SYSTEM-BATTERIES	SR 2400 MAX- 36 cells
PROPULSION SYSTEM-MOTOR	Cobalt 90 10T#22

5. DETAIL DESIGN

5.1. *WING*

In the detail design phase of the wing, the precise shape and measurements of the wing planform that would meet the performance parameters such as wing loading and T/W ratio were aimed to be calculated.

Performance equations require 10.76 ft^2 wing area of the E193MOD airfoil for 98 ft/s level flight at (L/D)max conditions. It was understood that it was not necessary to use control surfaces like flaps for take-off and landing. Yet when the changing trend of the wind speed about the competition site in the recent years was considered also, we decided on using flaperon (rather than using flaps and ailerons separately), which would be %20 of the reference wing area for a safer design. This conclusion also decreases the contribution of wing to RAC in respect of using flaps and ailerons separately.

After the meeting with the production team, it was understood that the production of a triple tapered planform was far more difficult than a double tapered planform. More strengthening had to be done on a triple tapered planform that should be assembled by mounting the many pieces of the wing to each other. Because of this negative aspect, the triple tapered wing would become heavier; thus it was eliminated. Even though its lift distribution deviation was slightly more than the triple tapered planform, the double tapered planform was very easy to manufacture.

The next problem was to choose one of the double tapered planforms. Taking wing area as 10.76 ft^2 and aspect ratio as 8.5, we compared the lift distributions of the double tapered planform to the elliptical force distiribution. We calculated Rated Aircraft Costs for the best results of the comparison. The conditions at which the double tapered planforms give close to elliptical lift are given below in Fig 5.1.a, b, c, d and e.

WING #	C1/C0	C2/C0	C3/C0	a/s	b/s
WING 1	1	0.8	0.375	0.4	0.5
WING 2	1	0.9	0.4	0.4	0.6
WING 3	1	0.5	0.425	0.35	0.6
WING 4	1	0.8	0.2	0.35	0.7
WING 5	1	0.7	0.425	0.35	0.7

Table 5.1. Geometry input values for wings in Fig 5.1.a, b, c, d and e

C1/C0: The ratio of chord length of the first break along the half span to root chord length

C2/C0: The ratio of chord length of the second break along the half span to the root chord length

C3/C0: The ratio of tip chord length to the root chord length

a/s: The ratio of untapered part's length along half span to half span

b/s: The ratio of first tapered (inner one) part's length along half span to half span

Fig 5.1.a, b, c, d and e

The last one in Fig 5.1 was observed to possess elliptical force distribution along with a low Rated Aircraft Cost that best suited us. So we decided to produce a wing with such measurements and planform.

We did the structural analysis for a material that we would use in manufacturing. The results of these analyses can be seen below in Fig 5.2.a and b.

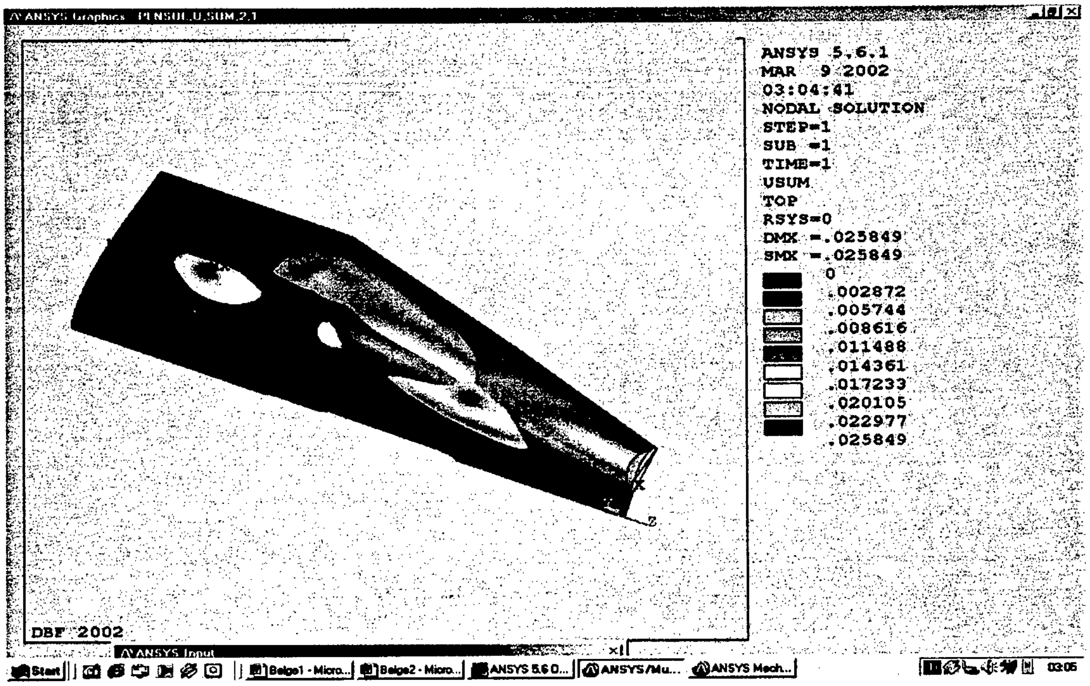


Fig 5.2.a Structural Analysis Result, total displacement on the wing if it is forced to the limits.

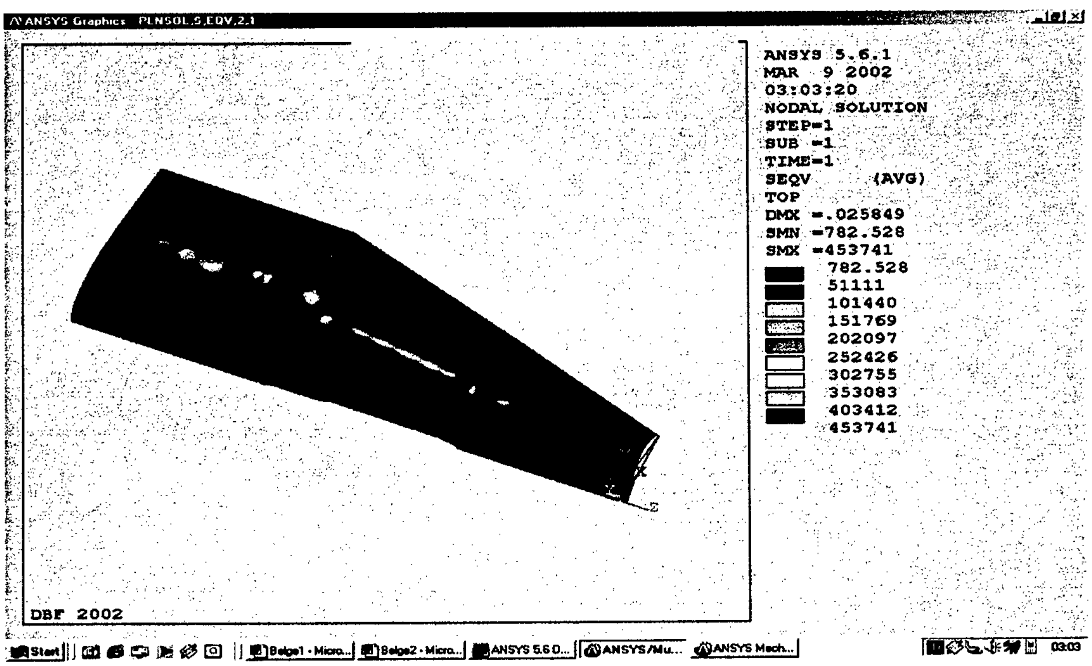


Fig 5.2.b Stress distribution on wing according Von misses

Depending on the structural analysis results, the parts with maximum stress would be strengthened with balsa and double layer of Epoxy-graphite composite.

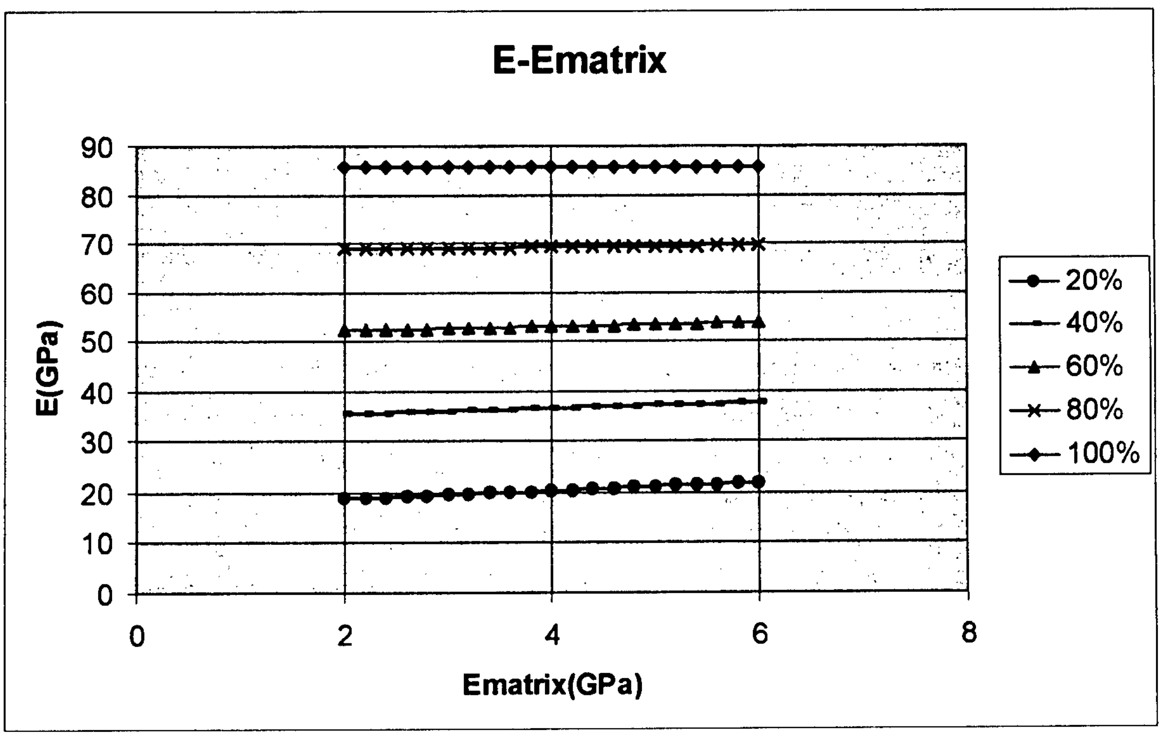
5.2.*FUSELAGE*

After deciding the entire configuration and completing the preliminary design phase, in the detail design part we examined the case more closely. In this phase the most important problems for the fuselage were centre of gravity problem and the structure of the composite material that will be used in the fuselage.

The aircraft industry has developed several technologies in order to increase the performance of an aircraft. The composite materials, which are defined as "heterogeneous mixtures of two or more homogeneous phases which have been bonded together", are of the products on this process.

As an important material in aircraft technology, different types of composites have been used in aircraft components until now. Among several choices, we have decided to make our model from composite materials. In order to form a composite material, firstly the types of matrix and the fiber had to be chosen. In the fiber area, there are several different types, such as glass-fiber, carbon fiber etc. as a result of discussions the glass-fiber won the race. In spite of their low performance than graphite and other carbon fibers, its cost and also its weight is less than the others. Among three types of glass-fibers the S-type is the most frequently used one in aircraft industry, because of its high tensile modulus (85,5 GPa) and high tensile strength (4590 MPa). In the matrix field, with its good mechanical properties and dimensional stability epoxy won the race. Besides to these properties epoxy resins are easily processed and their cost is less than the others.

When the composite material is produced, it is not possible to mix the matrix and the fibers in the exact ratio. In order to decrease the uncertainties in the material property, during the production phase the percentage of the matrix is made between 40-60 %. And the Young Modulus of the new material is get from the graphic.



5.3. *EMPENAGE*

Detailed design of tail surfaces depends on specific requirements, which are to maintain stability during steady flight, damping of encountered disturbances and to provide enough force for maneuvers.

In the detail design phase of empenage we used the preliminary design results. That is, we have the tail surfaces area values. These areas had been calculated according to the maximum available (harmonize with contest rules) and fitting tail surfaces. According to declared Cost model the tail surface areas are not important, but every feet of the body length increases our RAC. So we strove to make the shortest tail boom. In horizontal tail we would make 50 % of the total horizontal tail surface as elevator. In vertical tail the rudder area will be approximately half of the total vertical tail surface area.

To calculate the tail arms of vertical tail and horizontal tail we used the tail volume ratios. These tail volume ratios were in accordance with the recommended values for R/C's. The value of horizontal tail volume ratio, Vht, that is recommended for R/C whom wings bottom side has a little camber, is 0.58. The vertical tail volume ratio, Vvt, is 0.04.

$Vht=(Lht \times Sht)/(\bar{c} \times S)$ $\quad\quad$ $Vvt=(Lvt \times Svt)/(b \times S)$

Lht: horizontal distance between C.G. of the airplane and the aerodynamic center of the horizontal tail

Lvt: horizontal distance between the C.G. of the airplane and the aerodynamic center of the vertical tail

Sht: planform area of the horizontal tail (1.506 ft^2)

Svt: planform area of the vertical tail (0.925 ft^2)

$\bar{c}$: Mean aerodynamic chord of the wing (1.214 ft)

b: wingspan (9.613 ft)

S: wing planform area (10.76 ft^2)

The usual tailplane design process is iterative, in view of the fact that the moment arm cannot be determined accurately before the complete layout of the tailplane has been chosen and weight of each component has been found. But we have some statistical knowledge about the approximate weight of each component. By using these components weights we prepared an Excel sheet. In this sheet we considered only longitudinal C.G. of components, so of aircraft. Also we assumed that the wing's C.G. is at 1/3 chord length from leading edge. This sheet was shown as Table 5.2. In this table calculations were done in SI units. Then the results, C.G.'s, are converted to feet.

COMPONENT	Weight(kg)	X(mm)	m(x) (kgmm)
Wing	3	770	2310
Fuselage	1	800	800
Softballs	4,44	800	3552
Tail surfaces with servos	0,5	2400	1200
Landing gear	1	700	700
Motor	1	100	100
Batteries	2,3	600	1380
Servos on wing	0,1	920	92
Control Batteries	0,3	1450	435
Propeller	0,15	0	0
Tail cone	0,2	1550	310
Motor block	0,4	100	40
TOTAL LOADED WEIGHT	14,39		10919
EMPTY C.G.(mm)		740,402	
LOADED C.G.(mm)		758,7908	
Empty C.G. (ft)		2,428519	
Loaded C.G. (ft)		2,488834	

Table 5.2. C.G. calculation table

By using these values we found horizontal and vertical tails' moment arms: Lht=5.03 ft, Lvt=4.47 ft. There would be no disadvantageous effect to put the vertical a bit aft than the obtained value, Lvt=4.47. So we will do, and will fix it on the horizontal tail.

5.4. *LANDING GEAR*

After we have decided to use taildrager landing gear by aliminium alloys we analyzed it by Ansys. In this analyse we set geometry so that it would suit the propeller with shortest length to have the least weight. In Fig 5.3.a and b landing gear's geometry can be seen;

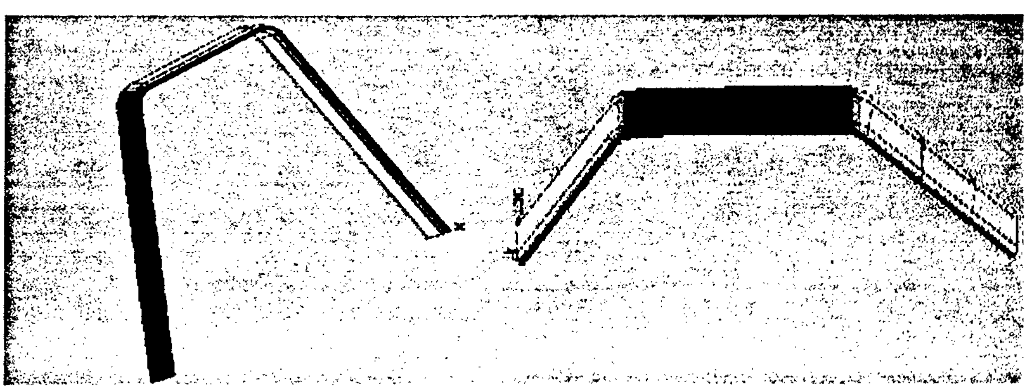


Fig 5.3.a. and b show isometric and top view of landing gear

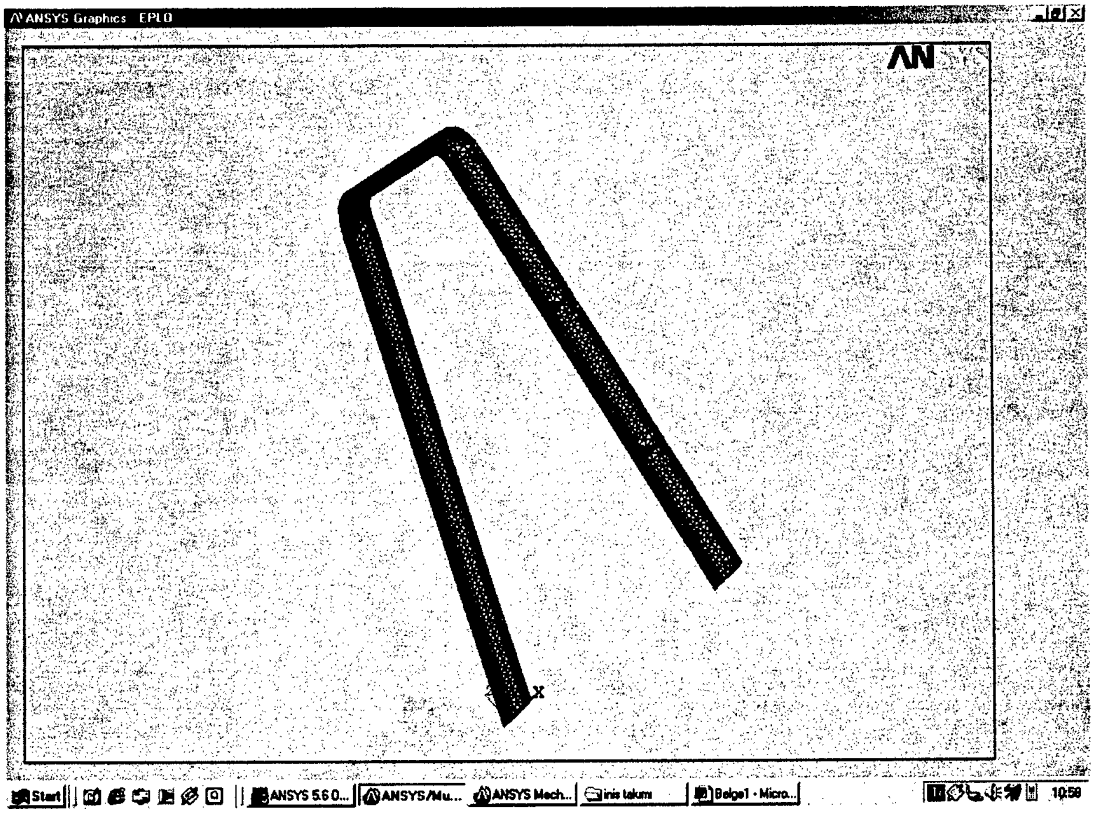


Fig. 5.4. Meshing process for landing gear

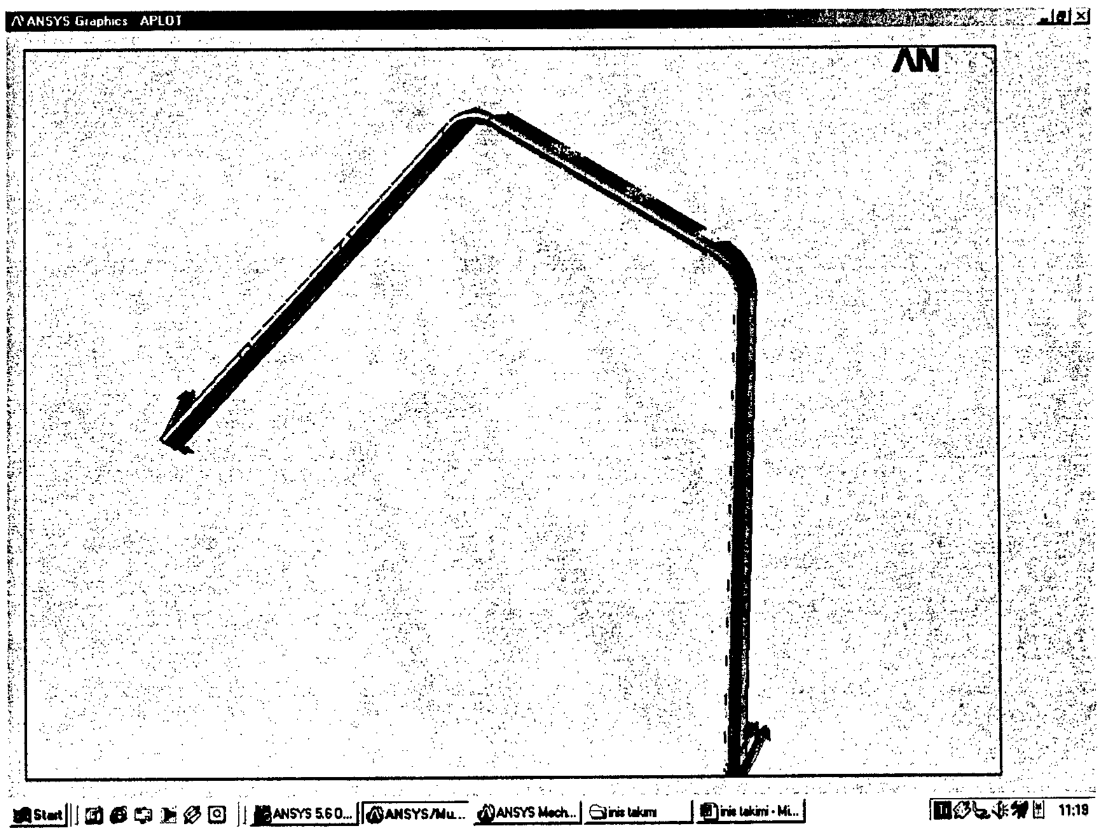


Fig.5.5. The possible forces that will be applied to the landing gear

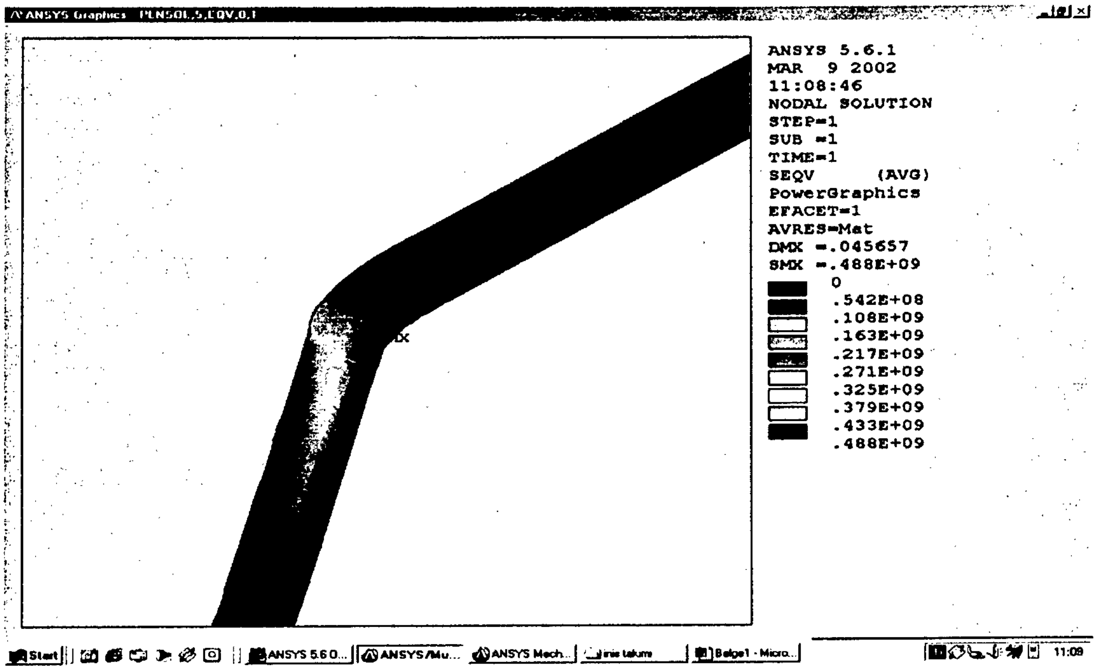


Fig. 5.6. The critical section found according to stress distribution of Von Misses

From Fig.5.6. it can be seen that the most stress occurs at the filleted parts of the landing gear. So in manufacturing process these parts would be strengthened. Landing gear will be attached to the wing, since it should be close to the CG.

5.5. Rated Aircraft Cost Estimation Table

Coef.	Description	Value	ATA-4
A	Manufacturers Empty Weight Multiplier	$100	
B	Rated Engine Power Multiplayer	$1500	
C	Manufacturing Cost Multiplier	$20/hour	
MEW	Manufacturers Empty Weight		22.050
REP	Rated Engine Power	(1+0.25*(#engines-1))*Total Battery Weight	
MFHR	Manufacturing Man Hours	MFHR=$\sum$ WBS hours	
	WBS 1.0 Wing(s)	8 hr/ft. Wing Span	9,613
		8 hr/ft. Max Wing Chord	1,444
		3 hr/control surface	2
	WBS 2.0 Fuselage	10 hr/body max length	8,075
	WBS 3.0 Empenage	5 hr/Vertical surface with no active control	1
		10 hr/Vertical surface with an active control	1
		10 hr/Horizontal surface	1
		15 hr/"V" tail	0
	WBS 4.0 Flight systems	5 hr/servo or motor controller	5
	WBS 5.0 Propulsion System	5 hr/engine	1
TBW	Total Battery Weight		4.880
		5 hr/ propeller or afn	1
RAC	Rated Aircraft Cost,$ (thousands)	(A*MEW+B*REP+C*MFHR)/1000	14.130

5.5. *DRAWING PACKAGE*

In all drawings dimensions are in SI units (mm).

Fig.5.7. Isometric view

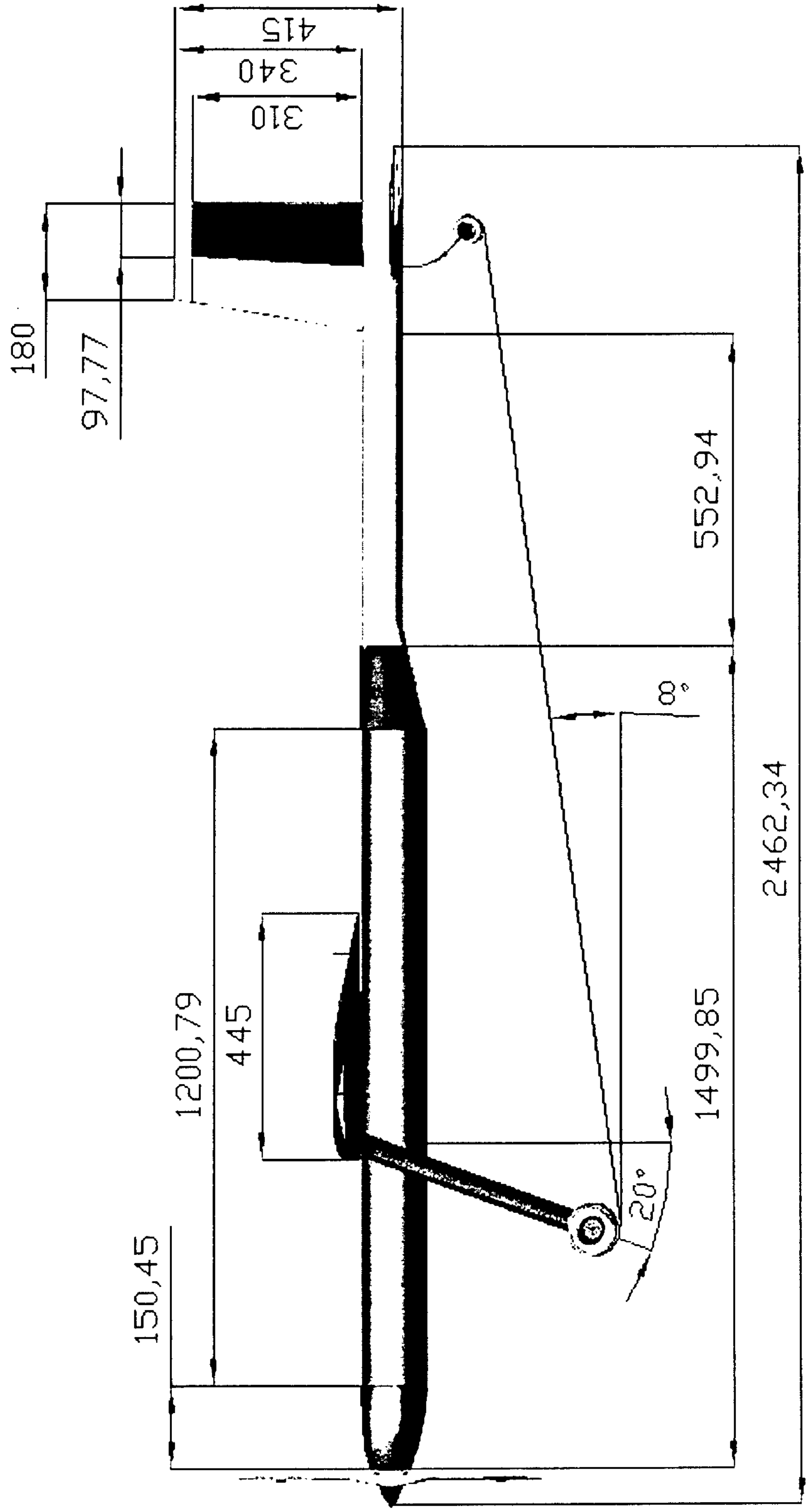


Fig.5.8. Side view

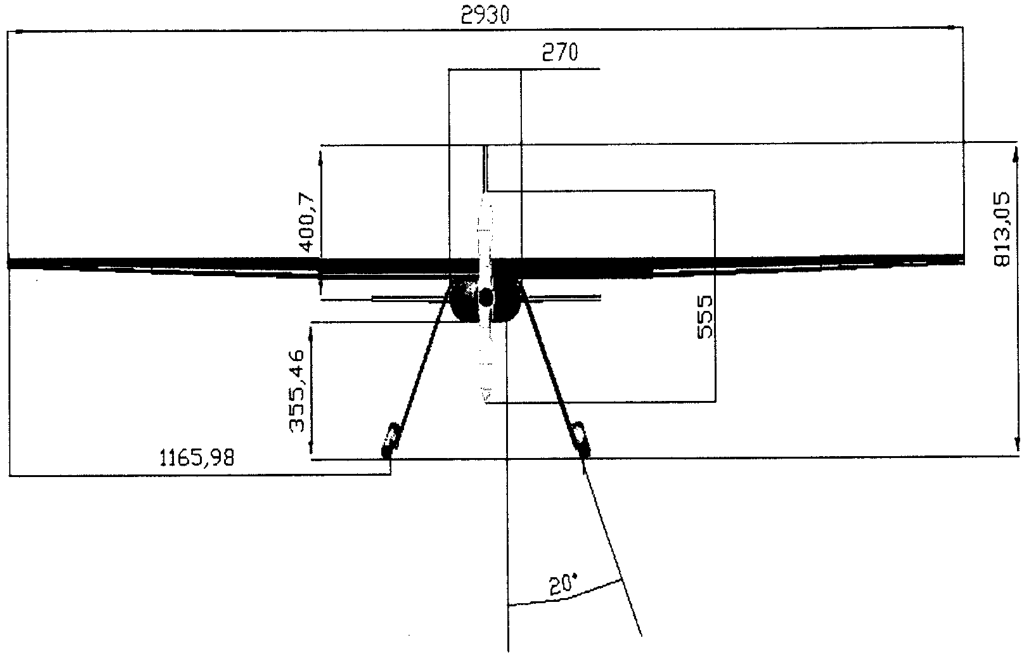


Fig.5.9. Front view

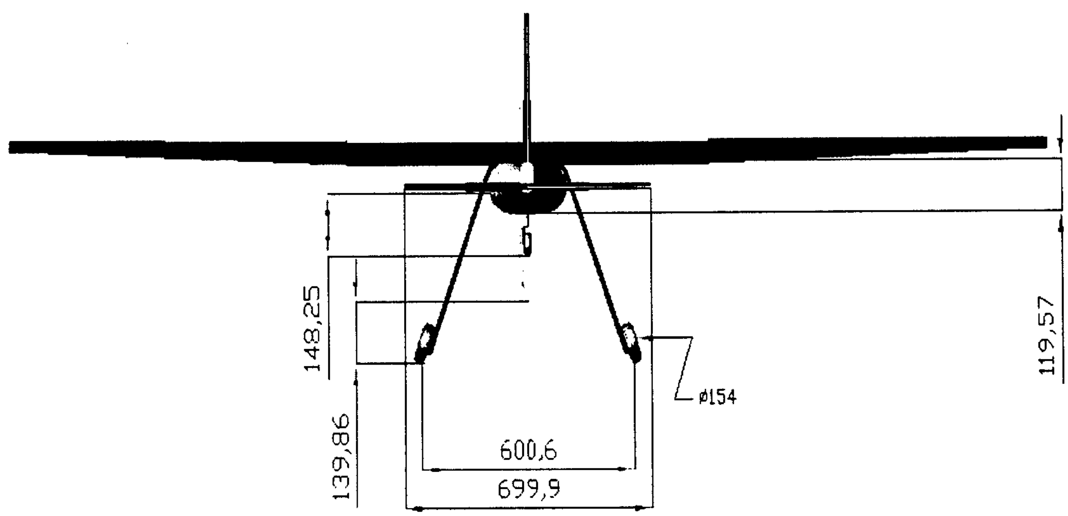


Fig.5.10. Rear view

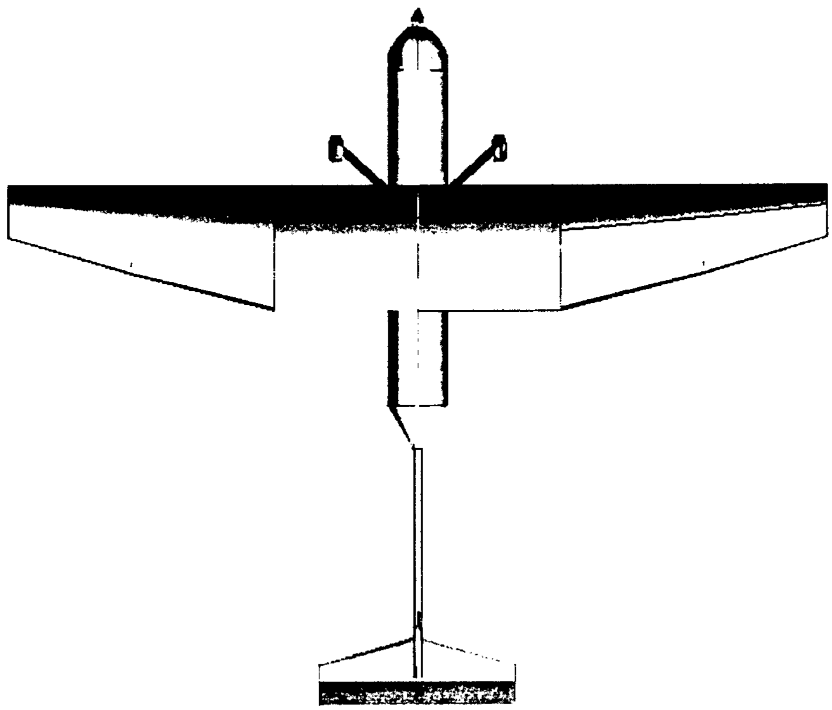


Fig.5.11. Plan view

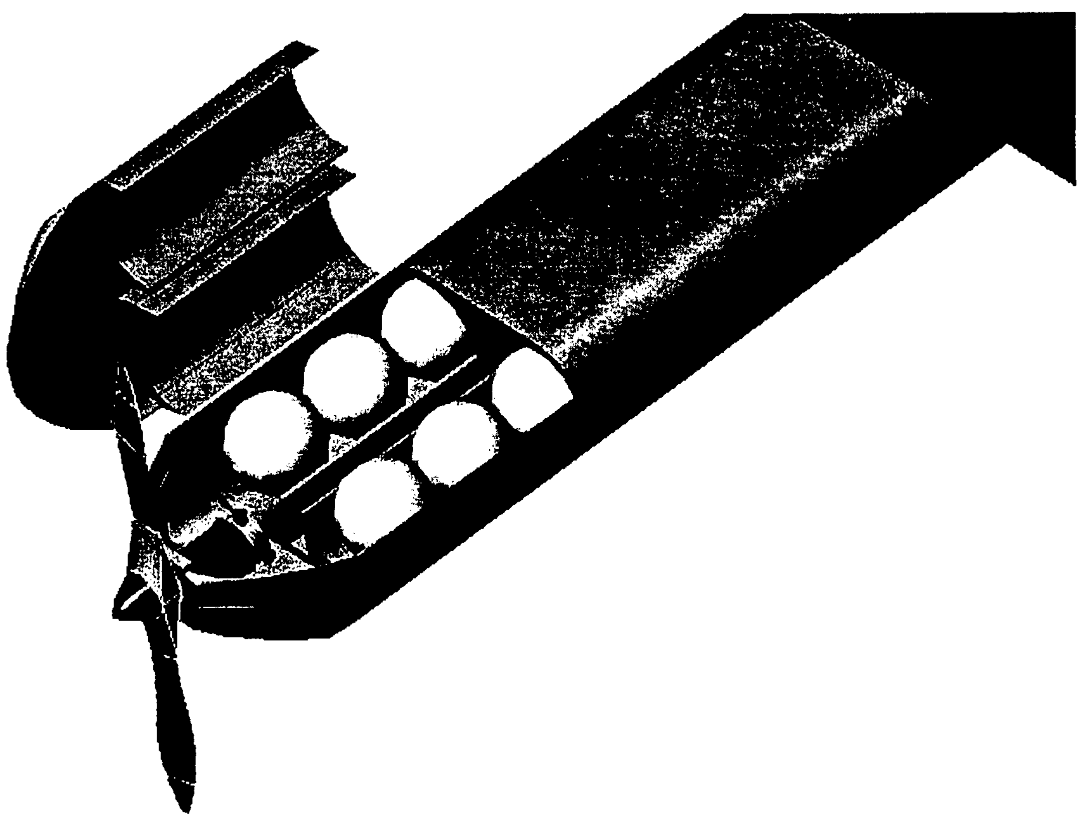


Fig.5.12. Loading-unloading gate

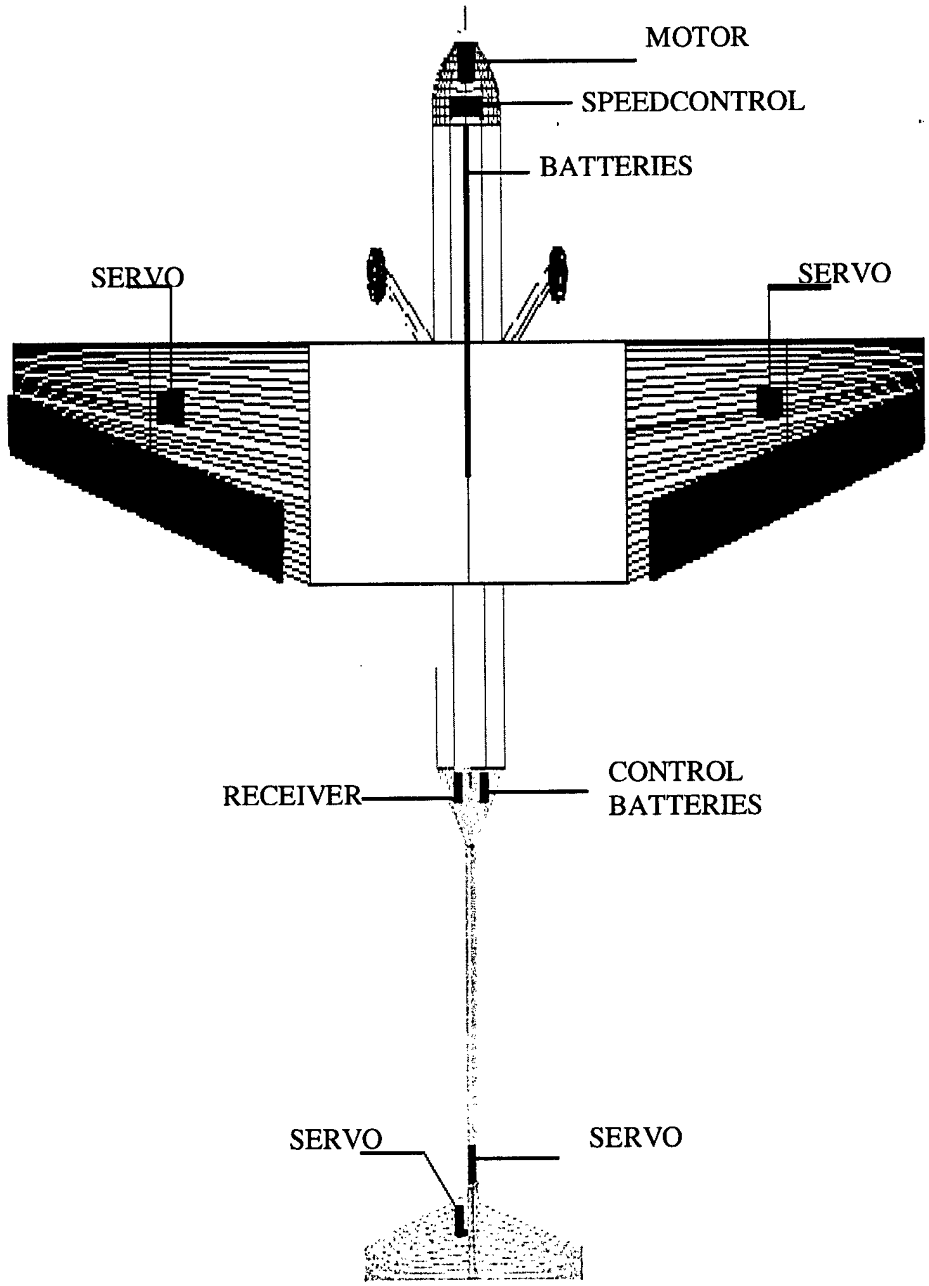


Fig.5.13. Control system and propulsion system elements

6. MANUFACTURING PLAN AND PROCESSES

6.1. *Process and Material Selection*

While investigating the manufacturing processes, the main aspect we took into consideration was our experience about cost and process. The reason we considered the cost was because we had a limited decent budget and it was hard to get sponsorships during economical depression in our country. Since time was limited, we did not have any chance to learn different process techniques and even if we had the time there was nobody to teach us different skills. We only investigated two processes for manufacturing: Wooden material processing and composite material processing. Exactly one person was experienced in wood/balsa working and another person had simple background knowledge. Wood would bring a heavy burden on our budget whereas composite material would cost relatively cheaper because it would be supplied for free by the sponsoring companies. Thanks to the last year's ITU participants, we gained quite an experience in composite material manufacturing. Two juniors of the last year's team taught everything they knew about composite processing to the manufacturing subgroup members. These members were ready for manufacturing after two weeks of theoretical and practical training. The other consideration about material and process selection can be seen in Table 6.1. In this table if a consideration is favor of us it takes high grades, if not it takes low grades. All these happenings caused us to choose composite material for manufacturing.

Consideration	Wooden Material Processing	Composite Material Processing
Cost	1	4
Experience	1	5
Weight	3	4
Strength	3	5
Manufacturability	2	5
Availability	2	5
Reparability	1	4
Reliability	1	5
Flexibility	1	4
Time Required	1	4
RESULT	Limited Usage	General Usage

Table6.1. Material and Process Selection Table

6.2. *Materials for Composite Processing*

6.2.1. *Cloth*

Unidirectional, or uni for short, has most of the fibers in one direction. Uni is very, very strong in the direction the fibers run. Properly laid up, it is about 25 percent stronger than an equivalent cross section of 4130 steel; however, it has no crosswise strength because the fill yarns are only meant to hold the cloth together

during handling. Uni is used in places like spars caps, where the we know the load will be applied in only one direction.

Bidirectional cloth is not as strong as uni, but can take the strain in both directions; it rarely is as strong in both; one direction is usually weaker. It will be used for covering of wing, fuselage and empennage.

6.2.2. *Foam*

Various types of foam can be used in composite aircraft. Manufacturers use it to make lighter, stronger kit components. The advantage of foam is its light weight and easy workability. Most can be cut with ordinary knives or a heated wire and shaped with a variety of inexpensives tools. Foam parts can be easily made and will be very light, but can't take any sort of load. The usual practise is to make a component from foam, then cover it with fiberglass. We will use Styrofoam.

6.2.3. *Fasteners*

The primary fastener will be the epoxy resin used to impregnate the fiberglass cloth and bond aircraft components together. Epoxies are two part systems, consisting of resin and the hardener. Mixing ratios and curing times, temperatures are important parameters.

6.2.4. *Peel Ply*

Strips of dacron cloth, called peel ply, have several uses in fiberglass work. Peel ply doesn't bond structurally with the fiberglass. It can be applied, wetted out with resin, and left to cure. Even when the cure is complete, the peel ply (and the resin atop and within it) can be removed with a simple tug. The top layer of fiberglass is left with a uniformly rough surface. Its major use is to ensure a good bond with a layup to be applied later. Peel ply is also good for holding resin in contact with cloth in awkward positions. For instance, a sharp corner might allow resin to flow downhill, starving the bond area. A piece of peel ply laid on the starved area will keep the resin in place. It also takes excess resin of fiberglass so makes it lighter.

6.2.5. *Hot-wire Cutting*

With styrofoam we can use the hot-wire method. An electrical current is passed through a fine wire, which heats the wire past the melting point of the styrofoam. When cutting a shape, two templates are tacked to the ends of a piece of foam. We turn on the cutter, and sink the wire into the foam until it contacts the templates on both ends. Then, we slowly move the wire around the outline, taking care that each end of the wire is at the same position on the template at the same time. We let the wire cut at its own pace; if it is pulled too hard, the wire will lag excessively and bow inside of the foam, distoring the final shape. When done, we are smoothing the foam with files or a sanding block. Fig 6.1. shows hot-wire instrument, and Fig 6.2. shows hot,wire cutting.

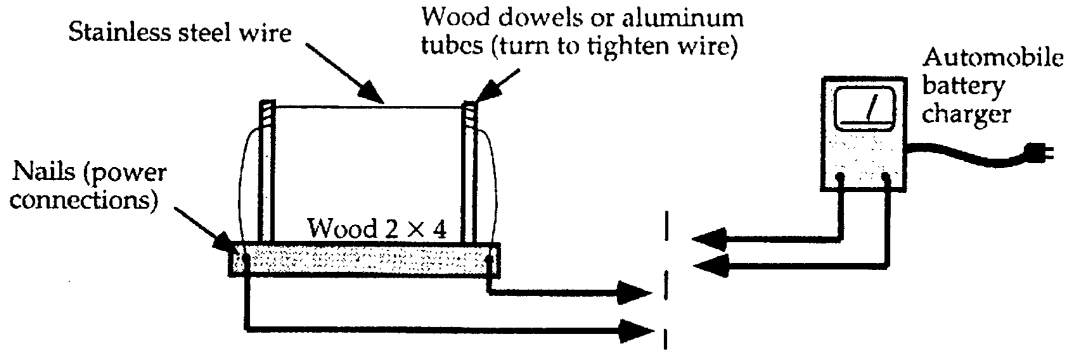


Fig 6.1. Hot wire instrument

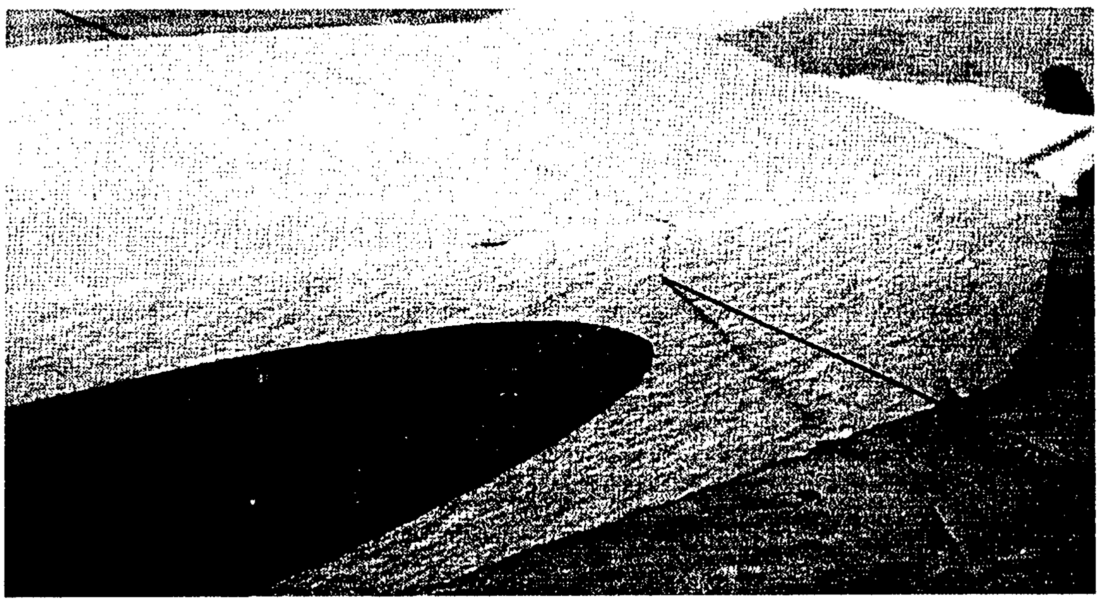


Fig 6.2. Hot-wire cutting

6.2.6. *Layup Operations*

When it becomes time to make a layup, we have one overriding goal: Saturate the fiberglass cloth with the right amount of resin-too little, and air bubbles remain in the laminate. Not only do they reduce the strength, they might allow water to seep between layers of fiberglass cloth. If this water freezes, it will cause delamination (seperation of the layups) and destroy the strength of the bond. Too much resin adds weight for no gain in strength. Too much in wet layup might actually cause the cloth to float, adding air bubbles as well as moving it out of position. The process is shown below in Fig 6.3.

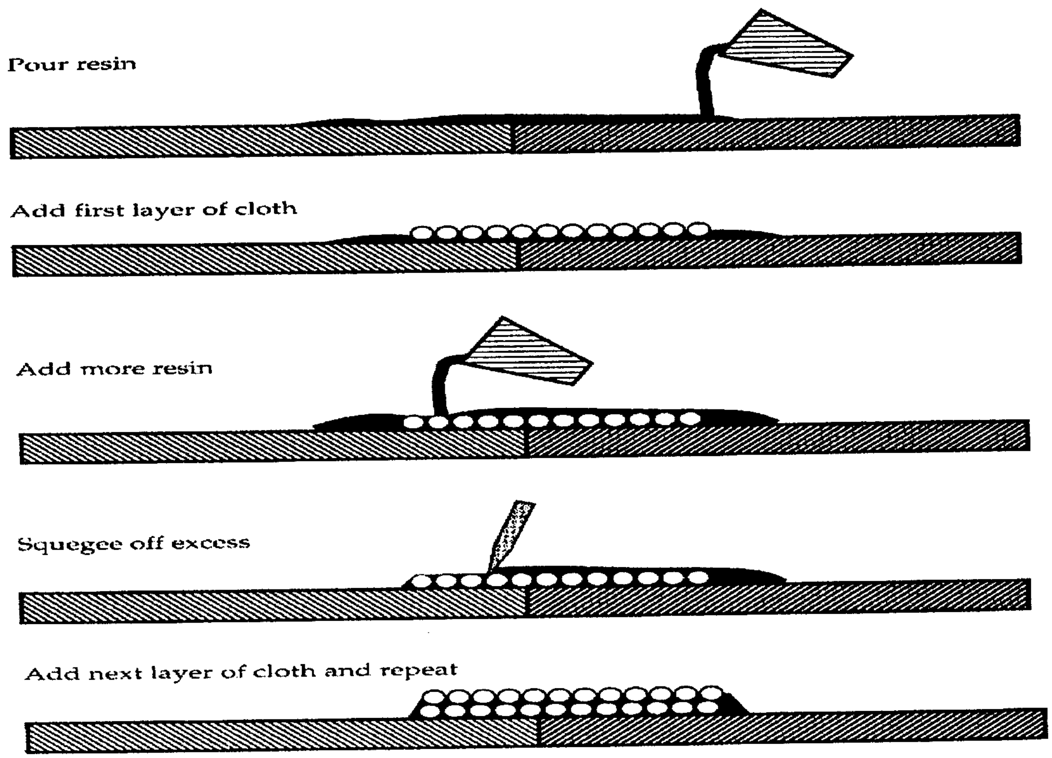


Fig 6.3. Layup operation

6.2.7. *Cutting the Cloth*

The direction makes a difference, strengthwise. The cloth is twice as strong when cut on a 45-degree bias. The bias orientation doesn't have to be exact; the effect of a slight error is negligible. Eyeballing is usually good enough. The final points: First, don't scrimp. The spsecified arae must be covered totally. Every fraction of an inch contributes to the total strength; if a cloth piece is too short, then the part is too weak.

6.2.8. *Finishing Fiberglass*

Peel ply is intended to produce the optimally rough surface for applying additional layers, not a smooth exterior finish.

6.2.9. *Manufacturing Plan*

We prepared a manufacturing plan that shows us the manufacturing process order. We are struggling to comply with dates. We succeeded in this aim to date. This plan can be seen from Table 6.2.

Table 6.2. MANUFACTURING PLAN

PROCESS	PLANNED DATE	FINISHED DATE
Exercises to learn exact process	Feb 01-14	Feb 14
Cutting wing parts	Feb 15-19	Feb 19
Cutting fuselage parts	Feb 19-22	Feb 22
Cutting motor block from foam	Feb 23	Feb 23
Attaching wing parts Strengthening wing parts	Feb 24-26	Feb 27
Covering the parts of fuselage	Feb 26- Mar 02	Mar 03
Preparing wooden motor block	Mar 03	Mar 03
Cutting tail surfaces from foam	Mar 04-05	Mar 05
Preparing fuselage- wing mounting places on wing and fuselage, servos places on wing	Mar 05-08	Mar 08
Covering wing with fiber-glass	Mar 08-13	
Covering tail surfaces	Mar 09-14	
Cutting flaperons parts from wing	Mar 14	
Cutting elevator and rudder parts from tail	Mar 15	
Preparing flaperon parts for assembly	Mar 15-17	
Preparing rudder and elevator for assembly	Mar 16-18	
Preparing tail boom	Mar 19	
Producing landing gear	Mar 20	
ASSEMBLY	Mar 21-27	

6.3. Manufacturing Major Components

6.3.1. *Wing Manufacturing*

Styrofoam, epoxy and glass-fiber were decided to be used as the main manufacturing materials of the wing. Carbon-fiber, balsa and beech would be used for strengthening the wing.

First of all, root, middle and end tip airfoil's templates were cut out of wood. Dots were spotted along the profile edges in order to make Styrofoam cutting easier and correctly. Foam as long as the middle constant chord part of the wing was cut out. Middle-wing profiles were stuck on both sides of this rectangular piece of foam. Hot wire, which is our foam-shaping instrument, was drawn through the foam starting from the leading edge and finishing the cutting process at the trailing edge of the profile. Cutting has been considerably easy because of the identical pair of profiles of the middle part of wing, whereas the long span of this piece caused flexibility problems in our hot wire while cutting. Adjusting the tension and flexibility of the wire solved this problem. The two different profiles of a tapered part were stuck to the opposite sides of another rectangular piece of foam. The most feared problem in this case would be wrong cutting and eventually wasting of material caused by the unevenly changing profile. To overcome this problem, the persons responsible for foam cutting should have

developed synchronized cutting skills by practicing frequently. The wing would be manufactured in five pieces along the wingspan: the constant chord middle part, inner and outer tapered wing sections on each side. These pieces would be mounted to each other with 5-minute epoxy after the cutting. Mounting sections are the critical points of the planform where the taper ratios change. We knew from the experiences of the last year's ITU group that, these mounting sections would be severely damaged if the aircraft was to crash. The same damage problem would be observed at the vertex regions of flaperons. We would be following an empirical method to overcome this serious problem. We would insert wooden beams into particular sections of the foam wing, which would make our flexible wing more rigid. At this stage holes would be opened in the foam for servos and foam would be dug out at certain places to insert wooden beams for the fuselage and landing gear fittings. After the signal cables for the servos and the wooden sticks are placed in the openings, the wing would be covered with fiber-glass 0.4422 oz/sq-ft and epoxy composite. Long carbon-fiber stripes would be used to increase the strength of the wing in accordance with load distribution. After the composite covering is done, flaperons are cut out of the wing and the inner section walls are covered by sticking balsa plates. The reason we would use balsa in the flaperon nest is for maintaining wing-flaperon connection and to facilitate flaperon motion. The last step during the manufacturing of the wing would be to place the servos and actuators in the authorized holes and make their connections with the flaperons.

6.3.2. *Fuselage Manufacturing*

Same kind of materials as in wing manufacturing would be used during fuselage manufacturing. But this time instead of Styrofoam, P-15 foam would be used because P-15 is %100 less dense than Styrofoam. Styrofoam is denser, thus heavier, but provides characteristics such as stiffness and strength, which we would desire for the wing. The fuselage does not strictly require high values of stiffness and strength for the foam. Foam cutting process is generally the same for all components of the aircraft. Suitable fuselage profiles are prepared and stuck to both ends of a suitable piece of P-15 foam. Hot wire is drawn around the fuselage profiles through the foam to produce the desired component. The body would be manufactured as top and bottom pieces. The inside walls of these fuselage sections would be covered with epoxy fiber-glass 0.7207 oz/sq-ft composite. As the strength of the thin-walled fuselage increases because of the composite material, we would be given the chance to comfortably work on the outside surface. The most important action on the fuselage surface would be to strengthen the fuselage-wing fittings. Other connection elements such as wooden beams would be mounted to the wing-fuselage and tail-fuselage mounting regions. After the engine block and servo-receiver cables are arranged and placed in appropriate place, the two pieces of the fuselage would be stuck to each other and the whole fuselage would be covered with two layers of epoxy glass-fiber (0.4422 oz/sq-ft). Carbon epoxy composite could also be applied if there was need.

6.3.3. *Empennage Manufacturing*

The empennage manufacturing would be done by the same methods. Styrofoam, epoxy, fiber-glass and balsa are the materials we will use for this component. Also we will use booms to extend the tail arm. This boom will be stucked to a cone which will be fixed at the end of fuselage.

6.3.4. *Landing Gear*

Landing gear would be manufactured from hollow pipe aluminum. This component will be fixed below wing. Wooden blocks would be used to fix landing gear to wing. Also some strengthening materials would be used in wing and on upper face of fuselage.

6.3.5. *Propulsion System*

Our aircraft propulsion system is shown below. The battery packs will be in fuselage between softballs. Motor will be placed in the front part of fuselage by a wooden motor block inserted in foam. We will also use a 1:2.7 gearbox and a two-blade propeller. The propeller has a size of 22x14. The most important thing according to our propulsion system subgroup is to achieve the battery- motor connection by using the shortest cable. So we can use the battery packs energy efficiently during flights. Fig 6.4. shows propulsion system installation.

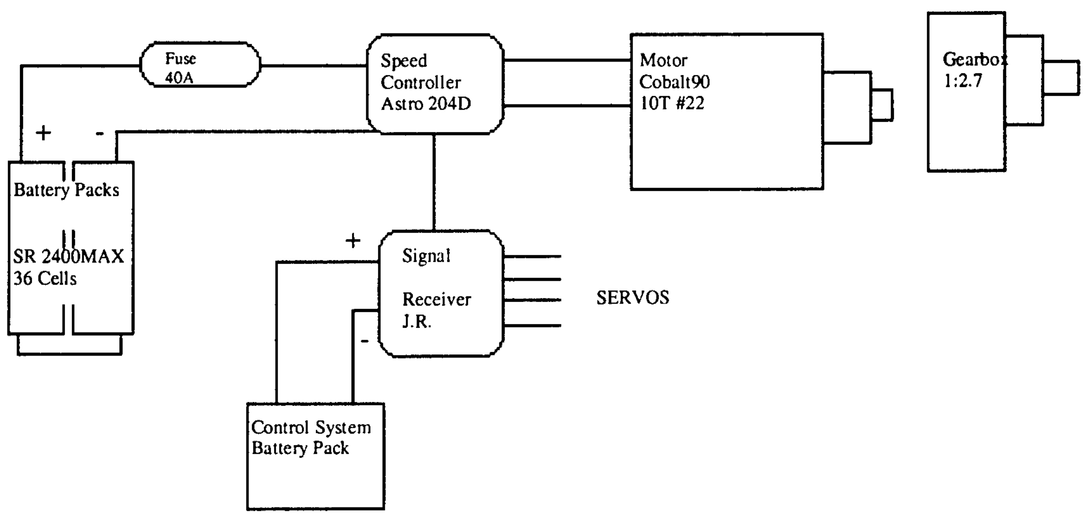


Fig 6.4. Propulsion system

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AIAA DESIGN/BUILD/FLY COMPETITION

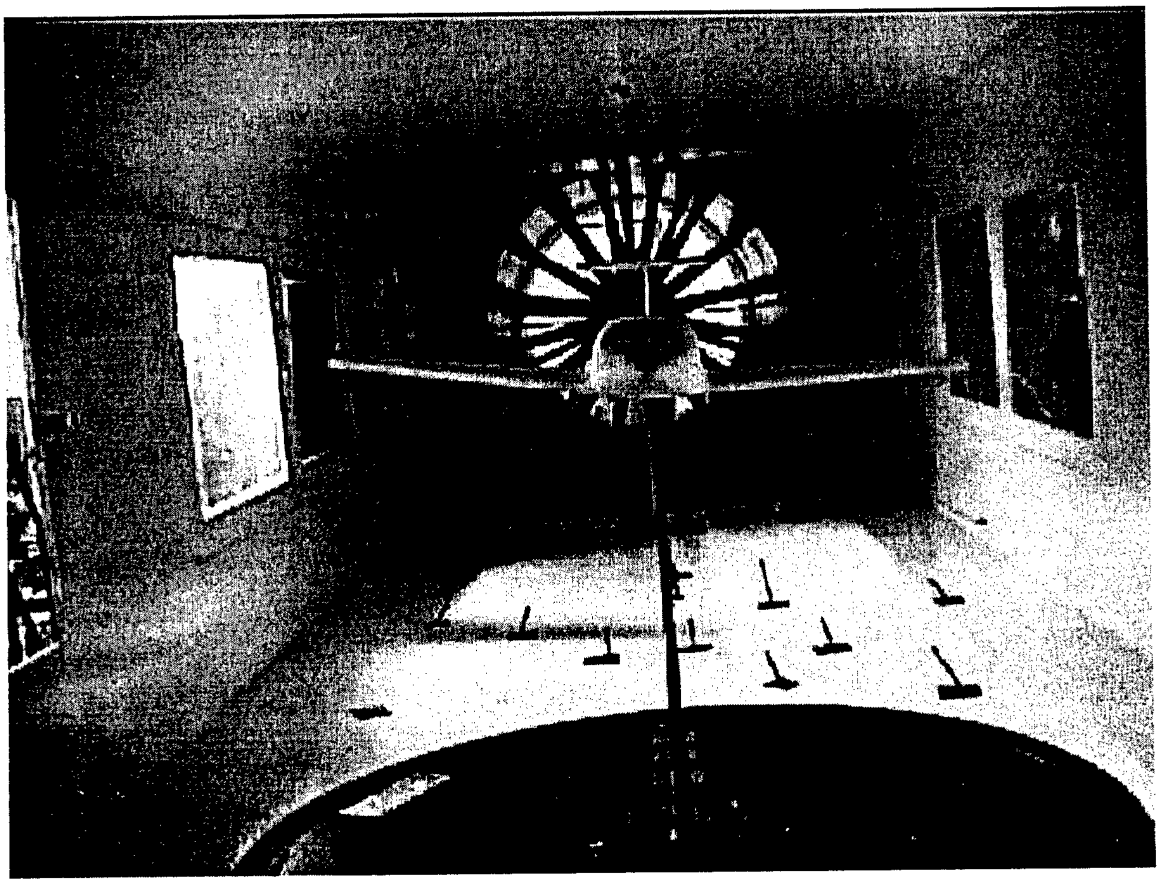

Table of Contents

1-Summary

The University of Arizona has chosen to compete in this years AIAA Design, Build, Fly competition. We have taken on the challenge of building an RC aircraft that will fly the required laps with and with out payload in the ten minute time allotted to the total flight. Keeping all of the requirements of the competition in mind, we have to design the best configuration that will produce the best plane. This configuration will be one that will allow us to have the best optimal flight with the fastest time. Not only did we have to consider the required limitations, but also the cost limitations. We had to look at the cost model and design around it in order to have the least hours, which would give a low cost, that would in turn increase our score. We would also have to choose a plane that would be capable and have the endurance to complete all three of the required missions within the ten-minute time limit.

In order to accomplish all of these goals, we chose to look at a few configurations. From there, we chose the one in which we thought would provide us with the best results. After a configuration was selected, we then broke the plane into its different parts (wing, tail, fuselage, landing gear, controls, and power) and analyzed each very carefully. We chose what we thought to be the best for each of these and began to build models. We then tested these models in a subsonic wind tunnel in order to test the performance of these choices. Using the wind tunnel and other calculations, such as the longitudinal stability and take off speed, we then began to construct our plane. From there came flight-testing and modifications in order to achieve a plane with the best possible performance.

2-Configuration Selection

The focus this year for both the University of Arizona DBF teams became weight and drag reduction. Looking at the speed, reducing weight and drag allows for more power available for speed. Looking at endurance, having a plane that weighs less means sufficient speed can be attained using less battery power, and our plane can therefore stay in the air longer. Therefore, in the selection of the configuration for our plane weight and drag became the dominant features.

After introducing some ideas, four basic configurations were selected and each team member researched one of them. The results were discussed at a team meeting, and a configuration was selected. The four configurations are shown in Figure 1.

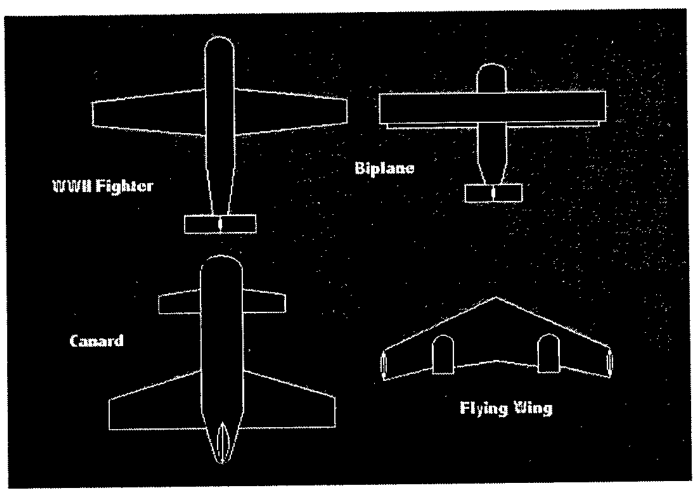

Figure 1: Researched Configurations

The following is a description of each configurations:

2.1-WWII Fighter:

This configuration represents the most conventional of the four studied. The WWII fighter presented a low weight and drag configuration, primarily due to its monowing design and sleek shape. The concept was originally thought of because a

fighter plane is made for high speed and performance. This made it more attractive than the biplane since removing one wing would reduce weight considerably and also reduce the frontal area profile and thus the drag.

This feature made it very attractive to our particular team since only one member had any experience with model aircraft construction, and that experience was limited to participation in the two previous AIAA planes. The more complicated configurations such as the canard and flying wing were determined as more suitable for teams with experienced members who knew the fundamentals necessary to design them correctly. Also, aerodynamic theory, particularly in regards to stability, applied more to this configuration than to the others

2.2-Biplane:

Even though this configuration suffers from higher weight and drag due to the extra wing, it was considered anyway primarily due to its past success. At the 2000 competition, many of the top teams were using biplanes and this observation was a primary factor which led to the adoption of this configuration by the U of A's AIAA team for the 2001 competition. As a result, the plane's performance last year was better than had been expected, and it ended up with a flight score in the top ten. Along with its proven capability, a biplane excels in lift and also is predicted to do well with maneuverability, something that is an important factor this year since precious time may be saved in turns if the aircraft can expedite them quickly. Also, with our limited level of experience, this configuration looked to be one of the easiest to design and construct since most of the previous year's techniques could be used. Despite these strengths

however, there was a strong desire to depart from this configuration since it did not fit within the dominant design drivers of weight and drag reduction.

2.3-Canard:

This configuration was offered some distinct advantages over other configurations. For one, the canard can actually give an upload to satisfy the longitudinal static stability requirements of the plane, as opposed to a download that must be induced on a conventional plane, so the overall lift would be greater. The configuration also has better lift to drag ratios, better stability (if designed right), better aerodynamics over the front of the plane (since the prop is usually in the rear), and the fuselage can be more compact due to the removed necessity of carrying a useless empennage. One of the disadvantages of the configuration involves risk. Since the plane is more difficult to design, and not designed correctly, the performance, particularly stability, will be very poor. Also, since the main wing lies in the tip vortex of the canard, it experiences poor induced drag behavior and has an odd lift distribution.

One non-technical feature that made the canard particularly attractive was the limited number of teams at previous competitions that had tried to fly one and were able to fly one well. With our team's limited experience, having success with a canard configuration would prove to be a very rewarding design experience and something that would gain our school's AIAA club quite a bit of prestige.

2.4-Flying Wing:

This configuration was not ever taken very seriously as a possible option but was included in our study partly for reasons of diversification, and also because it also offer several distinct advantages. It represents the most compact design and would save a lot

on weight and also competition cost. It would also be the most aerodynamically efficient design and present a tremendous savings on drag. Also in regards to a design experience, this would certainly be the most rewarding.

Naturally, such a design is one of the most difficult of aircraft configurations and not one to be attempted with our team's level of experience. Also, with the limited amount of time that was budgeted for our design phase, it was not expected that such a difficult design would be able to receive the amount of careful attention that it demanded. Regardless, it was examined to see if it offered astounding advantages that would compensate for the risk.

2.5-Final Configuration:

After our research and lengthy discussions over the different configurations, the WWII fighter and canard were the ones selected to take into the detailed design phase, so preliminary sizing began on both. It soon became apparent; however, that the canard configuration was suffering from some major design issues. One problem was the limited availability of pusher props. It was decided early on that the canard configuration would be a pusher since mounting the motor in the front would create a nose heavy and unstable plane. It was also unknown how the prop wash over the canard would affect its performance. Since our team was attempting to have props donated by Desert Aircraft, we felt it would not be courteous to ask them to special order a pusher prop. The other option, using a regular prop and reversing the polarity of the motor to make it spin backwards, was discarded since that severely decreases the prop's efficiency. Problems also arose with prop clearance, considering the fact that when the plane would rotate for takeoff, the prop would move closer to the ground, and therefore it became apparent that

the prop size would have to be limited. Finally, considering the lack of time for the design, it was decided to abandon the canard idea.

Shown in Figure 2 is a Pro-E model of the final configuration. Some features, such as the tricycle landing gear, depart from the typical WWII fighter but the tapered monowing and sleek aerodynamics are the retaining features that have been employed.

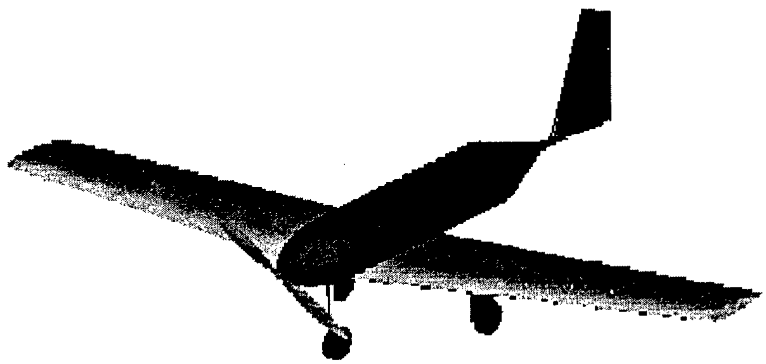

Figure 2: Final Configuration

3-Wing Design

The design of the wing mainly involved consideration of takeoff and landing requirements within our monowing configuration constraints. Consideration was also given to stability, construction, and assembly. The design of the wing was comprised of planform selection, airfoil selection, and wing sizing.

3.1-Planform Selection:

The planform shape of a wing has a significant effect on the aerodynamics of the plane and is therefore treated as a major component of the wing design. There are three basic types that are investigated, rectangular, elliptic and tapered. A rectangular wing is generally a poor choice of a planform shape in terms of its aerodynamic efficiency due to its large amount of induced drag. It is however used quite a bit due to its ease of construction, when drag or efficiency isn't a major design consideration. Also, the University of Arizona AIAA teams have used it in the past two years and were pleased with the results. This year however, low drag is a major design consideration; thus it was decided to look into other planforms. The rectangular wing was still investigated however for comparison.

An elliptic wing is the most aerodynamically efficient planform shape to have. The shape results in a perfectly elliptical lift distribution, which has a maximum Oswald Factor of 1, and the induced drag created from the tip vortex system is therefore the lowest of any shape with the same aspect ratio. Although construction of such a wing is very difficult, our team had the option of using Raytheon's, a local company, C & C foam cutting machine. Therefore, the elliptic wing was investigated.

A tapered wing represents a compromise between a rectangular and an elliptic wing. With the right taper ratio, it can approximate an elliptic shape. Although it is more difficult to construct, it is much easier than the elliptic and would not require the use of the C & C foam cutter. Oswald factors vary with taper ratio, but a tapered wing usually has much better drag characteristics than a rectangular. Other planform shapes, such as deltas or other wings with high sweep, were not investigated since these are more suited for high speed, high Reynold's Number flight.

The planform was selected by sizing all three at once and comparing results. Given the wing size parameters, aircraft weight, and airfoil information, the required takeoff speed was calculated. The Project Coordinator for the AIAA and expert modeler was then consulted to make a judgement as to whether the takeoff speed was too high. The procedure for calculating takeoff speeds is outlined in Appendix A, and the results are shown in the excel spreadsheet on the next page (Table 1).

In this spreadsheet, the input parameters are the aircraft, the wing span, root chord, the airfoil's maximum lift coefficient, the airfoil's lift curve slope, taper ratio (tapered wing only), T and Delta (tapered and rectangular wings only) which are correction factors for non-elliptical lift distributions. These correct the induced angle of attack, decrease and induced drag coefficient respectively. Calculated from these inputs are the wing area (S), the wing aspect ratio (AR), the decrease in the induced angle of attack (alpha), effective lift coefficient due to induced effects (Cleff), wing loading (W/S), and induced drag coefficient (Cdi). Finally, from the first and second column inputs, the takeoff speed (VTO), wing mean aerodynamic chord (MAC) and takeoff Reynold's Number (ReTO) based on the MAC are calculated.

Table 1: Wing Planform Results

Planform					
Elliptic					
Weight (lb)	25	S (ft^2)	8.836	VTO (mph)	**27.07**
span (ft)	7.5	AR	6.366	MAC	1.273
chord (ft)	1.5	alphai (deg)	5.673	ReTO	321497
Clmax	2.2	Cleff	1.670		
a (1/deg)	0.0934	W/S (oz/sqft)	45.27		
		Cdi	0.145		
Tapered					
Weight (lb)	25	S (ft^2)	9.000	VTO (mph)	**27.19**
span (ft)	7.5	AR	6.250	MAC	1.225
root chord (ft)	1.5	alphai (deg)	6.154	ReTO	310688
taper ratio	0.600	Cleff	1.625		
a (1/deg)	0.0934	W/S (oz/sqft)	44.44		
Clmax	2.2	Cdi	0.151		
T	0.065				
delta	0.02				
Rectangular					
Weight (lb)	25	S (ft^2)	11.250	VTO (mph)	**26.10**
span (ft)	7.5	AR	5.000	ReTO	365229
chord (ft)	1.5	alpha (deg)	8.451		
a (1/deg)	0.0934	Cleff	1.411		
CL	2.2	W/S (oz/sqft)	35.556		
T	0.17	Cdi	0.198		
delta	0.070				

For the planform comparison, identical spans and root chords were chosen for the three different wings. Naturally, the rectangular wing offers the most lift (lowest takeoff

speed) since it has more wing area for the span and chord dimensions. It also has the greatest amount of induced drag and a very low effective lift coefficient. The tapered wing, due to its selection of taper ratio, offers the same amount of lift as the elliptic wing for a slightly larger wing area and lower aspect ratio. The induced drag is higher but only by about 4%. The effective lift coefficients are also about the same. Although it is clear that the elliptic planform is slightly more efficient than the tapered, it was decided that the tapered wing would be selected since this can be made in the school lab and does not rely on using a local company's equipment. Also, with elliptic wings, tip stall can be a concern since the Reynold's Number near the tips can become very small on takeoff and landing. A tapered wing with a large enough taper ratio can avoid this.

3.2-Airfoil Selection:

The airfoil section of the wing is one of the most important parts of the wing design. The airfoil determines how much lift the wing can generate; to what angle of attack the wing may be pitched before stall, the profile drag of the wing, and the nose down pitching moment. The selection of the airfoil was therefore carefully selected after a bit of research. The following airfoils were researched:

Table 2: Researched Airfoils (Gott = Gottingen)

Clark Y	Clark Y-15	NACA 2312	NACA 2315	NACA 2412	NACA 2415
NACA 4212	NACA 4412	NACA 4415	NACA 4712	NACA 4716	NACA 6212
NACA 6612	NACA 6712	NACA 6716	NACA 9716	Gott 387	Gott 481a
Gott 498	Eppler 212	Eppler 593	Eppler 395	Eppler 397	

Data for the Epplers was found in Ref. [1]. Ref. [2] was used for the Clarks and Gottingens, and both Ref. [2] and the computer program "Visualfoil" were used for the NACA's. The airfoil properties of interest were maximum lift coefficient, lift curve slope, maximum lift/drag ratio, lift at zero angle of attack, drag coefficient at low angle of

attack and pitching moment at the mean aerodynamic center. The NACA airfoils proved easiest to work with since the "Visualfoil" program provided enough data, even if it was based only on sophisticated math models instead of wind tunnel data. According to the takeoff speed calculations, which were being done concurrently, it was determined that a high lift airfoil was going to be needed in order for the plane to takeoff at a reasonable speed. The maximum lift coefficient then became the most important parameter. To achieve this lift, high camber airfoils were considered more intently. Of these, the NACA 6714 seemed to perform well. The 6% camber is a lot for a standard airfoil, and placing the high camber point far back at about 70% of the wing chord leads to a high CLmax. The drag is not unreasonably high, and the airfoil has a reasonable angle of attack range as well as a fairly good lift/drag ratio. The main disadvantage of the airfoil however is its rather high pitching moment (due to the camber). This led to some challenges in the design of the horizontal tail. The main airfoil properties are listed in Table 4, and a plot of the airfoil is shown in Figure 3:

Table 3: NACA 6714 Properties

Clmax	2.2
a	0.0934 1/deg
L/Dmax	18.5
CL (α=0°)	0.8
CD (α=5°)	0.05
Cmac (α=0°)	-0.271

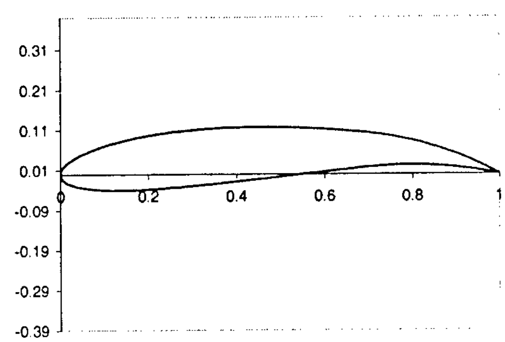

Figure 3: NACA 6714 Airfoil

3.3-Wing Sizing:

The primary consideration in picking the wing dimensions was to achieve a takeoff speed low enough for the pilot to comfortably handle the plane on takeoff and landing. Keith

Brock was consulted in order to get a pilot's subjective opinion on whether a speed was too high. Unfortunately, even with the high lift airfoil, the required takeoff speeds tended to be high. At one point, the wing was sized to have an 8 ft. span and 2 ft. root chord, and this set the takeoff speed at about 24 mph. After cutting the wing planform from tagboard and examining it, the size seemed to be unreasonably large. The wing was resized to a 7.5 ft. span and 1.5 ft. root chord, which raised the takeoff speed to 27.19 mph. The AIAA team members felt that, based on their experience, the estimate was too high. Even though many factors had been considered in the calculations, such as tip vortices and ground effects, it was decided to leave the wing sized at the 7.5 ft. design. The takeoff speed question was deferred to wind tunnel tests and flight tests since our experience told us that our plane would not have a problem getting off the ground. The excel spreadsheet, seen in Table 1, that was used to compare planforms, contains all the data for the current wing design in the tapered section. A scale drawing of the wing planform is shown in Figure 4.

Figure 4: Wing Planform

3.4-Wing Construction:

The wing was constructed using many of the same materials and techniques used in previous years by the AIAA teams. The wings are constructed from styrofoam using a hot wire bow. Airfoil templates sized to the root and tip chords were constructed by printing a 1:1 scale plot of the airfoil from the computer program "Winfoil", gluing the

printout to modeling plywood, and cutting around the line. The templates were then attached to either side of a block of foam, and the hot wire was then run around the templates. The trailing edges were then cut and replaced with balsa tailing edge stock. After the wings were sanded and components were installed, the wings were eventually covered with Ekonocote, an iron-on modeling plastic. To cut down on the wing tip vortex, the wing tips were sheared at 45 degrees. A wing spar, made from composite materials, offered the advantages of lightweight and high stiffness. The wings were joined together with 5 degrees of dihedral, and the fuselage was placed on top. The low wing configuration was chosen so that the motor would be elevated higher off the ground to give more prop clearance, and also so the plane would maneuver better around turns.

4-Tail Design

The tail designed consisted of the horizontal stabilizer, vertical fin, and the final construction.

4.1-Horizontal Stabilizer:

The main purpose of the horizontal stabilizer is for balancing the plane longitudinally. Since the wing inherently has a nose down pitching moment due to the camber of the airfoil, another component must be added to generate a moment that can combat that of the wing. A horizontal stabilizer placed behind the wing with an appropriately sized download can accomplish this purpose. In order to properly design the stabilizer, an excel spreadsheet (Table 4) was used employing both moment balance and neutral point location equations (see calculations in Appendix B).

Table 4: Longitudinal Static Stability Spreadsheet

Longitudinal Stability Calcs						
Moment Balance:						
CLw	0.8	Vhbar	0.602			
Cmwac	-0.271	ltbar	4.42	ft		
MAC (ft)	1.225	LCG	0.658	ft		
St/S	0.167	Lfuse	5.88	ft		
hCG-hacw	0.175			NACA 0012 with 3deg incidence		
				gives CLt = -.23		
Neutral Point Location:						
K	0.1	CLt	-0.2287	(download)		
aw (1/deg)	0.0934	Cltot	0.7619			
at (1/deg)	0.072					
deps/dalph	0.362					
a (1/deg)	0.1011					

The spreadsheet inputs are the wing lift coefficient at zero angle of attack (CLw), the pitching moment at the wing's mean aerodynamic center (Cmwac), the wing's mean aerodynamic chord (MAC), the tail to wing area ratio (St/S), the CG location relative to the wing's mean aerodynamic center (hCG-hacw), the static margin (K), the lift curve slopes of the wing and tail respectively (aw and at), the downwash angle slope (deps/dalph), and the lift curve slope of the plane (a). With these inputs and some rough dimensions of the fuselage, the parameters calculated were the horizontal tail volume ratio (Vhbar), distance between the wing and tail mean aerodynamic centers (ltbar), CG location relative to the wing apex (LCG), fuselage length (Lfuse), tail lift coefficient, (CLt) and overall lift coefficient due to the download (Cltot).

The dimensions of the horizontal tail were somewhat fixed by the competition cost mode, stating that a horizontal surface, which has a span more than 25% of the wing, will be considered a second wing. The corresponding cost then increases largely. Since the tail needed to be as large as possible, due to the large pitching moment, it had to fit. The span was set at 1.875 ft (25%). A chord length of 9.6 inches was chosen to give the tail a reasonable aspect ratio. The static margin was also set at a design value of 10%, which is comfortable for most planes.

The main parameter that was then varied was the CG location. It was placed by using the spreadsheet in Table 4 to find a compromise between a short enough fuselage length and a tail download which did not decrease the overall lift too much. The design placement is shown in Table 4 having a value of 17.5% of the mean aerodynamic chord (about 2.5 in). It was found by typing in different CG locations and observing how the

output parameters were adjusted by the spreadsheet. The overall lift coefficient has only been decreased to 0.7619 from 0.8, a significant amount but hopefully not too much. The required fuselage length is roughly 5.88 ft. Since there was concern expressed that this may be too long, the fuselage length was arbitrarily set to 5.25 ft and the question of stability deferred to the wind tunnel and flight tests. Given the approximate nature of the analysis, this length seemed to be roughly close to what it should be and any imbalance could hopefully be trimmed out. The tail download was achieved by selecting an NACA 0012 airfoil and placing it at 3 deg of incidence.

4.2-Vertical Fin:

Design of the vertical fin must be performed with characteristics like spiral stability and weathercock stability in mind. For this plane however, such issues did not seem to be of very much importance, and an appropriate trapezoid shape was chosen by observing the fins of the previous year's planes. Such questions of whether or not enough lateral stability would be available were deferred to flight tests.

4.3-Tail Construction:

The horizontal stabilizer was, like the wing, cut from foam by running the hot wire over airfoil templates. It was given a balsa wood spar and covered with an Ekonocote skin, for structural reasons as well as a means for attaching it to the vertical fin. The vertical fin was constructed using the previous year's methods, which consisted of cutting the flat trapezoid shape out of foam and sandwiching it between layers of carbon graphite. A rounded balsa leading edge was later added.

5-Fuselage Design

5.1-Selected Fuselage Configuration

We decided to go for a three abreast by eight deep softball configuration that will give us a small frontal area and a reasonably short fuselage length. The configuration yields a volume of 15x10x36 in^3 in the cargo area (see Figure 5). The 24 softballs will occupy an upper deck while a lower deck will be dedicated to a battery rack, measuring 2.5x5.15 in^2 that will be positioned on the CG of the aircraft and above the wing.

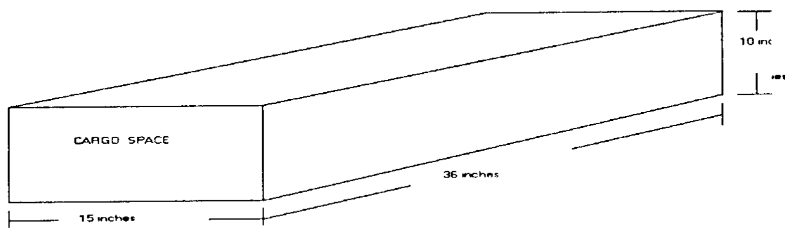

Figure 5: Cargo Bay Dimensions

The softball configuration was chosen based on the mission requirements. A two abreast configuration would yield a very long fuselage that would be penalized as described in the competition rules. On the other hand, the four abreast design had a large frontal area that would create a higher drag coefficient. Even though the cargo was short, the fuselage length (from the tip of the propeller to the trailing edge of either the elevator or the rudder) would be about the same (63 in., after the calculation for pitch control: see wing and tail design).

The engine area is covered with a thin fiberglass cowling. After the aft bulkhead (tail section), the fuselage will be gradually tapered upward on the bottom section of the fuselage and slightly leveled on the top section of the fuselage.

To reduce drag further, any outside edges will be rounded, especially around the hatch area. The hatch is designed to secure the payload (softballs) during the mission and for quick access (during loading and unloading).

5.2-Fuselage Material:

The fuselage will be primarily made out of styrofoam, reinforced with balsa wood and honeycomb strings lengthwise. The styrofoam was chosen due to its strength to weight ratio. However, the front and aft bulkheads will be made of wood reinforced with honeycomb panel.

5.3-Fuselage Assembly:

The fuselage will be embedded on top of the wing. Fairings or fillets will be designed to reduce sharp edges that produce drag, which takes away from the aircraft power required to complete the mission. For shipping purpose, the wing and the fuselage will be attached using two mating plates, so both wing and fuselage can be detached from each other. The hatch will be secured to the fuselage using quick disconnect fasteners for fast loading and unloading.

6-Landing Gear

6.1-Selection of Landing Gear

The first decision that had to be made was whether or not to use retractable landing gear. Based on trouble that the University of Arizona AIAA club was having in previous years trying to make retractable gear work on the club plane, it was noted that this might not be the route to take. Also, having retractable gear would require more servos, which would increase our cost and decrease our score. Finally, it was felt that there just was not enough time to use retractable gear due to its complicity. After deciding against retractable, a decision had to be made about whether to use a tail dragger or tricycle configuration. One benefit to tricycle gear is that its nose wheel steering is a little more stable than the tail dragger. The tail dragger weaves in and out a lot and is much harder to control. For an inexperienced pilot, as our team has, it is one more thing for the pilot to be concerned about. For the wheels, it was decided to use about 2-2.5 diameter rubber wheels. These will be large enough to hold our plane, but also are small enough that the drag is not greatly increased.

The material in which the rear gear will be made out of is a mixture of carbon graphite, fiberglass, and epoxy. It will be constructed over a preexisting wooden mold. This composite is light enough that it will not add too much weight to the plane, but will be strong enough to hold up against any conditions. Also, it has proven to work very well for the U of A AIAA club in past competitions. The nose wheel gear was bought at a local hobby shop and as used with a 4 inch diameter wheel.

7-Controls

The control consisted of the ailerons and the servos.

7.1-Ailerons:

When sizing the ailerons, the relative dimensions that we found for our wing design came from a diagram on page 49 of Ref. [3], which cited NACA Report #605 as its source. For the length, it was suggested to devote 30-40% of the semi-span of the wing. Looking at the size that each percent would give us and based on the entire wingspan, we chose to go with a slightly shorter size at about 35%. For the width, we chose to use 25% of the chord.

Rather than buying generic aileron stock at a hobby store, the ailerons were cut directly from the wing in order to preserve the airfoil contour. The trailing edge was replaced with balsa wood trailing edge stock and a thin layer of balsa covered the wing-mating surface.

7.2-Servos:

The servos being used are standard size Futaba servos. They produce 44 oz.-in. of torque. There are two servos being used for the ailerons, each mounted in the respective wing. There is one servo for the elevator and one servo for the rudder, which are both located in the back of the fuselage. The receiver is also located in the back of the fuselage by the two servos. An extension for the servos in the wing was added in order to connect them with the receiver.

8-Power Plant

Through the University of Arizona AIAA, there were two main types of motors that could be used, a gearbox and a direct drive. The gearbox engine is able to spin a larger prop at a lower rpm while the direct drive is able to swing a smaller prop at a higher rpm. There were also two sizes available, an Astroflight Cobalt 60 and Cobalt 90. Since the plane has not been fully constructed, the weight of the aircraft is unknown, so an estimation of 25 pounds, fully loaded was used. With the choice of what engine to use, there was also the choice of what propeller to use. There are a wide range of diameters and pitches to choose from. Because there is a current draw limit from the battery pack of 40 amps, certain propellers could not be used with some engines since they draw more than 40 amps. Not only is there a current limit, there is also a limit on the amount of batteries that can be used. That limit is 5 pounds. With these restrictions, a static thrust test was designed to determine which arrangement yielded the most thrust.

For this test, a very basic static thrust stand was constructed. This stand consisted of a wooden "arm" where the motor was mounted. Attached to this arm was a simple spring scale to measure the static thrust. The first engine tested was the Cobalt 60 gearbox. Through the use of different propeller sizes and pitches, a maximum of 9.5 pounds of static thrust was obtained. This value is far below what is needed, so no further tests of the Cobalt 60 gearbox were done. This also led to not testing the Cobalt 60 direct drive engine.

The second motor that was tested was the Cobalt 90 gearbox. Once again, several propellers were tested with different diameters and pitches. The maximum amount of static thrust that was obtained was 20 pounds. This was obtained by using a 28 x 14 inch

propeller. The diameter was 28 inches, and the pitch was 14 inches. For every full revolution, the aircraft should move 14 inches in the horizontal direction.

The final engine to be tested will be the Cobalt 90 direct drive. This test has not been completed because the engine manufacturer did not send the prop adapter that was needed. This prop adapter has recently arrived, and testing will be completed. Values higher than 20 pounds of static thrust will hopefully be obtained but is not guaranteed.

Static Thrust results

Propeller	Size	Thrust
Classic Series	24 x 12	12
Classic Series	24 x 14	13.5
Classic Series	26 x 12	14
Classic Series	26 x 14	17
Classic Series	28 x 14	18.5
JZ Zinger	24 x 12	13
JZ Zinger	24 x 14	15
JZ Zinger	26 x 12	15.5
JZ Zinger	26 x 14	17
JZ Zinger	28 x 14	18.5
MEJZLIK Modellbau	24 x 12	14
MEJZLIK Modellbau	24 x 14	16.5
MEJZLIK Modellbau	26 x 12	16
MEJZLIK Modellbau	26 x 14	18.5
MEJZLIK Modellbau	28 x 14	20

Table 5: Static thrust test results

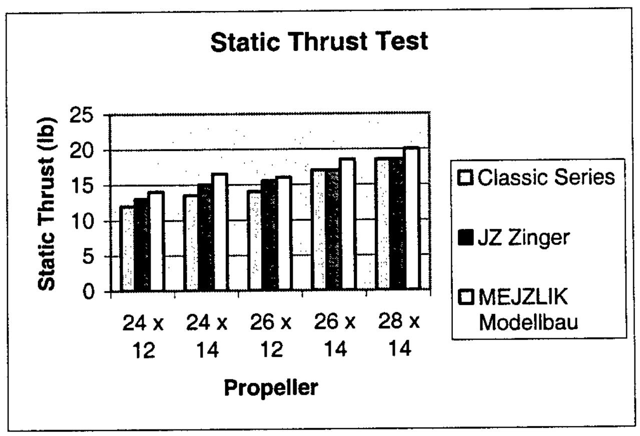

Figure 6: Static thrust graph

9-Wind Tunnel Testing

9.1-Set up of Wind Tunnel and Models

Figure 7: Model Inside 3X4 ft Undergraduate Wind Tunnel

The testing was performed in the University of Arizona's 3 ft X 4 ft Undergraduate Low Speed Wing Tunnel. The main purpose of the wind tunnel test was to determine the lift, drag, and pitching moment coefficients of the plane. The effect of the tail was also an important, but secondary, idea to be studied. In order to use the wind tunnel, two models were constructed. The models are 25% of full-scale size. The models were constructed out of foam and balsa wood. The wings and the fuselage were made out of foam, while the trailing edge was made out of balsa. The wings were first sanded, then covered with glue and spackle to conceal the cracks. Once this was dry, the rough edges were once again sanded. This process was repeated two more times to ensure a smooth airfoil and fuselage. The wings were then glued together and the fuselage was mounted on top of the wings. This was done by cutting a groove into the fuselage that

was identical to the airfoil. The fuselage was then glued on top. The next step of the model construction was figuring out a way to attach the model to the wind tunnel stand. This was accomplished through the use of a mating plate. A plate was submerged into the foam on the underside of the model and then was attached to the stand. This entire process was repeated for both models.

9.2-Description of Preformed Tests and Results

The first model was of a 7.5 foot wing span airplane. The second model was of an 8 foot wingspan. The first model tested was the 7.5 foot model without the tail. This model was then tested with the tail. This enabled comparison of the data to determine the effects of the tail. The final test consisted of the 8 foot model with the tail. No tail comparisons were done with this model because the wingspan of the actual plane is 7.5 feet. The 8 foot model was done to compare to the 7.5 foot model.

The following graph shows the lift to drag ratios of the three different tests.

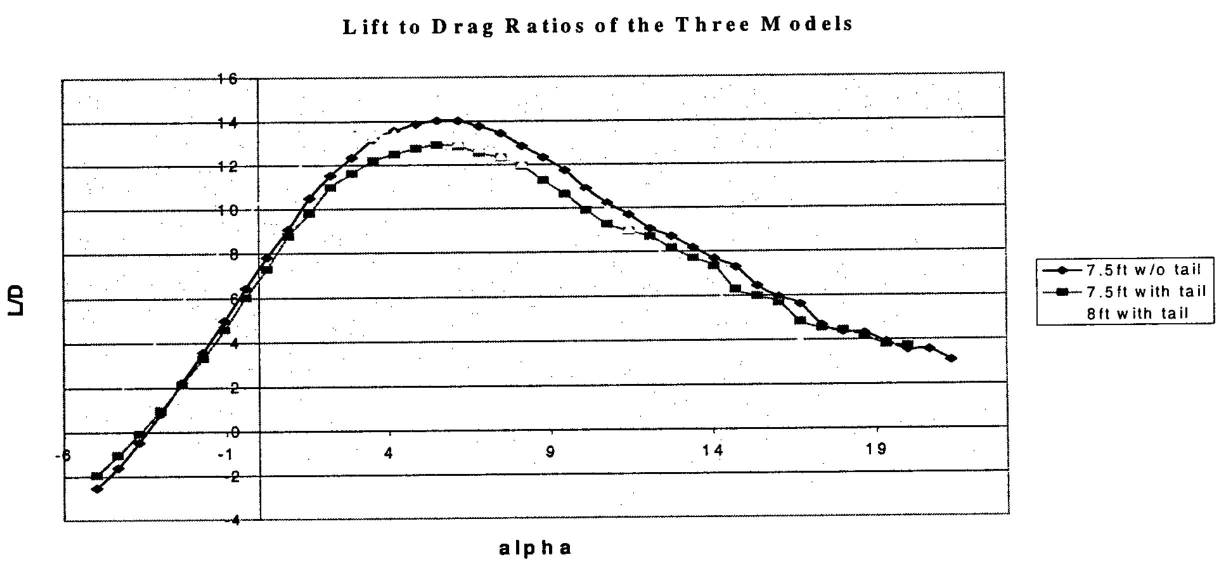

Figure 8

The next graph shows the lift, drag, and pitching moment coefficients for the 7.5 foot model.

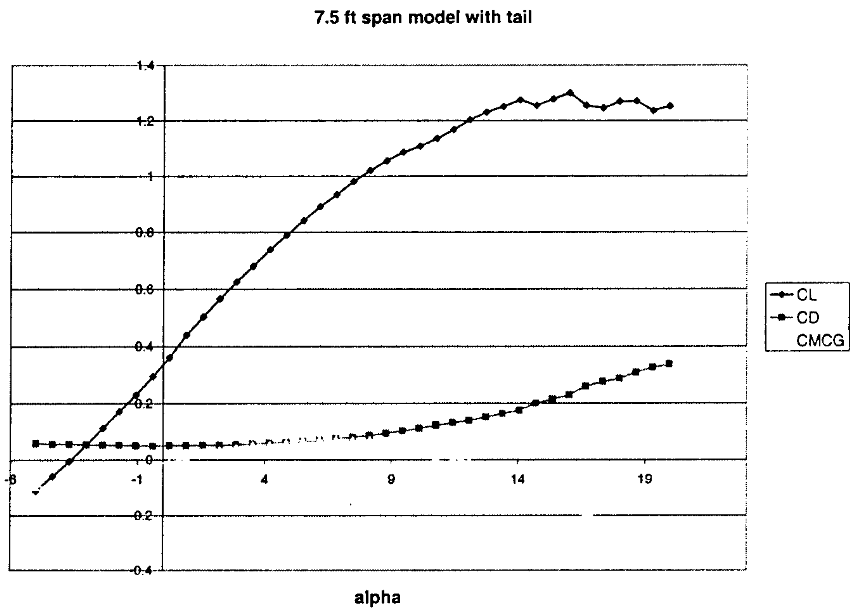

Figure 9

As you can see from the graph, the model has a C_L of about 1.3 and a C_D of about 0.38, which gives a lift to drag ratio of about 10. The effective CLmax is disappointingly small at about 1.3 as compared to the 1.625 calculated in Table 1. Reasons for this drop may be due to the integration of the fuselage, imperfections in the wing (which get magnified by a factor of 4 in scaling down), and also the fact that the test was run at a lower Reynold's Number than will be experienced in actual flight.

The final graph shows the pitching moment coefficients about the CG for all three test stages; 7.5 foot model without the tail, 7.5 foot model with the tail, and 8 foot model with the tail.

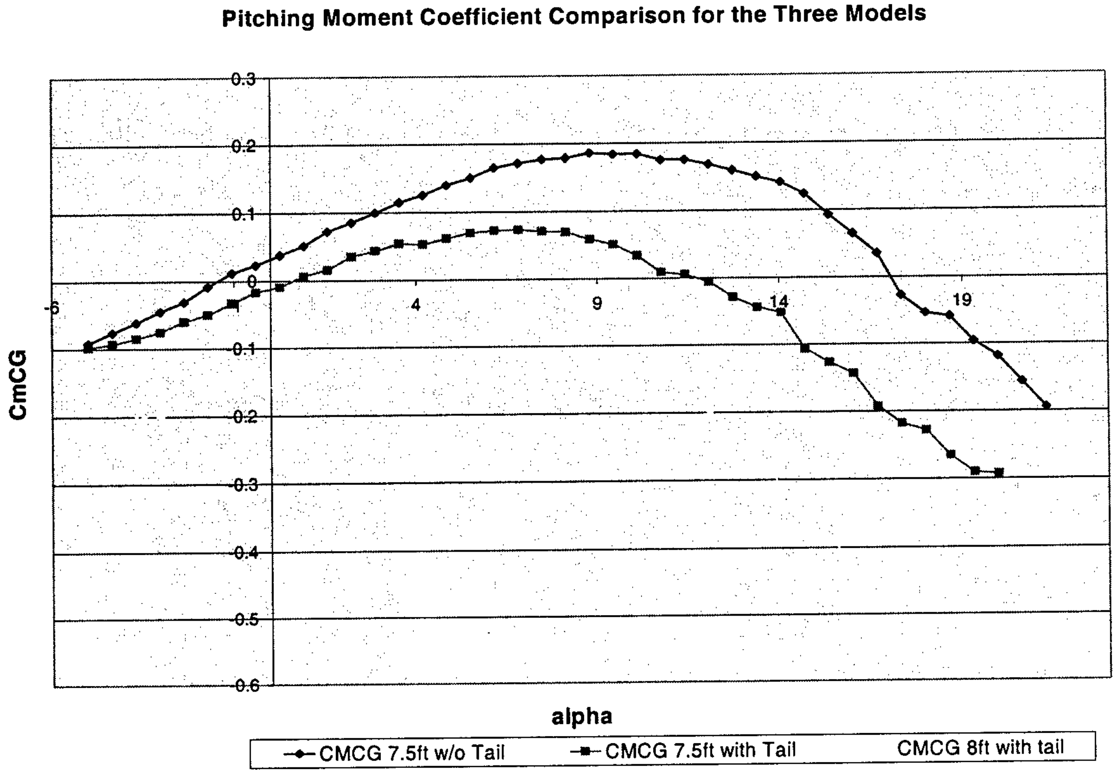

Figure 10

As you can see, the tail seems to be working since the curve for the 7.5 ft span model with tail is closer to zero than the w/o tail curve. The 8 ft moment coefficients are larger in magnitude than the 7.5 ft span model though. As mentioned previously, the range of positive moment coefficients is an unusual feature for an aircraft to have and the results of Figure 10 must therefore be treated with some skepticism.

10-Calculations

10.1-Wing Planform/Take off Speed Calculations

Input parameters:

Weight: W = 25 lb, Wing Span: b = 7.5 ft, Root Chord: $c_r = 1.5$ ft, $CL_{max} = 2.2$, Lift Curve Slope: a = 0.0934 1/deg (measured in Ref. [2]), Taper Ratio: $\lambda = 0.6$

Calculated parameters:

Induced angle of attack efficiency factor: T, depends on taper ratio, value is read from Figure 2 on page 5 of Ref. [3]. T = 0.065 (tapered wing, $\lambda = 0.6$), T = 0.17 (rectangular wing, $\lambda = 1$)

Induced drag coefficient factor: δ, depends on taper ratio, value is read from Figure 5.18 on page 340 of Ref. [4]. $\delta = 0.02$ (tapered wing, $\lambda = 0.6$), $\delta = 0.07$ (rectangular wing, $\lambda = 1$)

Wing Area: $S = \pi/_4 c_r b = 8.836 ft^2$ (elliptic), $S = \frac{c_r b}{2}(1+\lambda) = 9.0 ft^2$ (tapered)

$S = c_r b = 11.25 ft^2$ (rectangular)

Aspect Ratio: $AR = b^2/_S$ = 6.366 (elliptic), 6.25 (tapered), 5.0 (rectangular)

Induced Angle of Attack Decrease: $\alpha_i = \frac{C_L(1+T)GEFI}{\pi AR}$ (Equation 5.42 of Ref. [4] modified by factor in Ref. [3]), GEFI = 0.9 (ground effect factor, read from Figure 3 on page 34 of Ref. [3] assuming a wing height of 1ft)

$\alpha_i = 5.673°$ (elliptic), 6.154° (tapered), 8.451° (rectangular)

Effective Maximum Lift Coefficient: $CL_{eff} = CL_{\max} - a\alpha_i$, due to the downwash created by the tip vortex system, the wing sees a lower angle of attack and the overall lift coefficient is therefore reduced.

$CL_{eff} = 1.670$ (elliptic), 1.625 (tapered), 1.411 (rectangular)

75

Wing Loading: W/S = W/S*16oz/sqft = 45.27 oz/sqft (elliptic), 44.44 oz/sqft (tapered), 35.56 oz/sqft (rectangular)

Induced Drag Coefficient: $C_{Di} = \frac{C_L^2}{\pi AR}(1+\delta)GEFII$ (Equation 5.61 of Ref. [4] modified with factor from Ref. [3]), GEFII = 0.6 (ground effect factor, read from Figure 4 on page 34 of Ref. [3] assuming a wing height of 1 ft)

C_{Di} = 0.145 (elliptic), 0.151 (tapered), 0.198 (rectangular)

Takeoff Speed: $V_{TO} = 1.05\sqrt{\frac{2W/S}{\rho C_{Leff}}}$, definition of the lift coefficient, C_{Leff} and sea level density are used, takeoff speed is considered to be 5% greater than stall speed.

V_{TO} = 27.07 mph (elliptic), 27.19 mph (tapered), 26.1 mph (rectangular)

Mean Aerodynamic Chord: $\bar{c} = \frac{8c_r}{3\pi} = 1.273\,ft$ (elliptic),

$\bar{c} = \frac{2c_r}{3}\frac{1+\lambda+\lambda^2}{1+\lambda} = 1.225\,ft$ (tapered), $\bar{c} = c_r = 1.5\,ft$ (rectangular), equations taken from Table C.2 in Ref. [5].

Takeoff Reynold's Number: $\mathrm{Re}\,TO = \frac{V_{TO}\bar{c}}{\nu}$, sea level value of ν is used.

Re_{TO} = 321,497 (elliptic), 310,688 (tapered), 365,229 (rectangular)

10.2-Longitudinal Static Stability Calculations

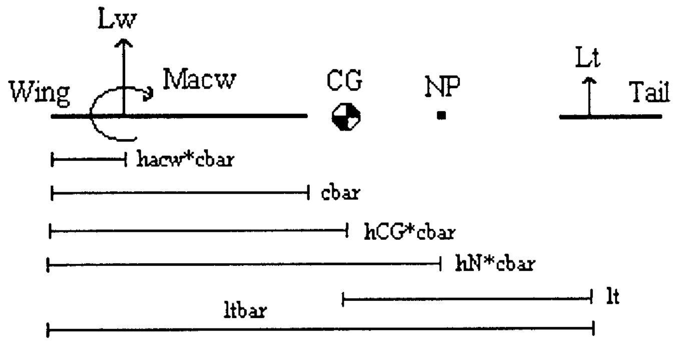

Figure 10: Wing and Tail Free Body Diagram

Moment Balance:

(Neglect drag and tail pitching moment contributions)

$$\Sigma M_{CG} = M_{acw} + L_w(h_{CG} - h_{acw})\bar{c} - L_t l_t$$

divide by $\frac{1}{2}\rho V^2 S\bar{c}$: $C_m = \dfrac{M}{\frac{1}{2}\rho V^2 S\bar{c}}$ $\quad C_{Lw} = \dfrac{L_w}{\frac{1}{2}\rho V^2 S}$ $\quad C_{Lt} = \dfrac{L_t}{\frac{1}{2}\rho V^2 S_t}$

$$V_H = \frac{l_t}{\bar{c}}\frac{S_t}{S}$$

Substituting in: $C_m = C_{macw} + C_{Lw}(h_{CG} - h_{acw}) - C_{Lt}V_H$

$$\overline{V_H} = \frac{\bar{l}_t}{\bar{c}}\frac{S_t}{S} = V_H + \frac{S_t}{S}(h_{CG} - h_{acw})$$

Equilibrium Balance Equation (Cm = 0):

$$\boxed{C_{macw} + C_{Lw}(h_{CG} - h_{acw}) - C_{Lt}(\overline{V_H} - {}^{S_t}\!/_{S}\,(h_{CG} - h_{acw})) = 0} \qquad [1]$$

Static Margin and Neutral Point Location:

$$h_N = h_{acw} + \frac{a_t}{a}\overline{V_H}\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)$$ (Equation 2.3,23 of Ref. [5])

$h_N - h_{acw} + h_{CG} - h_{CG} = K + (h_{CG} - h_{acw})$ $\qquad$ K ≡ Static Margin = h_N - h_{CG}

$$a = a_w \left(1 + \frac{a_t}{a_w}\frac{S_t}{S}\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)\right)$$

(Lift curve slope of the plane, from Equation 2.3,18 of Ref. [5])

Static Margin Equation:

$$\boxed{K + (h_{CG} - h_{acw}) = \frac{a_t}{a}\overline{V_H}\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)} \qquad [2]$$

Spreadsheet Algorithm:

1. Equation [2] is used to calculate V_{Hbar}: $\overline{V_H} = \dfrac{K + (h_{CG} - h_{acw})}{a_t/a\left(1 - \partial\varepsilon/\partial\alpha\right)}$

2. ltbar is calculated from this to obtain a fuselage length estimate: $\overline{l_t} = \dfrac{\overline{V_H c}}{S_t/S}$

3. Use Equation [1] with V_{Hbar} to calculate C_{Lt}: $C_{Lt} = \dfrac{C_{macw} + C_{Lw}(h_{CG} - h_{acw})}{\overline{V_H} - S_t/S\,(h_{CG} - h_{acw})}$

   Subtract tail download from overall lift: $C_L = C_{Lw} - S_t/S\, C_{Lt}$

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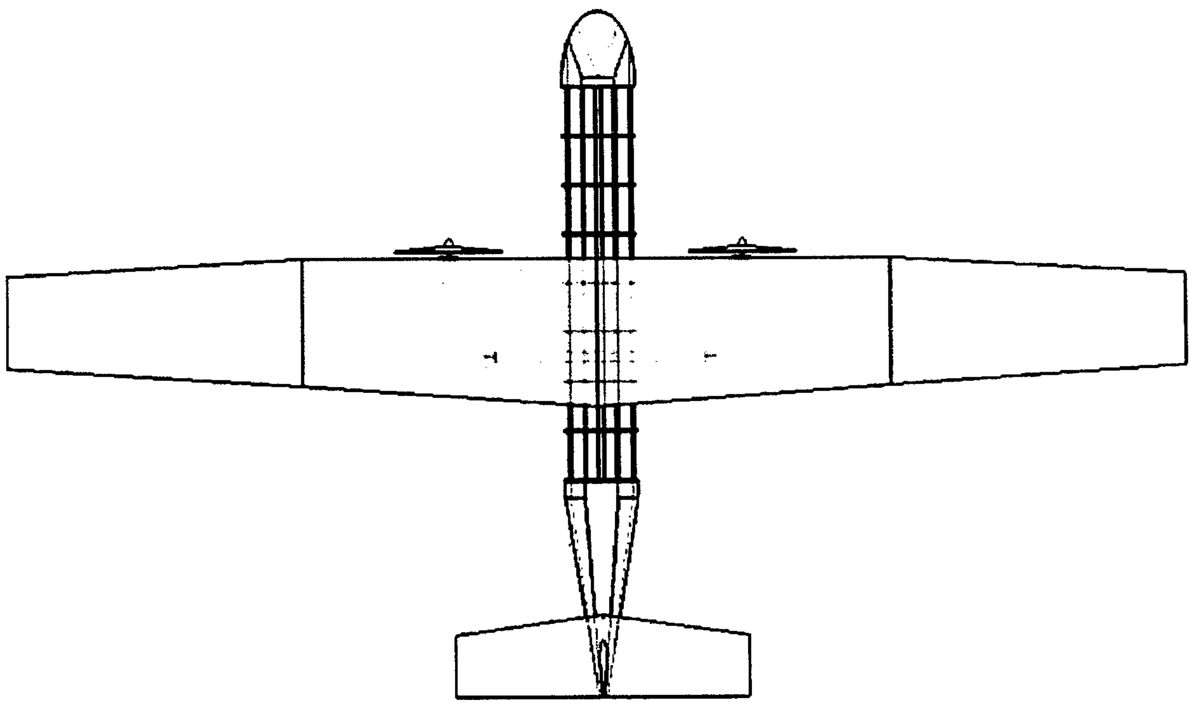


Learning Curve

Case Western Reserve University

March 12, 2002

1.0 Executive Summary

The development of an aircraft to carry 10-24 softballs around a predetermined track was accomplished in three phases. The conceptual design reviewed basic aircraft design concepts with attention given to fuselage and wing shape. Three concepts were reviewed including a flying wing design with the payload spaced through the length of the wing, a blended flying wing with a partial fuselage and payload spaced more along the longitudinal axis, and a standard wing and separate fuselage design with payload in the fuselage. The concept chosen was the wing and separate fuselage design because of more knowledge about this design and because of the ease of manufacturing and assembly. The preliminary design was drawn in Pro/Engineer and discussion was made as to the materials to be used for the bulkheads and supports, and also the electrical system. The materials used were based on analysis in Pro/Mechanica and cost considerations and the electrical and control system decisions were based on electrical analysis and cost considerations. The detailed design was the last phase in development which consisted of taking the preliminary design and detailing every part to the specifications desired and required, in other words, filling in the gaps of the preliminary design that were needed to construct and manufacture the aircraft.

2.0 Management Summary

During the fall semester of the 2001/2002 school year, the Case Western Reserve University AIAA Chapter formed a team to compete in this year's Cessna/ONR Student Design/Build/Fly Competition. The team was formed of 13 students from the Department of Mechanical and Aerospace Engineering. Of these students, 6 were seniors and the remaining were underclassmen. Our AIAA faculty advisor aided the team.

A project leader was selected at the beginning of the project to see that the project was properly coordinated and to ensure the completion of project tasks. The project leader served as the voice of the project members when dealing outside of the team. To better divide the tasks associated with the design and construction of the plane, three groups were formed. This allowed each group to be focused on a certain aspect of the project. A member was elected to head up the design of each subsystem. The leaders were then in charge of dividing tasks among the rest of the members of the team. The team structure is listed in the table below.

Project Leader: Eric Birch
Advisor: Dr. Jackie Sung

Airframe	Avionics	Propulsion
Leader: Dan Haylett	**Leader:** David Hiltner	**Leader:** Rob Falck

In order to set deadlines for the project, the team created a schedule of important dates and goals that needed to be achieved. The project leader was in charge of making sure that the schedule was followed and progress made. In addition, the leaders of the different groups were responsible for maintaining the parts of the schedule that fell in the area of their group. The schedule can be found on the following page.

The team met each week to discuss the project. During the initial stages of the project, this meeting time was used to brainstorm and finalize plans for the project. In the later stages of the project, this weekly meeting was used to update the entire team on the progress of each individual group and to bring up any topics that needed to be discussed as a whole group. The group leaders normally would assign tasks to other members of the team at these meetings. The group leaders would then follow up with those people assign to a task periodically throughout the week.

3.0 Conceptual Design:

After the various team members became familiarized with the basic rules of the competition and the basic requirements that must be met, the designing process commenced. In the early stages, we faced questions relating to basic structure of the plane. Before we could even consider the shape of the fuselage, wings, and tail, we had to decide how many wings it should have and how many fuselages. In order to avoid striking ideas prematurely, no ideas were crossed off until we completed brainstorming. Ideas included all combinations of one and two wing planes with one and two fuselages. These ideas were then narrowed down according to feasibility, reasonability, and contest rules. The next step was relatively hard compared to the first because at this point there were still several directions the plane could take, all of which were reasonable, and little hard data to compare them to each other with. We had decided that a one wing plane would be easiest to build, but we still had to decide where the balls would be stored, number of engines and placement, and wing placement. At this point we began to draw on our in house knowledge and experience, as well as doing some outside research to find the different advantages and disadvantages that each option offered. Among the issues that were considered were flight handling, power efficiency, cost, difficulty to build, and the designs ability to meet mission parameters. Flight handling included such features as maneuverability and stability and was extremely important because the pilot would likely be a newly trained RC pilot, or an experienced pilot who was unfamiliar with the plane. Power efficiency is important because it would be one of the key variables in how the fast the plane would fly. The cost is obviously important because of a limited budget and difficulty to build is important because of a limited time to build and a lack of experience within the group. Together all of the previous specifications are important because they will attribute to how the plane finally performs during the mission. Lastly it was important to keep in mind the parameters laid out before hand, because if it does not meet the mission specifications and follow the contest rules, it is in essence useless.

At this stage, we had three basic design structures being considered. As shown in the diagram to the right, from top to bottom: an inverted cargo plane, the classic cargo plane, a blended wing/body, and a dual fuselage design.

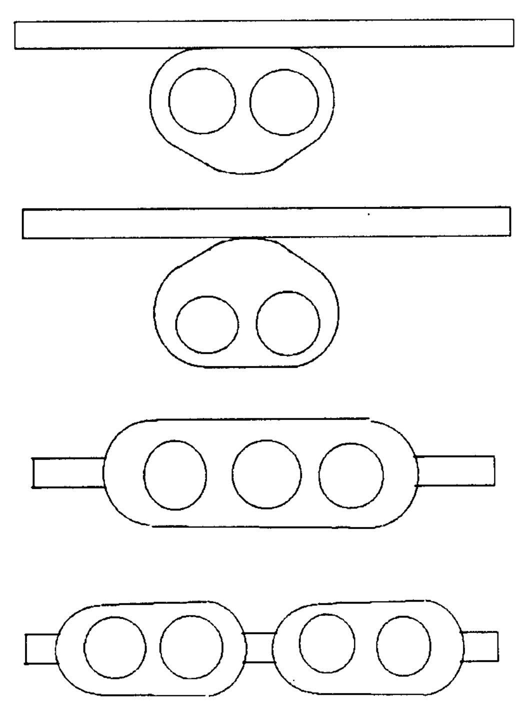


Figure 3-1: Fuselage Configurations

The basic idea behind the flying wing was that there would be no immediately recognizable fuselage, and that the balls would be stored in the center of an enlarged wing. This might also be referred to as a blended wing design. The big advantage with this design was that it should decrease the air drag and possibly even be a little

lighter than the designs with a more discrete fuselage section. However, we feared that it may be more challenging to construct and it would be much more difficult to control in flight. The dual fuselage design is rather self-explanatory. It would have a single wing, with two fuselages below it. While this design would introduce the fuselages to stresses primarily due to bending moments rather than torsion, the drag brought about by having two fuselages, each two softballs wide, was considered excessive and this option was therefore eliminated. The final design was basically your standard cargo plane design. This design might be described as your most basic plane, with an enlarged fuselage for bulky or weighty cargo; in this case, both. While this design may have a tendency to be slightly slower than a blended wing, it is more stable with the center of mass being suspended below the wings. Another advantage to this design is that it has past the test of time. As the Design/Build/Fly mission involves carrying cargo, we opted for this design. The fact that the fuselage is inverted with respect to the classic cargo plane design has two primary advantages. First, it allows more contact between high mounted wings and the fuselage, reducing the need for heavy reinforcements. In addition, it allows more room in the bottom of the fuselage for landing gear supports.

The different advantages of each design created a picture of what the team would want to strive for with our first plane. Because of it's in flight stability and flight efficiency, we decided that a top-wing design would be the best bet. We also decided that having two engines was probably the way to go. We also decided that an inverted triangle with rounded edges would be the most efficient shape for the fuselage because of the odd shape of the cargo and the importance of creating a design with minimal air drag.

In the selection of our configuration we used six design considerations:

- **Complexity** – Imparting the KISS principle on our design is important for two reasons. Increased complexity introduces more points of failure, and given our team's experience level, we decided that our first design should be relatively simple. Reducing complexity also reduces our rated aircraft cost by reducing the number of control surfaces and servos and the mass of the aircraft.
- **Weight** – Obviously weight is a primary concern in aircraft design. However, it can be offset by increasing power, to a point. Keeping weight down saves reduces the cost imposed by the Manufacturers Empty Weight Multiplier.
- **Cost** – While not a direct factor in the rated aircraft cost, our team must operate within a budget. Therefore cost must come into play when considering figures of merit.
- **Lift/Drag Ratio** – As mentioned above, weight can be offset through other design options. The L/D ratio is one such FOM. It helps to reduce the rated aircraft cost by keeping the number of empennages and other aerodynamic surfaces low. Keeping the L/D ratio low also minimizes power loss due to drag, thus boosting the lifetime of our batteries.
- **Maneuverability** – As limited maneuvers are a part of the mission profile, the aircraft will be expected to perform such maneuvers in flight. This applies primarily to the flight score of the aircraft rather than the rated aircraft cost.

- **Payload** – A large payload boosts our flight score. In addition, the inability to carry a payload efficiently will result in more mass from support structure and a larger aircraft, which is detrimental to the flight score.

The next big step in the design process was to hammer out all the finer details: engine placement, vertical and horizontal stabilizer placement, hatch locations, and building materials and techniques. Because of the shape of the plane and the weight considerations, it was decided that a frame for the fuselage would be constructed of connected cross sections. While aluminum is common in full-scale planes, balsa wood is more commonly the choice for model airplanes. Balsa is relatively strong, ductile, and easy to work with as well as being substantially cheaper than aluminum. Aluminum, however, has a high strength to weight ratio and would still be used for some sections of the plane. The cross sections would be made out of lightweight birch plywood and later covered with a thin sheet of balsa, with hollow aluminum rods making up the internal connections within the frame. Semi monocogue construction is also common for wings, but instead a foam core approach was chosen. Crashes are relatively common for this type of plane, and wings are usually one of the first parts of the plane to suffer damage. The foam core approach is somewhat easier to construct, light in weight, and easier to replace. For some sections of the plane, such as the nose cone, the cross sectional approach falls short as well. Because of the high curvature, it would call for such small spacing between the cross sections that the technique looses its advantage. Instead, a third option needed to be found. Because this area should bear no load, several types of molded or formed pieces of material were considered including injection molded plastic, paper-mache, and fiber-glass. While the injection-molded plastic would work well, it is an expensive process for only one part, and was not readily available to us. Paper mache on the other hand was a technique that any member of the group could perform and could be molded around any shape we could carve out. The actual paper material, however, was not durable and would not handle any type of poor weather or working conditions. Finally, we settled upon the technique of fiberglass. The part may be created in a similar fashion to paper mache, but was much more durable.

4.0 Preliminary Design

In order to move from the best of the conceptual designs to one preliminary design many design considerations and decisions must be made. In this crucial step there is a balance between analyzing the decision more than the methods are worth and educated estimations. Because this design project is a complicated machine with countless parameters, we needed to simplify the parameter where it was reasonable to do so and assigning a system of Figures of Merit to certain aspects of the design in order to qualify the design decisions made.

The design scheme we had come up from the conceptual design has many aspects that were left to vary, which were configuration, dimensioning, and material selection. In order to deal with these decision categories we needed to consider them somewhat separate at first, although, they are highly coupled. As

the plane design took shape though we were able to iterate to a more finalized design. This approach to problem solving may not have been the best, because in many areas the interconnection of parameters was strong and through this method we had to be very careful we did not forget about these dependencies.

	Weight	Flying Wing		Dual Fuselage		Single Fuselage	
Complexity	0.3	-1	-0.3	-1	-0.3	1	0.3
Weight	0.2	0	0	-1	-0.2	0	0
Cost	0.15	-1	-0.15	0	0	1	0.15
L/D Ratio	0.1	1	0.1	0	0	0	0
Maneuverability	0.05	1	0.05	-1	-0.05	0	0
Payload	0.2	-1	-0.2	1	0.2	0	0
Total	1		-0.4		-0.35		0.45

Table 4-1: Figures of merit for general configuration.

4.1 Aerostructure

The configuration of the plane was mainly decided by the conceptual design and analysis in design parameters was mainly in dimensioning and material section. So in order to make some decisions we need to state what we have. We have a cargo type plane, high wing with a t-shaped tail with two engines hanging from the wings. This is a very simple and routine design, however, actual design and construction is anything but simple. In the area of aero-structures the method attack will focus on various components of the plane and the problems and questions that must be answered. These places include the storage section of the fuselage, the nose, the tail, the wings, and the landing gear. In answering important design questions about these components we will have a picture of the preliminary design.

4.1.1 The Fuselage

The fuselage up to this point is required to have a certain shape and that shape was decided to be dictated by two rows of softballs and space for batteries and other avionic equipment. The shape of the cross-section of the fuselage had to have the smallest area and still meet these requirements, so the result is a smooth triangular shape that is wider on top and tapers toward the bottom. In this section we could fit the batteries and other essentials. The materials were also an important factor. In order to make the fuselage have the strength to carry the load of the softballs and keep this shape we decided on a bulkhead type construction. This way we could construct the shapes of the interior holes and shape of the bulkheads any shape. With these conditions set we could vary two three dependent design parameters to optimize performance. Those are material of bulkheads, number of bulkheads, and spacing. If we combine the number of bulkheads and spacing into one parameter called the bulkhead spacing density, then we can find a relationship between it and the material parameter. Basically the as the material gets stronger we need less bulkheads. We first found a material whose properties, modulus of elasticity and density, where most favorable. We choose ¼" balsa plywood because of a high modulus, low density, and strength on two perpendicular axes. Because this is fairly strong wood we did not need a high bulkhead spacing density.

In order to connect these bulkheads we needed fairly strong connections. Because of the positioning of the balls and the landing gear it was decided to have one main tube toward the bottom of the fuselage between the rows of balls, and to have many smaller tubes serving as both rails and secondary support. We then decided to use aluminum tubes because of the added impact resistance and structural rigidity. The design parameters of the tubes are an outer diameter and an inner diameter. We wanted them to be as close as possible without going under a strength threshold. In order to provide added strength and a surface to monokote we decided to sheet the fuselage thin 1/32" balsa sheeting over most of the fuselage. However in the structurally crucial areas such as the landing gear and wing section and also the nose and tail connections we needed to allow for stronger connecting plates.

4.1.2 The Nose

The nose that will have to fit on this design had to be a very awkward shape because of the triangular shape of the fuselage cross-section. It has no axial symmetry, so a simple parabolic nose shape would be rather difficult to make. We had planned on making use of the CNC mills on campus to make a very accurate model from pine. Fiberglass would then be molded over this to create the actual nose. However, we soon realized that the complex geometry we wanted to have was simply beyond the memory limitations of our CNC mills. We therefore decided to make a first order approximation of a parabolic nose using a manual band saw. One main concern with this method is that we need to keep the surface finish of the fiberglass smooth. Also we need to devise a way that we could quickly take the nose cone off/on and remove and fill the plane with softballs. In order to do this, we decided to hold the cone on with rubber bands and slots and pegs around the diameter of the fuselage cross-section. This was beneficial because it is simple, it provided an inexpensive break away point, and it did not interfere with the aerodynamics.

4.1.3 The Tail

For the tail we could use the same attachment technique, because we were going to store the avionics and electronics in the tail and it needed to be easily accessible. This piece was to be built in two stages: 1) the wooded section 2) the fiberglass section. The wooden section was to be made with bulkheads, spars, and sheeting with a built in taper that section would then rigidly connect with the fiberglass end where the horizontal and vertical stabilizers were connected. The basic purpose of this section was to provide a smooth taper, hold the servos and radio, and to be as light as possible.

The second part of the tail had to hold the horizontal and vertical stabilizers, while holding a fairly complex shape. We could have decided to make this section out of balsa, but the stress concentrations at the joints of the stabilizers would be to great for the balsa wood. Also to make the stabilizers have a smooth, filleted connection with the body would be very difficult with balsa construction. We decided to use fiberglass and use the same construction techniques as we did for the nose.

Since the actual horizontal and vertical stabilizer do not have to carry as much stress we could use balsa construction to make the symmetric airfoil shape. We would cut airfoils of the appropriate

dimensions, then use balsa spars to hold them together. For the surface we decided to put monokote on the top.

4.1.4 The Wings

First of all we knew because the plane needed to be transported that the wings would need to be detachable. Also we decided that the wings needed to have the motors mounted to them. This configuration first yielded the concept of a fixed wing section that was centered over the fuselage that extended far enough to mount the motors, so they would be to the plane and the fixed portion would take the stresses and bending moments from the entire wing. Then into that we could slot the rest of the wing into it.

This method, while it provides a nice way of varying the strength where the wing connects to the fuselage, it led us to the difficult problem of connecting or slotting the wing extensions into the base section. We proposed many connection techniques, but they seemed to either be to bulky or to weak. In order to improve on this design configuration we decided to have the wing extend across the fuselage and make up one connected piece. The method of connection to the fuselage would be by a fiberglass collar around the underside of the fuselage, which extends under the wing out to the motors. Where the bottom is one piece that conforms smoothly to the wing and fuselage then on the top there is another fiberglass piece that fits on top, then bolts down. This configuration gives us the ability to extend the wing as one piece across the fuselage while still providing the strength out to the motor. It also gives us the ability to make any shape with the fiberglass. We decided to make the fiberglass make somewhat of a tapered fillet under the wing to make the stress concentrations at the joints continuous. This design calls for the fiberglass to go underneath the fuselage in order to cancel the tensions on either side of the wing. This is very beneficial because the landing gear, which is going to be place slightly back behind the quarter chord mark, will be extending out of the fuselage at that section. The landing gear was decided to extend from the main aluminum tube and out of the skin of the fuselage. In order to provide maximum strength for the landing gear in the moment about the center of the tube we could put reinforcing points in the skin of the aircraft. Those reinforcements would then be attached to the fiberglass section making them even stronger.

Now that we had a way of connecting the wing to the fuselage we needed to design the structure of the wing itself. The wing was to be one piece and have a NACA airfoil shape of 2412. By this point in the construction we knew what wing area we where going to need from the power requirements and an estimated time of flight. We also found that a good aspect ratio for our type of use would be around 6-8. These specifications told us that we needed to have a wing that was nearly equivalent to a 12' x 15" rectangle from the top view. We also determined that the maximum stress on a wing that is rectangular in its top view and is acted on by apposing point moments on the tips of the wings is at the center of the wing. The value of the maximum stresses or equivalent bending moments with respect to the distance away from the center falls away linearly. However if the cross sectional area of the wing increases toward

the center, the slope of this relationship will decrease. The wing will be supporting the same load, but there will not be such a high maximum stress in the center. Also with a tapered wing there are benefits in

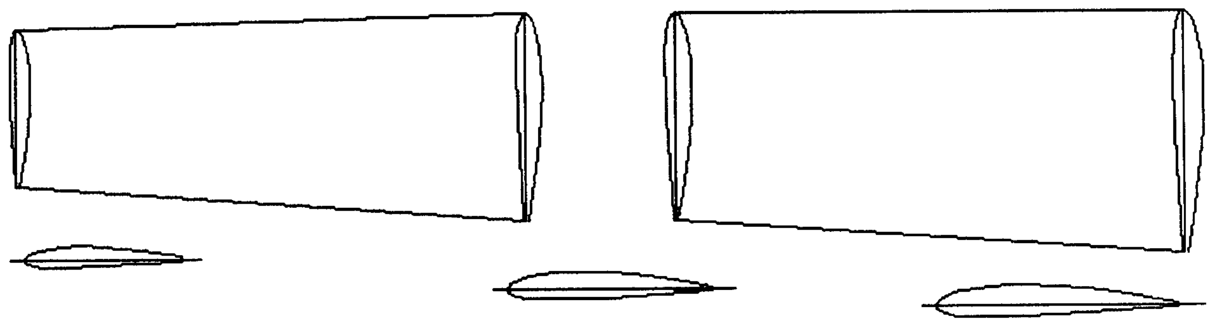


Figure 4-1: Wing planform and cross section

stability in yaw and benefits on the coefficient of drag on the wings. Toward the middle of the plane we need the leading edge of the wing to be relatively straight. In order to taper the wing area we tapered the trailing edge of the wing and keep the leading edge straight for the half of the wing, then toward the outside the both edges where tapered. In order to taper the NACA 2412 airfoil we created the airfoil shape then simply scaled the chord and consequentially all the dimensions of the airfoil. This gave us a very good idea of what the wing needed to look like, but now we needed to know how to manufacture that shape.

The complicated taper and airfoil could have been constructed with balsa airfoil cross-sections and balsa and/or aluminum wing tubes, however, that design for strength in wings is somewhat inefficient. This is due to the fact that the strength of the wing in this design comes from the inside the cross braces and airfoil shapes are on the inside. This is inefficient because the maximum normal stress on the beam occurs at the outer-most point on the beams cross-section or the point farthest from the cross-sections centroidal plane. This is where the support and the strength should be. This is accomplished by sheeting the wing. In order to sheet the balsa construction for strength much more weight is added to the already heavy interior. In order to make the wing's main support on the outside we needed a way of holding the shape of the wing on the interior that does not add to the weight and that is not required to support the bulk of the stresses. The ideal material for this application was Styrofoam, and to cut out these complicated shapes hot wires are used. After the shapes are cut we decided to use 1/16" balsa sheeting to cover the wing. Also to add strength in tension we planned to add lines from tip to tip tacked down above and below the wings. All of these design improvements go into making

the wing the strangest that it can be using the specifications as needed, because the test for holding the plane by its wing tips will require that our wings stand up to incredibly high stressed throughout the wing, and especially at the center of the wing.

Wing and Power Loading Requirements

To determine the wing loading we used preliminary estimates of our aircraft weight of 35lbs. This lets our aircraft takeoff at an airspeed on the order of 30mph. The resulting wing loading during takeoff is on the order of 35oz./in^2. This number is rather right, but certainly within reason for a cargo aircraft.

Power requirements at this stage were approximated using tables available on Astroflight Inc.'s website (http://www.astroflight.com/Contest.HTML). Extrapolating the available data for a 35lb, 12ft wingspan aircraft gives a takeoff power requirement of 824W. During cruise, where propulsive power goes into overcoming drag, the power requirements are only 158.7W.

5.0 Detailed Design

Airfoil Selection

In order to find the optimum airfoil for our aircraft, NACA four, five, and six digit series airfoils were sorted by (1) coefficient of lift at take-off without flaps (maximum angle of attack), (2) coefficient of lift at take-off with a single flap, (3) coefficient of drag at cruising (minimum drag), (4) coefficient of lift at cruising, and (5) the ratio of coefficient of lift over coefficient of drag at cruising in an *Excel* spreadsheet. These numbers were from the tables in Theory of Wing Sections by Ira Abbot and Albert Von Doenhoff (McGraw-Hill, 1949). These parameters were chosen to find the airfoil that would produce the most lift and the least drag at the lowest angle of attack.

The two most important parameters were coefficient of lift at take off without flaps and lift-to-drag ratio at cruising. Flaps were not considered for the design in order to simplify construction and save six hours on our MFHR, which would translate to a savings of $120 on our RAC. A high lift-to-drag ratio would allow the aircraft to lift its payload with smaller, more easily transported wings and less power needed to maintain level flight. The top ten in take-off lift and the top fifteen in lift-to-drag ratio are listed in Tables 1 and 2 respectively:

Table 5-1: Top Ten Airfoils by C_L Max Without Flaps

NACA nomenclature	C_L max clean	alpha (degrees)
63-210	1.8	14
4412	1.7	14
1412	1.6	15
2410	1.6	15
2412	1.6	16
2415	1.6	16
23012	1.6	16
23015	1.6	16
23018	1.6	16
64A410	1.6	14

Table 5-2: Top Fifteen Airfoils by C_L/C_D Ratio

NACA nomenclature	C_D min	C_L at C_D min	alpha	C_L/C_D
64_1A212	0.001	0.2	0	200
64A410	0.004	0.4	0	100
64_3-618	0.005	0.4	0	80
65_3-618	0.005	0.4	0	80
64-210	0.004	0.3	1	75
64_1-212	0.004	0.3	2	75
65-410	0.004	0.3	0	75
65_1-412	0.004	0.3	0	75
65_2-415	0.004	0.3	0	75
66_2-415	0.004	0.3	0	75
66_3-418	0.004	0.3	0	75
2408	0.006	0.4	2	66.66667
2410	0.006	0.4	1	66.66667
2412	0.006	0.4	2	66.66667
0010-35	0.003	0.2	2	66.66667

To provide a more practical visual comparison, the top ten airfoils in each of the five categories mentioned above were graphed comparing (1) cruising lift versus wing area, (2) cruising drag versus wing area, and (3) take-off speed versus wing area without flaps. The following formulas calculated the lift and drag:

(1) $F_l = 0.5\rho C_l V^2 A$

(2) $F_d = 0.5\rho C_d V^2 A$

For take-off, the lift force equaled 55lbf, the maximum allowable weight of the aircraft. The density of air at 2000ft, the average altitude of ground level in Kansas, was used for rho. According to Table A-26E in Fundamentals of Thermal-Fluid Sciences by Yunus Çengel and Robert Turner (McGraw-Hill, 2001) this value was 0.07210 lbm/ft^3. The cruising velocity was estimated at 45 ft/s, a rough value based on the length of the course and the maximum amount of time the aircraft had to complete the course. The wing plan form area was set ten square feet, two wings of 5 foot span times 1 foot chord. Similar airfoils had similar characteristics in these areas. Samples of these graphs are in Figures 1, 2, and 3 below:

Figure 5-1: Wing Area vs. Cruising lift for NACA 1412 to 2415 airfoils

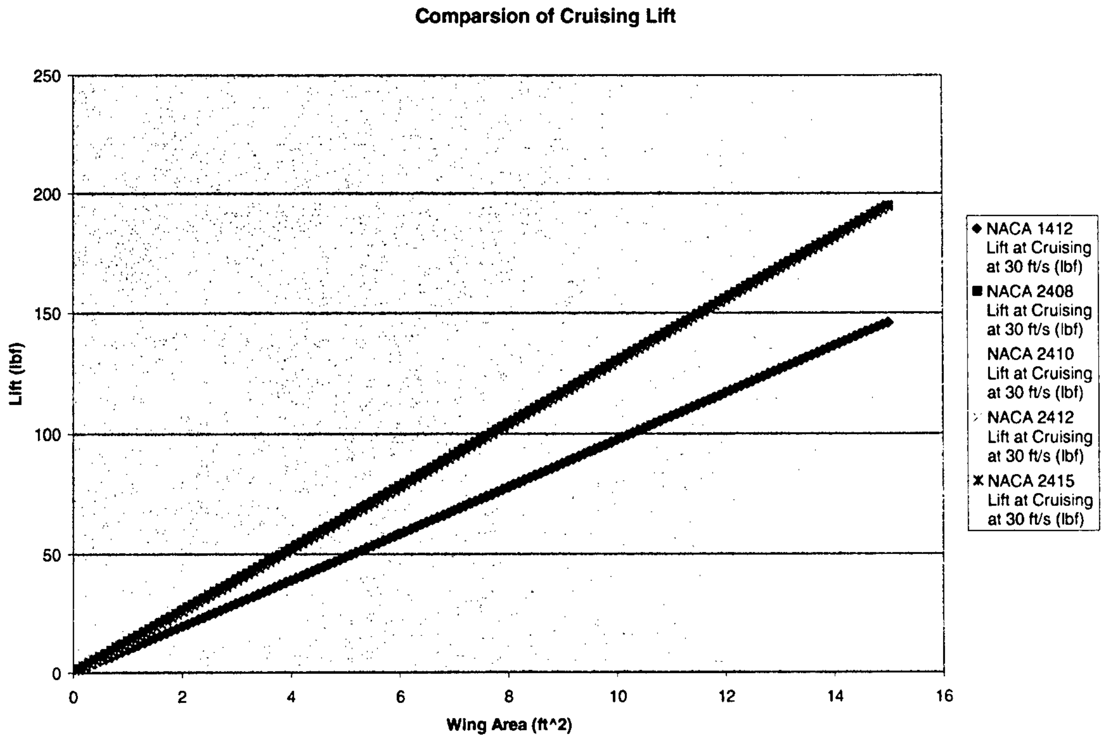


Figure 5-2: Cruising Drag vs. Wing Area for NACA 1412 to 2415 airfoils

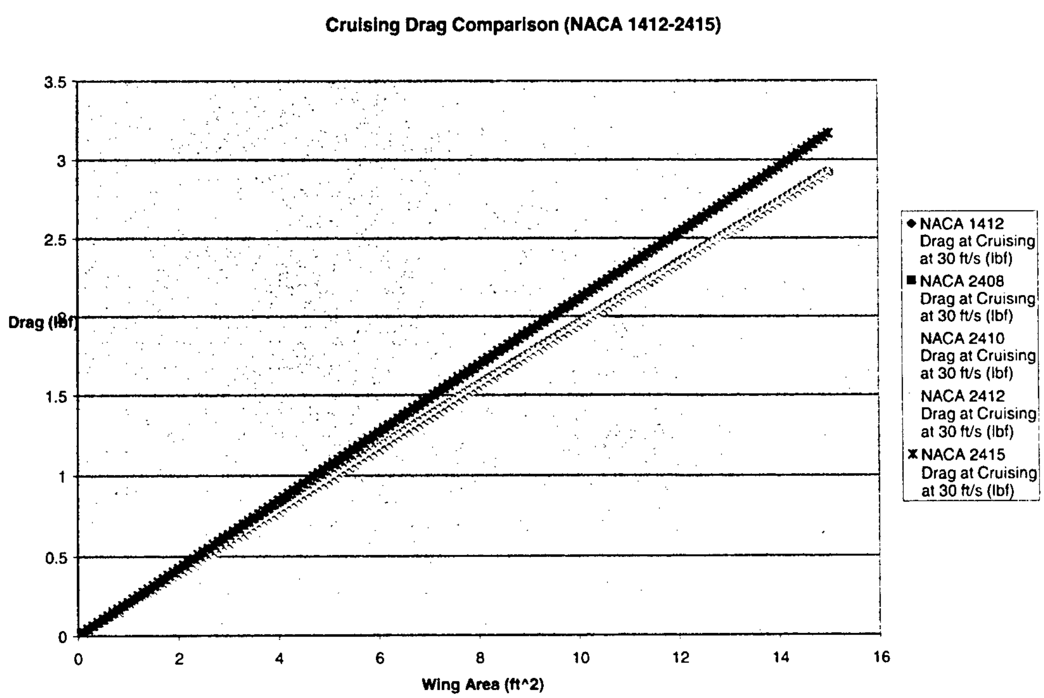
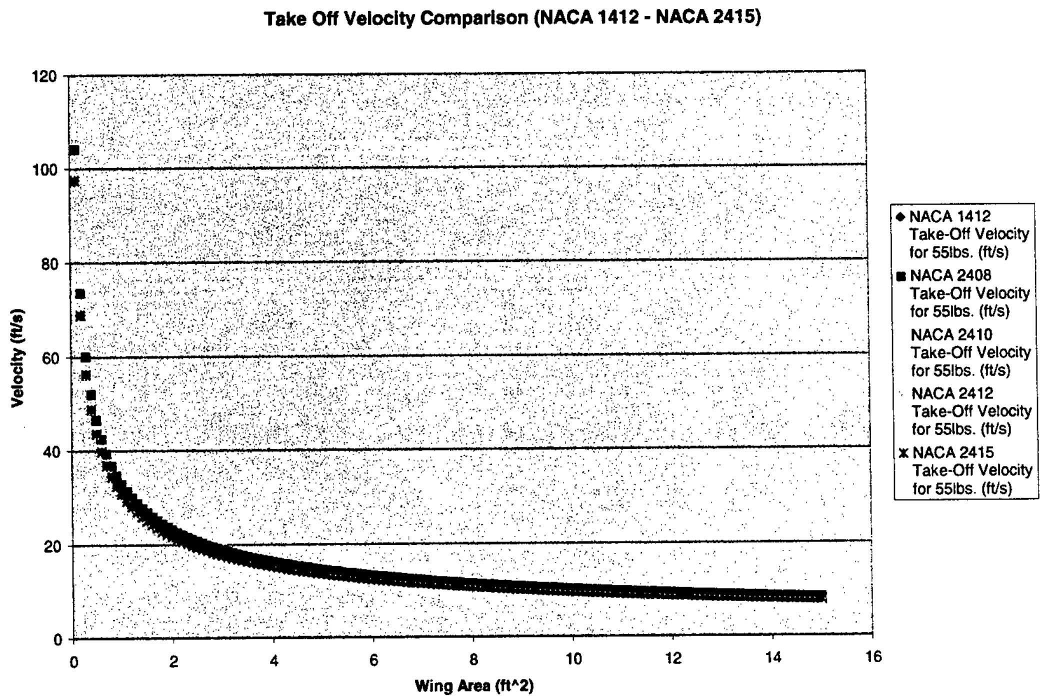


Figure 5-3: Wing Area vs. Take-Off Velocity for NACA 1412 to 2415 airfoils



The NACA 2412 airfoil was chosen for the wings of our airplane. The 24XX series of airfoils provided high lift at relatively low drag and similar performance across the range of angles of attack in the flight envelope. Although many five-digit and six-digit airfoils had better lift to drag ratios, the 2412 provided the one of the best overall lift coefficients. Also, our student Pro/Engineer drafter could easily interpret the numbers of the NACA four digit series to draft the airfoils, but he could not do so for the five and six digit series airfoils.

Propulsion System Design

The Learning Curve's propulsion system was designed to provide the highest performance possible given the restrictions imposed. Unlike most typical aircraft designs where propulsion, airframe, and various other subsystem sizing is done through iteration, our design was more or less built around a propulsion system that could provide the highest thrust to weight ratio possible for a flight time of about 10 minutes. The primary constraints on the propulsion system design are:

- Maximum current draw of 40A in motors and batteries
- Maximum battery weight of 5 lbs

Motor Selection

The preliminary selection of motors involved examining power to weight ratios of various brushed motors produced by Astroflight Inc. Figure 1, below, compares various models. Based on this information provided, the Astroflight 640S would offer the highest performance.

Circuit Design and Battery Selection

The propulsion batteries have been designed to last as long as possible given the restrictions of using no more than five pounds of NiCad batteries for the propulsion subsystem. Figure 2 shows the general layout of the circuit. The batteries feed provide power to the speed controller, which in turn, sends output current two the motors down two parallel paths. This is done to ensure that each motor is receiving the same signal, thus minimizing the problem of having two differing output powers.

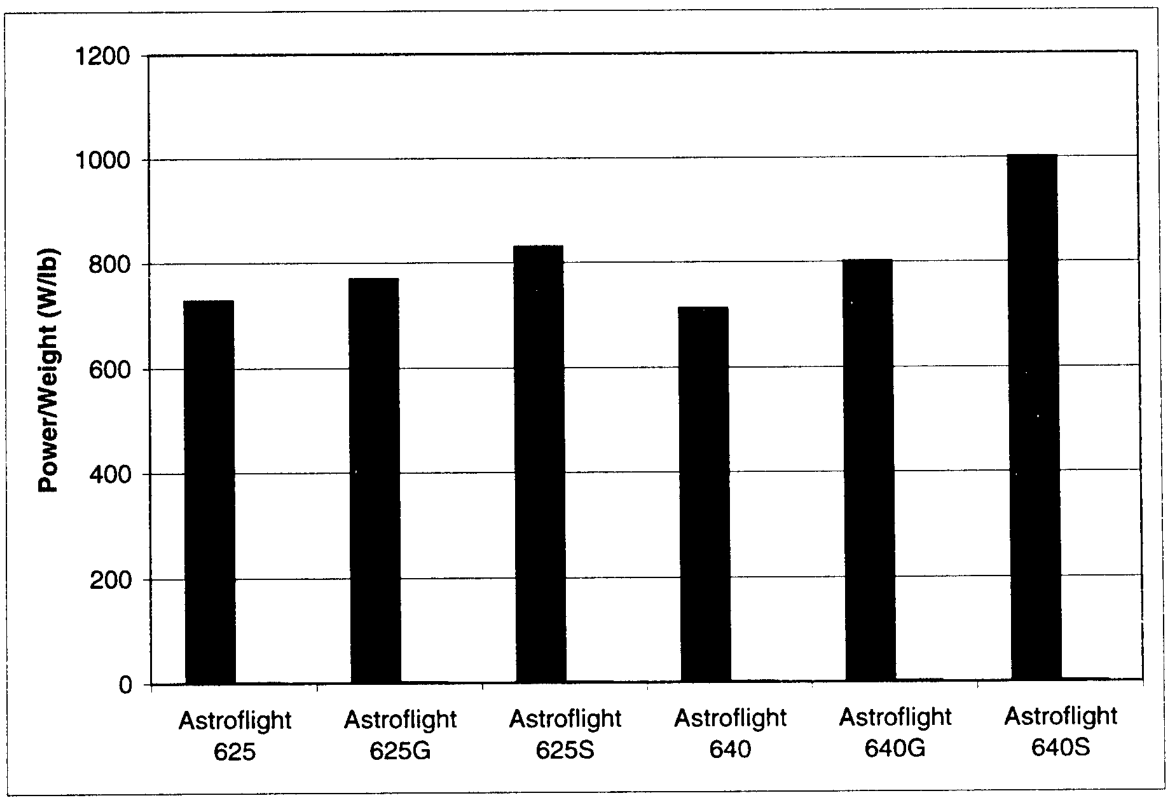


Figure 5-4: Power to Weight Ratios of Various Astroflight Motors

The Astroflight 640S, described in the previous section, requires 24V for peak performance. This requires we use twenty 1.2V cells in series per parallel path. The number of parallel paths, then, is simply a matter of battery weight. We found the optimal battery to be a 2400mAh cell, manufactured by SR Batteries. The batteries provide a current density to weight ratio of 1263mAh/oz, far exceeding any other battery we found. Using the figures from power loading, 824W for takeoff and 189W for cruise, we

calculated the per-charge lifetime of the batteries. Given 24V for our batteries and a lifetime of 2400mAh per cell, our circuit can provide takeoff power for approximately 4.1 minutes, and cruising power for 7.2 minutes.

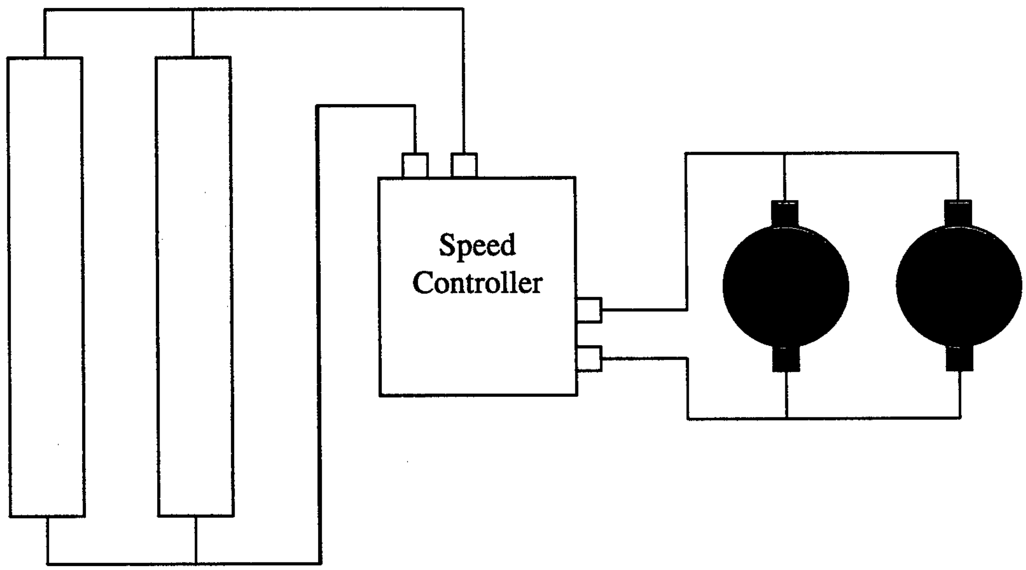


Figure 5-5: Propulsion System Circuit, showing batteries , speed controller, and motors

Tail – Vertical and Horizontal Stabilizers Once the conceptual design of the tail section was complete we could then start to quantify the design parameters of the selected configuration. We choose the upside down T – shape for the positioning of the vertical and horizontal stabilizers. The parameters that are of paramount concern are the dimensions of the stabilizers. The airfoil shape of the stabilizers was already determined to be symmetric, and the NACA number for the airfoil is NACA 0012. The horizontal stabilizer is used to stabilize the plane in pitch, meaning its area must produce a damping moment. This effect must be varied in magnitude for planes with different moments of inertia about the center of gravity in the pitch direction. The formula that we used to calculate the area of the horizontal stabilizer depends on the moment arm of the tail or the distance from the CG to the center of the stabilizer TMA, the mean aerodynamic chord of our wing MAC, and the total wing area WA.

$$HTA = (2.5*MAC*20\%xWA)/TMA$$

For our plane the numbers we used were TMA = 48 in., MAC = 15 in., and WA =2196 in^2, and this calculation yields a stabilizer area of 343.1 in^2. Then in order to determine the aspect ratio we used a common percentage of the aspect ratio of the wing around 83 %. The aspect ratio of the wing is 9.44, so

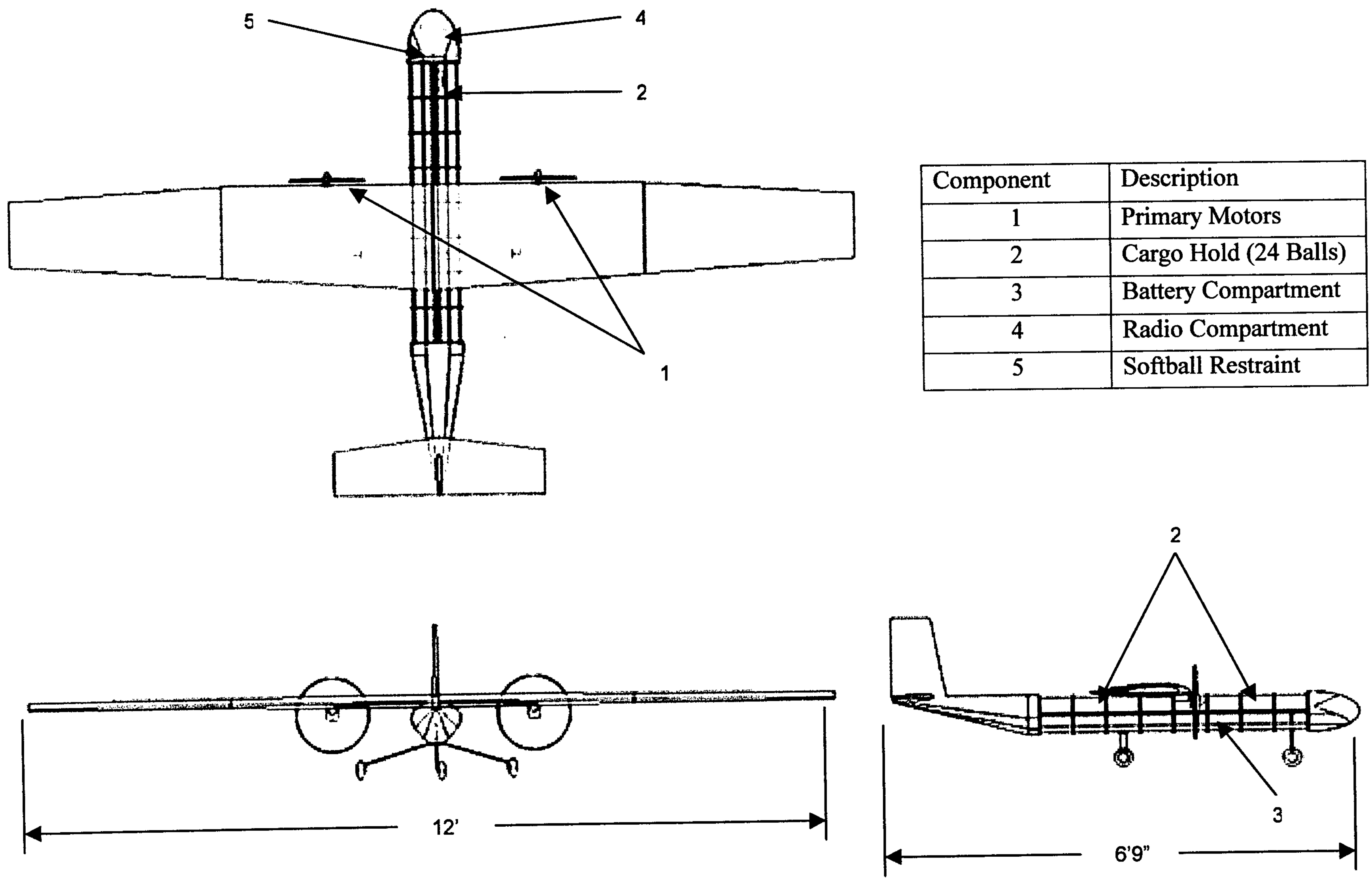
we used a horizontal stabilizer aspect ratio of 7.87. Using this ratio and the area we could find the length of the stabilizer, 40.2 in, which would make the mean aerodynamic chord of the stabilizer 8.53 in. Because we wanted to minimize the drag that this component produces we want to sweep back the leading edge. If it is swept back too much we will loose the aerodynamics of the stabilizer. The determining factor is the cruise speed of our aircraft, because our aircraft is expected to have a slower cruise speed we chose to only taper it by a ratio of root chord to tip chord of 1.4. This gives us the final dimensions of the horizontal stabilizer, where the root chord would be 9.83 in and the tip chord would be 7.02 in.

For the vertical stabilizer the area would mainly be dependent on a few variables, which are the moment arm from the CG to the tail and some measure of the size of the wings. The method that was very effective in stabilizing the yaw motion and in making the plane spirally stable was the idea of a center of lateral area. The center of lateral area is the centroid of the shape created from the lateral cross-sectional area. The method holds that the area of the vertical stabilizer should be such that the center of lateral area is at 25 % of the tail moment arm from the CG. This after doing some calculation and estimates of lateral area we determined that the area of the vertical stabilizer should be 119 in^2. Since it is common to have the root chords to be the same for the horizontal and vertical stabilizers we called the root chord to be 9.83 and the tip chord 7.02.

6.0 Manufacturing Plan and Processes

Construction of the fuselage for Learning Curve consists of birch plywood bulkheads, linked with thin aluminum spars. The unique shape of our fuselage would make for complicated and time-consuming manufacture. To eliminate this problem, we opted to use CNC (computer numeric control) mills to cut the complex shape of our fuselage bulkheads. After designing the bulkheads in Pro/Engineer™ they were imported to MasterCAM™. From there, cuts we're specified and the pieces were cut on a HAAS™ CNC mill using 3-axis machining. Instead of consuming a considerable amount of labor on a bandsaw, each piece took only 6 minutes to complete on average.

The construction of the tail section is semi monocogue, with birch plywood bulkheads and balsa supports. Cross sections will be cut on the CNC mill to save time. The loads on the tail do not justify the additional weight of aluminum supports. The fiberglass components (the nosecone and the wing root structure) will be constructed by layering fiberglass over molds.

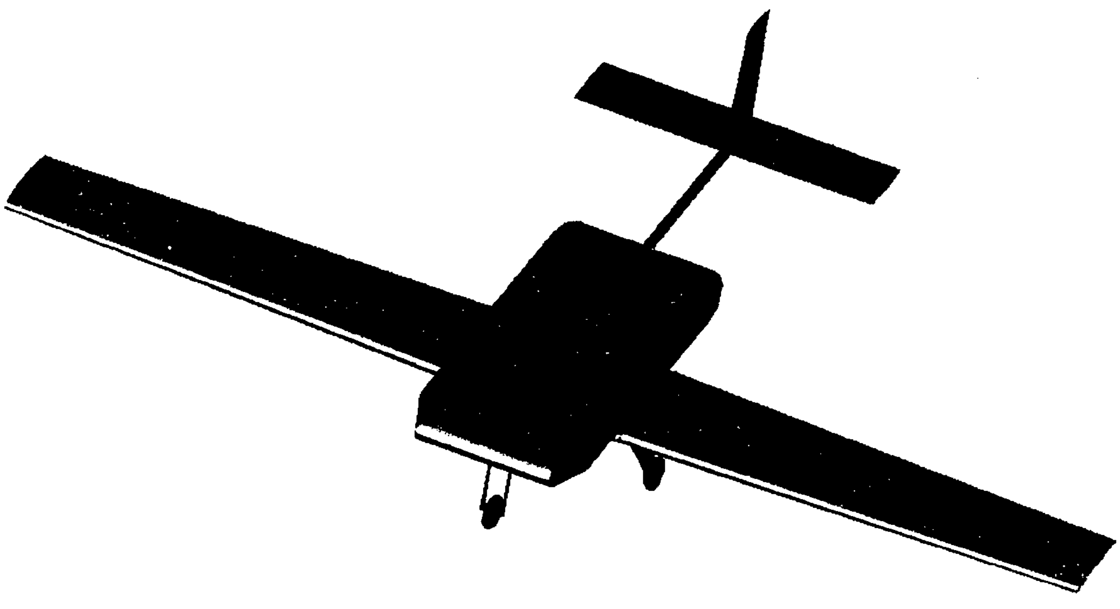


Component	Description
1	Primary Motors
2	Cargo Hold (24 Balls)
3	Battery Compartment
4	Radio Compartment
5	Softball Restraint

Figure 5.6-1: Basic 3-View

2001/2002 AIAA Foundation Cessna/ONR Student Design Build Fly Competition

Design Report



"Daedelus"

West Virginia University

March 2002

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Executive Summary

1.1 Competition Overview: A group of students from the School of Mechanical and Aerospace Engineering at West Virginia University have chosen to participate in a class called Design-Build-Fly. This class is intended to give students experience in the design and construction of small, unmanned, remote control aircraft. The aircraft is to be designed for this year's Cessna/ONR Student Design Build Fly Competition.

The airplane must be an electric, propeller driven aircraft that can fly with or without a selected payload, and be designed to complete three runs of a designated course. The course consists of three parts, each with two laps, for a total of six laps. The first and last two laps are to be flown with an empty payload. The third and forth laps are to be flown with a payload of softballs varying in number from 10 to 24. These six laps may be repeated as many times as possible within an allotted ten-minute time period. Any laps that exceed the ten minute time period results in a penalty.

The members of the West Virginia University team decided that the creation of two teams designing two different aircraft would be a good approach to this year's competition. This report follows the development of the second team known as "Daedelus." The primary goal of this team was to design an aircraft that would achieve the highest score possible in the competition. The score is calculated by multiplying a Written Report Score by a Total Flight Score and then dividing by a Rated Aircraft Cost.

Every year the competition varies in both the mission specifications as well as the payload that must be carried. The mission for this competition requires six laps around two posts placed 1000 feet apart. After the initial takeoff of the airplane, the pilot must navigate around the first post to the back side of the course. Once the plane is once again flying straight and level, the pilot makes a horizontal 360° turn. This occurs for laps 1, 2 and 5, 6. After the first two laps are completed the pilot is to land the airplane. At this time the payload is added and the pilot is to fly two more laps around the same course without the horizontal loop. Then the plane is to land at the end of the forth lap and the payload is removed. The last two laps are to be flown in the same manner as the first two laps. The payload for this year's competition is softballs. The plane must carry between 10 and 24 softballs.

The aircraft design must adhere to imposed constraints created by contest officiators that must be followed. These include 1) the fuselage must not be wider than 18 inches, 2) the plane must be fully electric with brushless motors and have no more than five pounds of batteries, 3) the takeoff distance must be less than 200 feet.

During the beginning stages of the conceptual design, the team chose to investigate many possible configurations. This was a critical step because of the experience and young age of the team. There is only one member of the team who is returning to the competition as well as

the only person with experience in airplane construction. The members of the team divided the design of the aircraft into several different categories: fuselage configuration, structure, payload, aerodynamics, propulsion, and electronics. Each category was investigated and several different concepts were chosen for group discussion. Each concept was then judged based on few basic criteria: aerodynamics, complexity, weight, ease of construction, and cost. A simple Microsoft EXCEL program was written to evaluate the effects of several airplane characteristics on the overall competition score. It was determined that the lack of experience warranted a simple design both in theory and construction. However, this did not mean that the design would necessarily be less competitive.

1.2 Design Tools: The design tools used to complete this aircraft were chosen based on the experience of the team leader and availability. The conceptual design phase had the team looking at Dr. Jan Roskam's series, *Airplane Design*, for insight into successful configuration selection. The computer aided design tools that were applied for this aircraft design are as follows:

- MATLAB
- Microsoft Excel
- AutoCAD

A program was created in MATLAB that was to perform two major functions. The first task was determining the dimensions of the aircraft, considering the selected planform of the plane that would provide the highest score. The second task of the program was to iterate certain design parameters until the parameter met certain design criteria.

A simple program was written utilizing Microsoft Excel. This spreadsheet was designed to generate an estimated construction time-table. From this time table, decisions were made as to the feasibility of each aircraft design feature. If the construction time was deemed too long or the construction to difficult, parameters were adjusted.

The computer program AutoCAD was used to verify the design of each aircraft component. This involved verification that that each component fit well with each other and that there were no unforeseen design flaws, before construction began.

2.1 Architecture of the Design Team: In order to compete in this year's Cessna/ONR Student Design/ Build/ Fly Competition, a team of underclassmen was assembled at West Virginia University. The team consisted of three seniors, one junior, and three sophomores. All seven are members of the dual degree program in mechanical and aerospace engineering. Due to the inexperience and small number of students, the approach the team used was for everyone to be involved with as many aspects of the design as possible. This was not always a feasible option and some members missed certain areas of the design and construction. Allowing all members of the team to experience all aspects of the design and construction of the airplane allowed each member to gain experience that is needed for future competitions. The first task to be accomplished was to designate a team leader. The obvious choice was Michael Plyler due to his previous experience (Design/Build/Fly 2000-2001). It has been his task to bring the new students up to speed on principles and practices of the design, construction, and flight of remote controlled aircraft. After choosing a team leader, a treasurer, and secretary were named to keep track of funds and important items of business, respectively. After deciding a hierarchy of leadership, goals were set and recorded.

2.2 Design Team Personnel Assignments: The different design and construction tasks were split amongst the seven members of the team. Table 1 displays the members of the team and the areas that they participated in. A member of WVU's first team was put in charge of material acquisition.

Table 1: Group Assignments

Fuselage / Payload	Structures	Aerodynamics	Propulsion
Seun Babatunde	Seun Babatunde	Seun Babatunde	Michael Plyler
William McCartney	Brian Jones	William McCartney	Sommer Roach
Michael Plyler	Michael Plyler	Michael Plyler	
Sommer Roach	Aaron Shinn	Aaron Shinn	
Brianne Williams	Brianne Williams		

During the first semester the group met once a week to work on the project at hand; the second semester the group met twice a week. The extra meetings were needed to stay on schedule as the construction of the plane. At each meeting, goals that needed to be accomplished were set, and members of the team were given specific tasks to accomplish. There were some items that required extra time, and had to be completed outside of the

designated meeting times. To overcome these hurdles a few individuals put in the extra hours and accomplished the tasks for the team to remain on schedule.

Table 2 shows a layout of participation on the design by the team, a scale of 3 being the highest level of participation and 0 meaning no participation. Team members compiled all necessary documentation on the plane to use in the report. Sketches were collected and translated into a drafting package. The team was well lead and organized. The goals set forth in the initial meetings were accomplished.

Table 2: Members of Team Daedulus Participation ranking

Member	Design			Construction				Testing	Report
0 –least; 3 –most	Fuselage	Wing	Tail	Fuselage	Wing	Tail	Internal	Flight	
Aaron Shinn	2	3	1	3	3	2	2	2	0
Brian Jones	1	2	2	2	2	2	2	2	0
Brianne Williams	2	1	1	3	3	3	3	2	0
Michael Plyler	3	3	3	3	3	3	2	2	3
Seun Babatunde	1	2	2	3	3	3	3	2	0
Sommer Roach	2	2	2	3	3	3	3	2	3
William McCartney	2	1	2	2	2	3	3	2	3

A very tentative time schedule was created during the beginning of the planning phase of the aircraft. After the planning phase was completed and the general layout of the aircraft was determined, a more detailed plan of the design was determined. Figure 1 on the following page shows the time schedule that was chosen along with the actual time schedule that was followed. As can be seen, the two time schedules were very different.

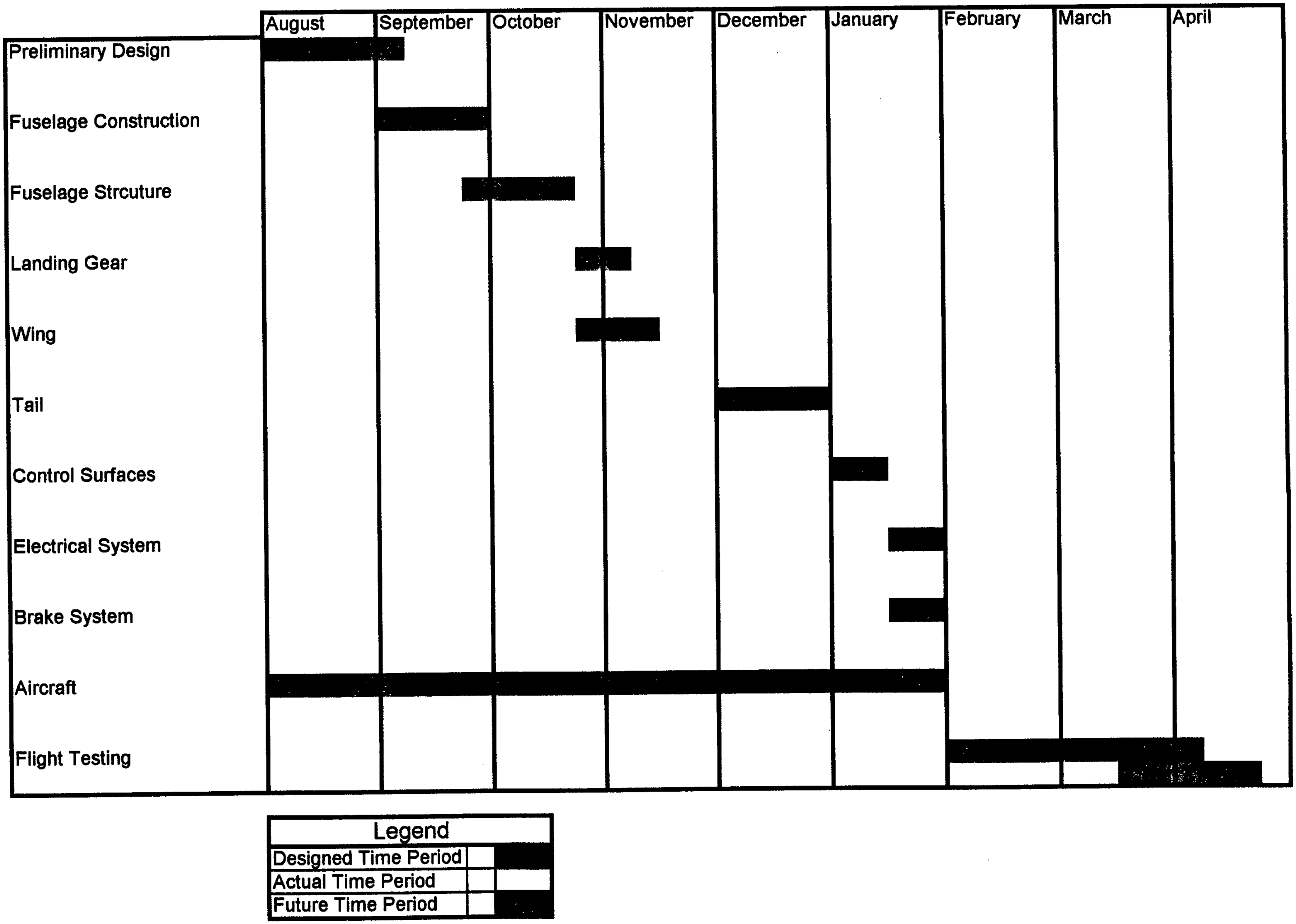


Figure 1: Time Schedule for the Construction of the Airplane

Team Daedulus is a very inexperienced team with only one member as a returning veteran of the Design Build Fly Competition. This lack of experience meant that the new students were required to do a considerable amount of research. They were given a set of aircraft design parameters that they were to investigate and come to some conclusions. These conclusions were designed to give the students experience in examining a given set of design criteria and then coming to a design decision. A majority of the time, the new students lacked the knowledge and experience with actual construction. All the academic knowledge does not replace construction experience.

After the students had completed a week-long research project on the design requirements, the team leader, Michael Plyler, listened to the team's suggestions. Taking these suggestions into consideration, a few ideas for each design component were chosen for further analysis.

The contest rules were more specific for the aircraft configuration than in past years. The fuselage was not to be any wider than 18 inches and the softballs may be carried in only a few limited configurations. It was important to incorporate these restrictions into the overall design process.

3.1 Fuselage: There were two fuselage configurations that were considered. The first configuration was to carry 24 softballs in a 2 column X 12 row configuration. The second option was to carry the same amount of softballs but in a 3 X 8 configuration. The latter configuration was chosen because WVU Team One had chosen a configuration of 2 X 12. It was important to the members of Team Daedulus that the aircraft designed by both WVU teams not be identical.

There were four options that were considered for the construction for the fuselage. The first was to make a balsa wood build-up airframe. The second option was to create a fuselage made of foam with balsa sheeting on the external surface. The third option was to design a composite fuselage that required the construction of a mold. The final option was to create a composite fuselage with a different technique, which would require the same steps involved in making a mold, but would end at the construction of a plug. The plug would be made of solid foam insulation and covered with multiple layers of carbon fiber and fiberglass. The foam would then be removed by pouring acetone over the foam, which would dissolve the foam leaving a composite fuselage.

3.2 Wing: The only wing configuration that was considered for this design was a mono-wing. It was immediately determined that a bi-plane would not be necessary because there was no wing span limitation. There were two wing planforms considered. The first was to use a basic

rectangular wing with no sweep and no taper. The second option for the wing was a relatively high tapered wing with no sweep.

3.3 Tail: There were three tail configurations chosen for consideration. The first tail configuration that was considered was a conventional configuration with a single horizontal tail and single vertical tail. The second configuration that was considered was a T-tail. The final configuration that was considered was a V-tail.

3.4 Landing Gear: Several options were considered for the landing gear. A standard fixed landing gear system with two main landing gear and a single nose gear was considered. The same standard configuration was considered with the addition of a retracting system.

3.5 Control Surfaces: The control surfaces for the vertical and horizontal tail were easily chosen to be conventional surfaces that span the entire distance of each component. The wing control surfaces had the option of using two different configurations. The first option is the most common option of most airplane designs. This option utilizes ailerons and flaps. However, a second option and a more common option for small remote controlled aircraft is to have only one control surface on each wing panel. These are known as flaperons.

3.6 Power and Propulsion: The rules of the competition have limited the teams to only five pounds of NiCad batteries and the motors must be from the Graupner or Astro Flight families of brushed electric motors. The propeller and motor configuration for this aircraft was chosen to be a dual engine airplane with the propellers mounted on the fuselage.

Final Design

3.1 Fuselage: The selected fuselage was a configuration carrying 18 balls with a lay-out of 3 X 6. The method for construction of the fuselage was chosen to be a composite structure. To create the composite structure, a plug of the fuselage was made out of foam and covered with the necessary composite materials. The foam was removed by using acetone to dissolve the foam, as mentioned above.

The dimensions of the fuselage were dictated by several factors. It had to be able to carry the 18 softballs as well as have enough volume to fit all the other necessities. There had to be room for the batteries, front landing gear, servos, and the mechanisms for the braking system. The dimensions of the fuselage were set at 17.5 inches for the width, 4 inches for the thickness, and 39 inches for the length.

3.2 Wing: The configuration that was chosen for the wing was a mono-wing. A compromise was made for the wing planform. For small model aircraft the wing that inherently has the best stability is a common rectangular wing. However, this aircraft design borders on the small aircraft and large aircraft. It truly is neither a small nor a large aircraft. Large aircraft typically have tapered wings because a large taper angle drastically reduces drag. It was

decided that a wing that had a small taper ratio would meet the requirements for this design. The dimensions of the wing can be found in Table 3.

Table 3: Wing Dimensions and Characteristics

Characteristic	Dimension
Span	9 ft
Root Chord	13.4982 in
Tip Chord	9.4968 in
Area	1239.6 in^2
AR	9.41
Taper Ratio	.7

3.3 Tail: The tail that was chosen was the conventional tail with a rectangular horizontal tail and a swept vertical tail. It was chosen for many reasons. The tail that offers the best aerodynamics is the V-tail. However, the V-tail could not be used because it would require the airplane to have five servos as compared to four servos for the conventional tail. This extra servo drastically impacts the contest score. The V-tail would also add an additional vertical surface which would also hurt the score. The competition rules limit the span of the tail to 25 percent of the largest span of the plane. This makes the tail span 27 inches. The area of the tail was set at 25 percent of the wing area. This made the tail area and the chord 309.9 in^2 and 11.5 inches respectively. The dimensions and characteristics of the tail can be found in Table 4.

Table 4: Tail Dimensions and Characteristics

Characteristic	Dimension
Span	27 in
Chord	11.5 in
Area	309.9 in^2
AR	2.35
Taper Ratio	1

3.4 Landing Gear: It was decided that the small reduction in drag due to the addition of a retracting system would not warrant the weight addition. Therefore the team chose to make fixed main gear out of carbon fiber. A composite main gear gave the benefit of strength and aerodynamics. The nose gear is a standard remote control system.

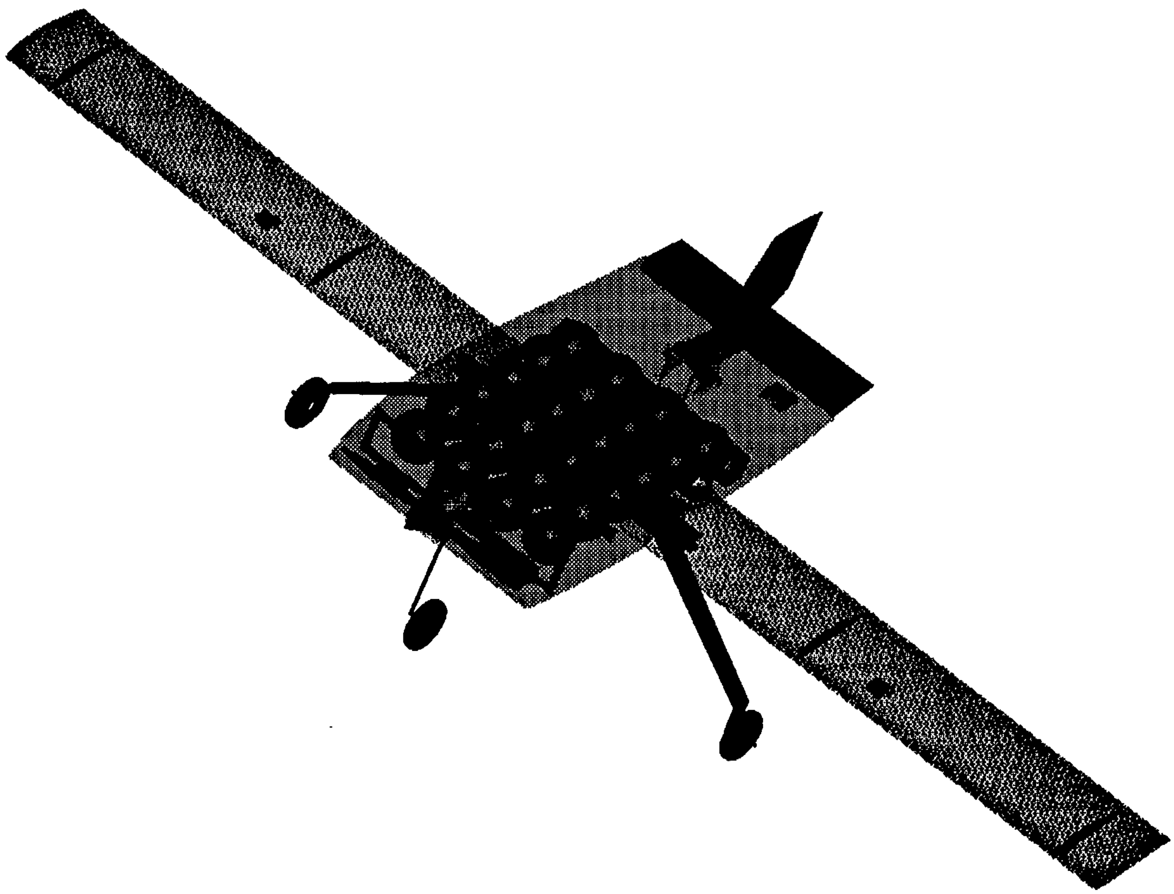
3.5 Control Surfaces: It was decided that a conventional control surface system for the tail would be used. It was also decided that in order to avoid any control problems, the tail control surfaces would be oversized by 15 percent. The wing control surfaces were designed as flaperons.

3.6 Power and Propulsion: It was decided that all five pounds of NiCad batteries would be necessary. The number of propellers was set at two.

CESSNA/ONR Design/Build/Fly

Student Competition

Proposal Report



Oklahoma State University

Black Team

March 12, 2002

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1 EXECUTIVE SUMMARY

This report details the Oklahoma State University Black Team's methodology used to develop a successful aircraft design for the 2002 Design/Build/Fly competition. This year's competition required a team of students to design, fabricate, and compete an unmanned remotely controlled aircraft in the contest held in Wichita, Kansas. The design process that was followed to develop a competitive design involved three phases. These phases of development, which included conceptual design phase, preliminary design phase, and the detail design phase, are summarized below.

The Black team began its design development by dividing into three groups: Aerodynamics, Structures, and Propulsion. Each of these groups began evaluating this year's mission requirements and weighing out the crucial constraints set forth by the rules and regulations. After understanding the rules, the groups began investigating possible design alternatives for all aspects of the design. Concepts were then compared to the rules of the contest and rated aircraft cost (RAC).

The goal of the conceptual design phase was to choose a final aircraft configuration, so many alternative configurations were investigated. The alternatives investigated encompassed the following areas of design: planform configuration, wing location, empennage configuration, landing gear type, payload configuration, material type, and propulsion system configuration. The range of planform configurations investigated included Bi-Wing, Canard, Dual-Fuselage, Conventional, Flying Wing, and hybrids of each. Possible wing locations were determined to be high wing, mid wing, and low wing. Empennage configurations considered were V-tail, T-tail, conventional, cruciform, and no tail. The landing gear types considered were oleo type, and bow type. For the payload, two ball, three ball, and four ball abreast configurations were reflected on along with mixtures of each. These payload configurations then determined approximate shapes of possible fuselages. Construction materials that were compared were carbon fiber/foam and balsa wood/MonoKote. Propulsion systems investigated were one propeller with one motor, one motor with two propellers, and two motors with two propellers.

The most valuable step taken towards the development of a design by the black team was the adaptation and use of an optimization code that iterated an optimal score within the contest parameters using Mathcad. The use of this code was a responsibility of the Aerodynamics group. The program took into account all the contest rules and the rules of flight to find the performance needed for an optimal score in the competition. The code assisted the team in the conceptual design phase by allowing the aerodynamics group to obtain a feasible estimate of a predicted contest score for any configuration of interest. After this was done for the planform configurations mentioned above and their hybrids, the team determined that a modified flying wing design had the highest scoring potential and therefore, was the optimal design for this particular contest.

The modified flying wing, or ModWing, is a hybrid between a pure flying wing and a conventional plane. Its "fuselage," or body wing, is a reflexed trailing edge flying wing airfoil. Its characteristics counteract the negative pitching moment induced by the rectangular main wing that discretely joins to it. Because of this stabilizing characteristic of the body wing, a much smaller body length was possible, which significantly decreased the RAC and allowed for the maximum abreast softball count without increasing form drag like a typical wide-bodied fuselage would. The body flap, or elevon, was implemented, which was penalized much less in the RAC than a horizontal stabilizer.

Throughout the preliminary design phase, the program was modified to include updated and more accurate information provided by each group. The propulsion group developed a code that allowed them to analyze a range of motors, propellers, and batteries and thus be able to return useful data to the aerodynamics group for the optimization program. The structures group developed a useful weight model for the optimization code that took wing area and chord length into account. After these improvements were complete, the aerodynamics group determined the necessary body wing and main wing airfoils, wingspan and chord length. They also found the static stability and the control surface sizes.
The structures group completed stress and load calculations and also finished technical drawings necessary to construct the prototype.

During the detail design phase, the manufacturing processes for prototype construction were determined and construction of the prototype commenced. With the prototype built and flown, unexpected issues could be worked out and addressed for the fabrication of the final aircraft. The aerodynamics group finished the dynamic stability calculations and finalized predicted aircraft performance. In addition, the propulsion team used a dynometer to select an ideal propeller for prototype test flights according to the current characteristics of the aircraft.

The final design includes an AstroFlight FAI-25 motor powered by fifteen cells of SR2400 Nickel-Cadmium batteries. The body wing airfoil is an Eppler 326 reflexed trailing edge flying wing airfoil, whereas the main wing has a Selig-Donavon 7062 high lift, low Reynolds number airfoil with two degrees of dihedral. The ailerons, the rudder and the nose gear are powered by Hitec 85MG Mighty Micro servos rated at 49 oz-in and the elevon is powered by a Hitec 225MG Mini servo rated at 67 oz-in torque. The vertical stabilizer's airfoil is made with a NACA 0009. The main wing and the vertical stabilizer are build out of a foam core and carbon fiber; the body wing and the landing gear are built with honeycomb core and carbon fiber.

The predicted mission performance for the unloaded sorties is a takeoff distance of 34 feet and a cruise velocity of 83 ft/s. Similarly, for the loaded sorties a takeoff distance of 168 feet and a cruise velocity of 75 ft/s is to be expected. The total projected time to complete one mission is 5 minutes and 42 seconds.

2 MANAGEMENT SUMMARY

In the Spring of 2002, a team of Oklahoma State University students was assembled with the intention of competing in the AIAA/Cessna/ONR Design/Build/Fly Competition. The team consisted of seventeen seniors of which fourteen were aerospace engineering students and three were mechanical engineering students.

2.1 Architecture of the Design Team

Led by the chief engineer, three technical groups were formed to best suit this year's challenge. These groups are aerodynamics, structures, and propulsion. These divisions were created to capitalize on each student's varied interests and technical background, which in turn improved the efficiency of the team. Each appointed group leader reported to the chief engineer, who facilitated communication between the groups and assigned responsibilities to each group. Figure 2.1 illustrates the team architecture.

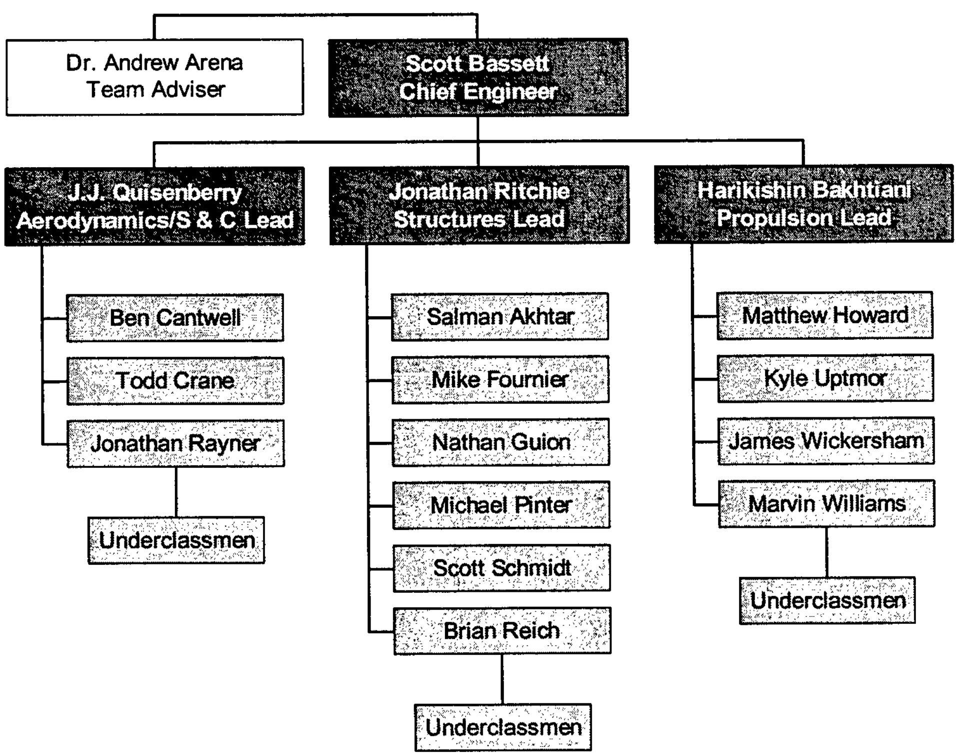


Figure 2-1: Architecture Chart for OSU Black Team

2.2 Design Personnel and Assignment Areas

The aerodynamics group primarily deals with the configuration of the plane. It develops and refines the optimization program throughout all phases of design. The group is responsible for the external

configuration, including wing airfoil selection. This group is also responsible for the sizing of the aircraft's surfaces, including wing area, aspect ratio, empennage sizing. They will also ensure that the stability and control of the plane is perfected.

The structures team is primarily responsible for the construction of the prototype and final aircraft. They first select the construction method and material to be used. They will also provide the CAD drawings of the internal and external layout of the aircraft since these are necessary to the construction of an accurately built aircraft. They are also responsible of the internal layout of servo placement, payload placement, etc. To some extent the aerodynamics group will need to be the final word on where the payload must be placed to ensure a symmetric loading about the center of gravity since this year's competition includes flying with and without payload. They will also conduct the stress analysis needed for the aircraft.

The propulsion group is responsible for selection of the propulsion system after testing and computing variations of systems. They will work closely with the aerodynamics group to make the optimization program more precise. They are responsible for the selection of batteries, motor, propeller, and gearbox. Since the propulsion system is very expensive and can be easily receive permanent damage, they are also responsible for the maintenance of the system.

Team Member Name	Title	Assignment Areas
Scott Bassett	Chief Engineer	Overlook all Aspects of Design
J.J. Quisenberry	Aerodynamics Leader	Optimization/Stability Calcs
Ben Cantwell		Airfoil Selection
Todd Crane		Control Surfaces
Jonathan Rayner		Optimization/Sizing Surfaces
Jonathan Ritchie	Structures Leader	Wing Construction/Organization
Salman Akhtar		Construction/Jigs
Mike Fournier		Landing Gear/Brake Design
Nathan Guion		Internal Layout/Servo Placement
Michael Pinter		Structural Analysis/Construction
Scott Schmidt		Construction/Jigs
Brian Reich		CAD Drawing/Foam Cutting
Harikishin Bakhtiani	Propulsion Leader	Motor/Prop/Battery Selection
Matthew Howard		Propeller Selection
Kyle Uptmor		Battery Selection
James Wickersham		Motor Selection
Marvin Williams		Motor Selection

Table 2-1: Team Organization

2.3 Management Structures

The design team meets regularly twice a week as an entire group to allow the chief engineer and the advisor to shed light on any pertinent issues that require the immediate attention of the team. The majority of the brainstorming needed for the conceptual design phase was performed in these meetings. At least once a week, the chief engineer and group leads would meet to discuss progress and the latest developments in each group. The group leaders would then communicate to each of their groups these topics as to remain on schedule. Each technical group meets separately during the week at least once a week. Any change in the internal or external configuration is discussed during these meetings so that each group is aware of the developments of the others.

2.4 Milestone Chart

Since there were more than 30 people involved in this project and they were split up into three different groups, the chief engineer had to ensure that the schedule was followed closely to make any progress at all. To be effective, the chief engineer developed a milestone chart seen in Figure 2-2 to have a visible plan for all to see with deadlines to be met. This chart shows the actual and predicted timing of the major events in the aircraft design.

ID	Task Name	Start	Finish
1	**Conceptual Design Phase**	**Tue 1/8/02**	**Wed 2/13/02**
2	Conceptual Sketches	Mon 1/14/02	Fri 1/18/02
3	Conceptual Sketches (Actual)	Tue 1/15/02	Fri 1/18/02
4	Initial Optimization	Mon 1/14/02	Fri 2/1/02
5	Initial Optimization (Actual)	Tue 1/8/02	Wed 2/13/02
6	1/4-Scale Model Construction and Testing	Fri 2/1/02	Sat 2/2/02
7	1/4-Scale Model Construction and Testing (Actual)	Sun 2/3/02	Mon 2/4/02
8	Configuration Freeze	Mon 2/4/02	Mon 2/4/02
9	**Preliminary Design Phase**	**Tue 2/5/02**	**Thu 3/7/02**
10	Prelim. Tech. Drawings	Tue 2/5/02	Mon 2/18/02
11	Prelim. Tech. Drawings (Actual)	Mon 2/11/02	Mon 3/4/02
12	Wind Tunnel Model Construction	Wed 2/6/02	Sat 2/16/02
13	Wind Tunnel Model (Actual)	Tue 2/12/02	Mon 2/25/02
14	Wind Tunnel Tests	Sat 2/16/02	Mon 2/18/02
15	Wind Tunnel Tests (Actual)	Fri 2/22/02	Tue 2/26/02
16	Purchase Propulsion Parts	Tue 2/5/02	Mon 2/18/02
17	Purchase Propulsion Parts (Actual)	Fri 2/15/02	Wed 2/27/02
18	Purchase Construction Materials	Tue 2/5/02	Mon 2/18/02
19	Purchase Construction Mat. (Actual)	Fri 2/8/02	Thu 3/7/02
20	**Detail Design Phase**	**Mon 2/18/02**	**Wed 4/24/02**
21	Prototype Construction	Mon 2/18/02	Tue 3/5/02
22	Prototype Construction (Actual)	Wed 2/20/02	Sun 3/10/02
23	Prototype Flight Testing	Thu 3/7/02	Sun 3/24/02
24	Prototype Flight Testijng (Actual)	Sun 3/10/02	Sun 3/24/02
25	Final Design Freeze	Wed 3/27/02	Wed 3/27/02
26	Final Aircraft Construction	Mon 3/25/02	Fri 4/5/02
27	Flight Testing of Final	Sat 4/6/02	Mon 4/22/02
28	Flight Evaluation	Thu 4/18/02	Mon 4/22/02
29	Final Presentation to Public	Wed 4/24/02	Wed 4/24/02
30	**Documentation of Design**	**Fri 2/1/02**	**Tue 3/12/02**
31	Proposal Preperation	Fri 2/1/02	Wed 2/27/02
32	Proposal Preperation (Actual)	Fri 2/1/02	Wed 2/27/02
33	Proposal Draft Deadline	Wed 2/27/02	Wed 2/27/02
34	Proposal Draft Revision	Fri 3/1/02	Mon 3/11/02
35	Proposal Draft Revision (Actual)	Sat 3/2/02	Mon 3/11/02
36	**Proposal to AIAA**	**Tue 3/12/02**	**Tue 3/12/02**
37	**Competition**	Fri 4/26/02	Sun 4/28/02

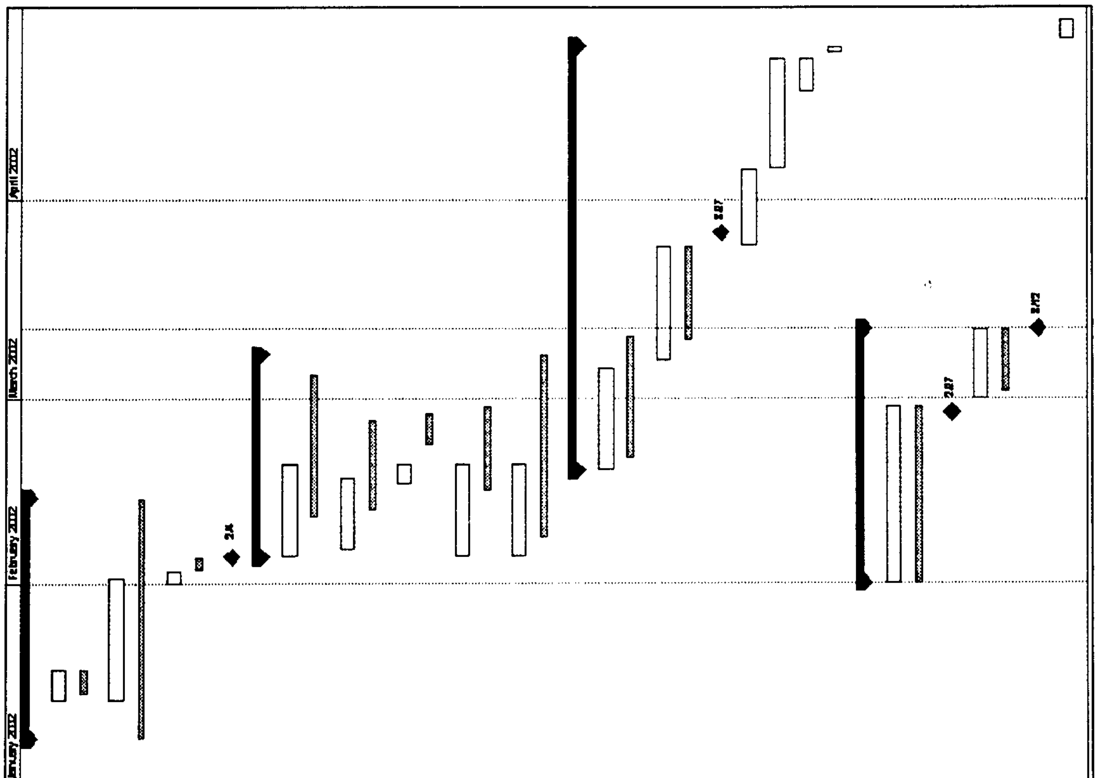


Figure 2-2: Milestone Chart

3 CONCEPTUAL DESIGN

During the conceptual phase, the general design was settled upon. To attain the concept with highest scoring potential, many aspects of the mission and the parameters that affect them were considered. In the conceptual phase every leg of the mission was analyzed. From this analysis, design parameters and their importance were attained. Then the design possibilities were narrowed and compared using various figures of merit. Upon completion of the evaluation a final design was chosen.

3.1 Mission Analysis

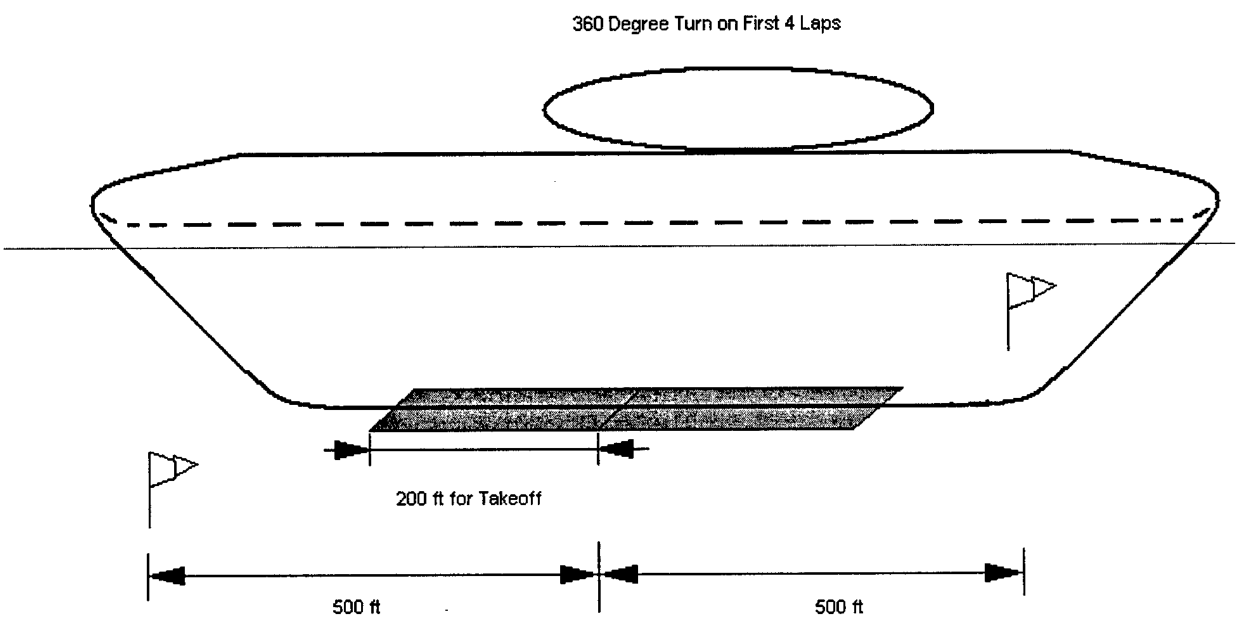


Figure 3-1: Flight Profile

The mission profile for this year's AIAA Design/Build/Fly contest was to design an aircraft to fly three sorties in each of the five allotted missions. The best three mission scores are summed to get the total flight score. The mission consists of three sorties, an empty sortie making two laps each with a 360° turn, a loaded sortie making two laps each with a 360° turn, and an empty sortie making two laps with no 360° turns. Each sortie begins with the takeoff of the aircraft. The takeoff is limited to 200 feet. Upon leaving the runway, the plane must climb to cruise altitude. The plane will cruise until it reaches either the upwind or downwind pylon upon which it will make a 180° turn. Both pylons are 500 feet from the starting line. At the end of the cruise portion of the flight the plane must slow and descend for landing. When the plane touches down it must roll to a complete stop before being attended to. Once stopped, the plane must be

loaded or unloaded and set for the next leg of the mission by the support crew. Each mission must be completed in 10 minutes or less.

3.1.1 Pre-Conceptual

In order to truly generate an appropriate aircraft design for this year's competition, the aerodynamics group developed a flight analysis program to understand the characteristic patterns in the contest. The program was designed with the contest rules in mind to optimize score. The score this year is dependent on Flight Score and Rated Aircraft Cost (RAC). The RAC represents the technology level and size of the aircraft by multiplying characteristics of the aircraft by values that simulate the cost and time to build the aircraft. The objective of the program was to observe not only how altering the parameters of the aircraft affect flight performance, but also how it affected RAC and Score as an overall result. Since Score is a complex function of every aspect of the aircraft, the importance of any one factor could not be determined on inspection alone. As such, the flight analysis program was utilized find the optimum value of all factors. By varying each parameter of the aircraft, the program allowed the team to begin to develop the type of aircraft that would compete with the greatest potential for success.

3.1.2 Important Design Parameters Investigated

Parameters important to flight performance were chosen after analyzing the mission profile. These flight parameters were investigated to maximize performance. Cost parameters were chosen after analyzing the Rated Aircraft Cost equation. .

<u>Cruise Velocity:</u> Since lap time is one of the major contributors to Score, the cruise velocity was extremely important. Furthermore, since the location of this year's competition is in Wichita, Kansas the consideration of wind was also important. With a higher wind speed, a fast plane is more likely to succeed than a slow one.

<u>Payload Capacity</u>: Flight score is directly related to the amount of payload carried. It is critical that the design configuration be optimized for payload weight and capacity.

<u>Lift:</u> Lift is obviously important on takeoff, but aircraft geometry stems from the necessity of having to lift the aircraft and its payload in the restricted distance. Therefore, the lift of the aircraft affects the cost due to the weight and geometry needed to takeoff.

<u>Drag</u>: Considering that flight score is a direct result of flight time and high velocity preferred. Since drag directly hinders velocity, minimizing drag will provide better scoring potential. The reduction of drag will result from a sleek design.

<u>Payload Configuration/Fuselage Length:</u> Since the length of the payload governs the length of the fuselage, these two characteristics are closely related. The more balls that are put abreast, the shorter the fuselage needs to be. The fuselage length can also be adjusted by changing how the forward and aft ends are tapered. If vertical tapering is performed, the fuselage can be shortened considerably when compared to lateral tapering. However, some configurations require lateral tapering in order to be more streamlined.

Battery Weight: Varying battery weight has numerous consequences. Increased battery weight provides a faster plane with less wing area and span due to increased power. However the increase in battery weight of the battery adds to the overall lift necessary for the plane to takeoff.

Wing: All aircraft geometry affected the RAC, but the wing size has a direct correlation with aircraft lift. The characteristics of the wing can be broken into the following three categories aspect ratio, area, and span. Stall is directly related to Aspect Ratio (AR). A low AR wing will stall at a higher angle of attack than a high AR wing. However, there are obvious tradeoffs when it comes to the choice of the final wing AR. As AR increases, the coefficient of lift for a wing of set planform area also increases. AR also changes the lift coefficient relative to angle of attack. An increase in area allows a greater weight to be lifted but also adds weight and form drag to the plane. The greater the span, the higher the RAC, but this will also increase the lift of the wing.

Each of the parameters stated above played an important roll in the mission performance and RAC. However, to design the optimal configuration it was necessary to know which parameters will have the greatest effect on scoring potential.

3.1.3 Sensitivity Analysis

The sensitivity of each parameter to score was analyzed using the simulation code. The code simulated every phase of the flight mission. Then the design parameters were varied and their effect on score was documented. From the results of the variation it was clear the degree to which changing the parameter affected the scoring potential of that configuration. From this information, emphasis was placed on the most important parameters to ensure the achievement of the most consistently high scoring configuration.

3.1.3.1 Assumptions Made and Justification for Their Validity

During the course of conceptual evaluation, several factors were assumed in order to accelerate the selection of a configuration.

The initial use of the code utilized historical data such as propulsive efficiency, battery characteristics, lift, and drag. Propulsive efficiency, lift, and drag were taken from previous year's competitions, and the battery type used in the code was researched from the top finishers of last year's competition. These estimations provided a good foundation for analysis.

When configurations were compared in the flight analysis program, it was assumed that the same weight model, wing characteristics, motor configuration, and lift coefficient applied to each. This allowed the comparison to focus on the benefits of the configurations.

3.1.3.2 Results

The sensitivity analysis generated useful data that directed the design choices. In the included figures, score was multiplied by 10 for a more simple representation.

Payload Capacity: It can be seen from Figure 3-2 that as the payload increased the score increased. It was this trend that illustrated the significance of having a design configuration that is capable of carrying 24 softballs.

Lift: It can be shown that increasing the lift allows for a higher score (Figure 3-2). This is most likely cause by the reduction in wing area and battery power required for takeoff. However, multiple consequences were weighed when judging coefficient of lift. High lift allowed for shorter takeoff distances and larger weight limits, but at the plane's low Reynolds number an increase in lift caused the plane to experience an increase in induced drag at cruise.

Drag: Decreasing the drag of the plane had a significant effect on the overall score. Less power was required for each flight, thus reducing the required battery weight significantly. Since the plane was also able to achieve a higher cruise velocity the plane completed laps in less time. This trend lends itself towards a more streamlined shape than traditionally seen. If the drag can be reduced by 50%, the Score could be increased by nearly 20 percent (Figure 3-2).

Payload Configuration/Fuselage Length: The contest rules state that the softballs must be in rows of at least two abreast and the fuselage width constraint of 18 inches limited the payload configurations considered to rows of 2, 3 or 4 abreast. The 2 abreast configuration created a long slender fuselage. This configuration had the lowest drag coefficient, however, the long body required to carry all of the softballs created a large RAC penalty. The 3 abreast configuration had a slightly higher drag coefficient than the 2 abreast configuration but a shorter fuselage length. The optimization code showed that the RAC savings of the shorter fuselage overpowered the increased drag effects for the 3 abreast configuration, allowing this configuration to have a greater scoring potential than the 2 abreast configuration. Due to taper ratio limitations, the fuselage length of the 4 abreast configuration was nearly identical as that of the 3 abreast configuration. The added width also caused an increase in drag for the 4 abreast configuration. The optimization code again showed the 3 abreast configuration had a greater scoring potential.

Battery Weight: From the analysis the battery weight was determined to have been the most sensitive to scoring potential. This was due to the battery's effect on flight performance and RAC. A large battery pack allowed for large amounts of power to be devoted to takeoff and cruise, allowing for quick takeoff and high cruise velocities, however battery weight is heavily penalized in the RAC calculation. Likewise, lower battery weight gave a smaller RAC but resulted in a slower plane takeoff that needed a larger area and wingspan for equal lift. The analysis predicted that for any given configuration there was an optimal number of battery cells. This was evident in the peak of Figure 3-2.

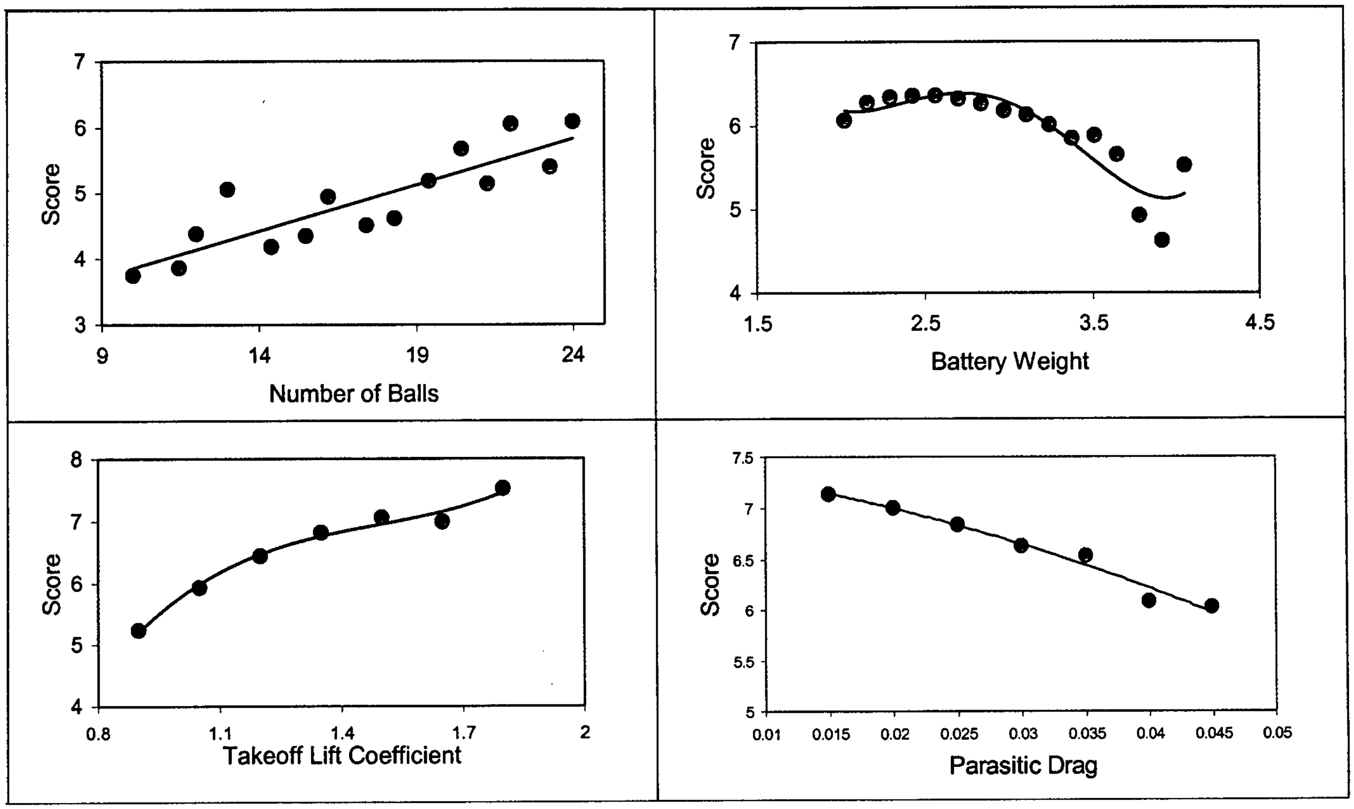


Figure 3-2: Sensitivity Analysis

Wing: The wing size was a direct result of the restriction on takeoff distance. A heavy aircraft or an aircraft with a small power source required a wing with a large lifting area in order to meet the distance restriction on takeoff. As stated earlier a large lifting force causes a large induced drag force. The slow cruising velocity resultant from high drag is not desirable and thus a very high lift wing is not optimal. The fastest flight times will likely produce the best flight score. However, to attain high cruise velocities low drag and high engine power are needed. The lowest drag airfoils generally produce the least lift and high engine power result from large amounts of batteries. From the trade study a moderate lift wing with low drag is warranted.

RAC: It seemed intuitive that the lowest RAC would produce the highest score. However, the results of the sensitivity analysis show a peak or optimum RAC for a given design configuration (Figure 3-3). The trend illustrated that as the RAC decreased the scoring potential for that configuration increased up to a point, eventually the configuration's flight performance was sacrificed in order to reach the lowest RAC. . Many of the cost criteria for RAC affect flight performance. Therefore, RAC minimization alone may not improve scoring potential.

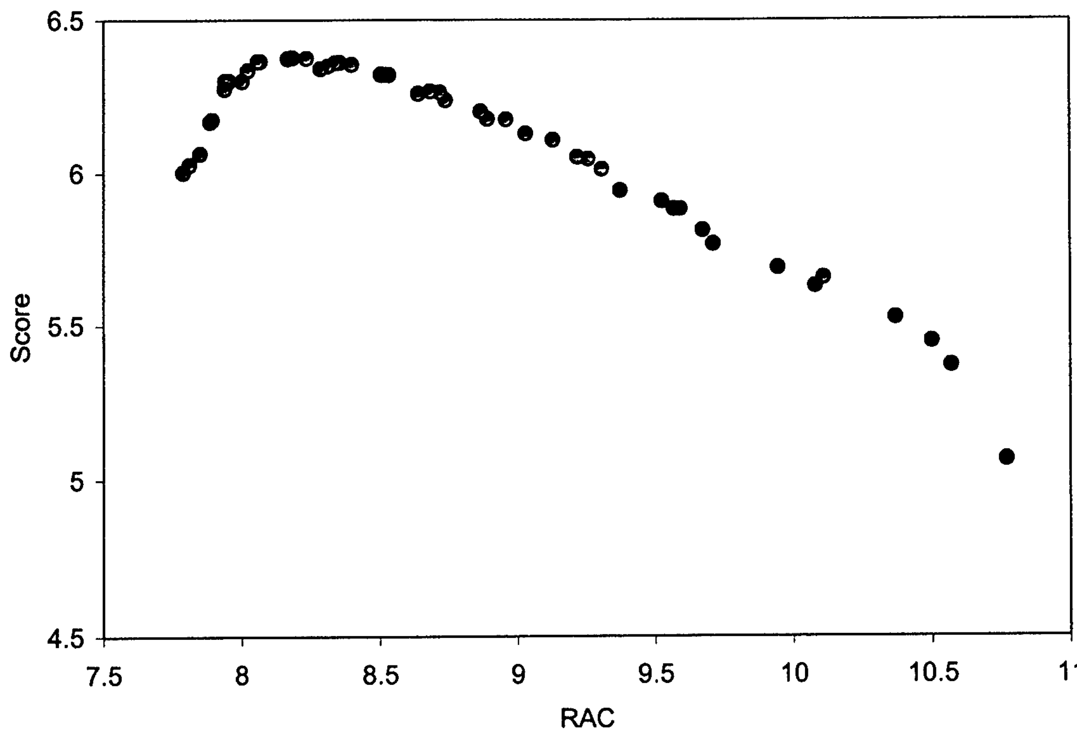


Figure 3-3: RAC vs. Score

3.2 Alternative Configurations Investigated

From the results of the sensitivity analysis key parameters were used to evaluate various design configurations. The decision matrices are used to evaluate the designs and make choices. The weighting parameters are based upon the importance of each factor and their relation to the overall configuration of the aircraft.

3.2.1 Figures of Merit

Modularity: Modularity allows for error. In the event of an accident the main portions of the aircraft, wing, fuselage, and landing gear, will break at a weak link. This link would be replaceable and inexpensive, a nylon bolt for instance.

Ease of Construction: The aircraft-testing phase will be thorough and it is likely that parts will need to be replaced. If each piece of the plane is easily constructed then there will be more time to focus on improving the aircraft.

Control: To be competitive in this competition three of five runs need to be completed. This will not be done well or at all with an uncontrollable aircraft.

RAC on Non-performance Issues: It is not possible to rule on all configurations based solely on RAC because many of the major factors in RAC weigh heavily in aircraft performance such as battery count and wingspan. However, there are a few items factored into RAC that do not affect performance such as

body length, empennage configuration as it is related to control surfaces and servos, and wing control surfaces as they relate to servos.

Score/RAC: The theoretical ability of a configuration to score well is very important. The change in flight characteristics of a configuration may have a positive contribution to aircraft performance but a negative contribution to RAC. As a result, a simulation of the configurations performance and the configurations RAC must be done together to justifiably rule on some configurations.

Payload Accessibility: A long ground time can hinder an otherwise superbly designed aircraft. The ability to load and unload the payload quickly and easily is essential.

Efficiency: Efficiency directs the overall design. The success of the aircraft improves as more useful energy is harnessed from the available energy.

3.2.2 Planform Configuration

After generating the trade studies and sensitivity trends using the flight analysis program, the design team set out to establish the optimum configuration to meet the challenges of the mission profile. In order to achieve the maximum score, our team investigated many possible configurations. It should also be noted that many designs were easily eliminated due to the restrictions placed on the team by the regulations of the contest, our manufacturing capability, and budget. With this understanding, the overall designs were trimmed to five basic designs. The Bi-Wing, Dual Fuselage, and Flying Wing were selected for their exploitation of trends observed in the flight analysis program. The Bi-Wing design exhibits increased lift, the Dual Fuselage has a shorter fuselage, and the Flying Wing has a very short fuselage and a shorter span. In addition to these, the Canard was added for its improved stall characteristics. The Conventional was used as a baseline for comparison.

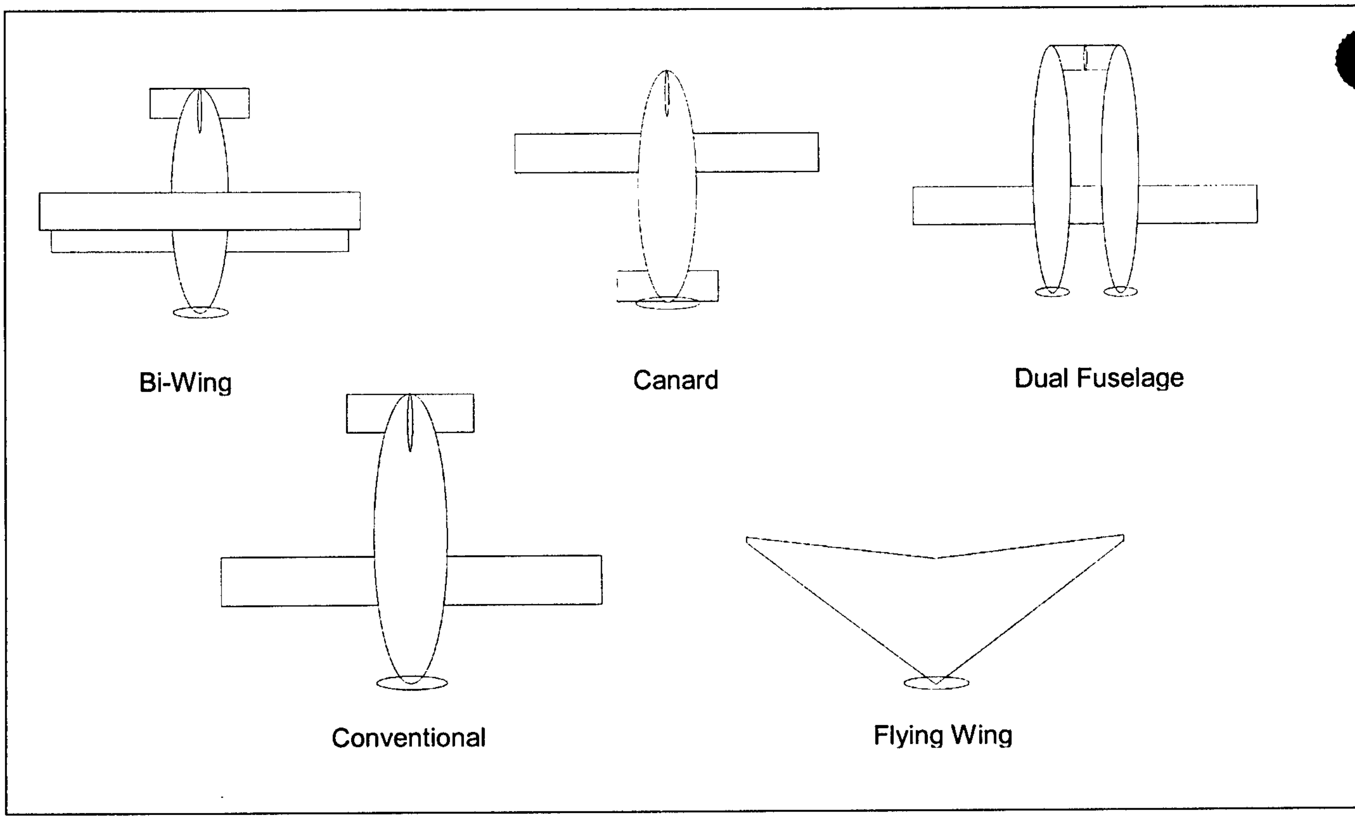


Figure 3-4: Alternative Configurations

From the results of the sensitivity analysis, key parameters were used to evaluate various design configurations.

3.2.3 Wing Location

Three wing locations, high, mid and low, were considered for the contest. A high wing location creates a natural dihedral effect on the plane and good wing tip clearance. However, a high wing creates a problem with loading and unloading the softballs. A mid wing location would allow for the top of the plane to be used as a payload hatch. Attachment of a mid wing would be more difficult than a high or low wing. A low wing location allows for a single piece wing to be built that could be bolted to the bottom of the fuselage A low wing allows the top of the fuselage to be used as payload hatch for easy loading and unloading of the softballs.

Figures of Merit	Weight	High Wing	Mid Wing	Low Wing
Stability	0.1	1	0	-1
Payload Accessibility	0.5	-1	0	1
Ease of Construction	0.2	0	-1	0
Modularity	0.2	1	-1	1
Total	1.0	-0.2	-0.4	0.6

Table 3-1: Wing Location Decision Matrix

3.2.4 Empennage Layout

Due to the RAC penalty for vertical and horizontal surfaces, a tailless aircraft appeared to have the highest scoring potential at first glance. However, the instability in yaw associated with a tailless aircraft was not acceptable with the expected wind conditions in Wichita. Several tail configurations were then considered for the contest. A V-tail was initially considered because it had a 5-point reduction in RAC penalty when compared to a conventional tail. A V-tail would be more difficult to construct than a conventional tail because of having to mount the two surfaces at the correct angle. When the conventional tail was further investigated, it was discovered that running the rudder and nose gear with the same servo would negate the V-tail's RAC advantage. Cruciform and T-tail configurations were also considered. Both configurations required increased strength in the vertical surface to support the horizontal surface. This increased strength would cause both configurations to be heavier and more difficult to construct than a conventional tail.

Figures of Merit	Weight	Conventional	Cruciform	T-tail	V-tail	No Tail
RAC	0.5	0	0	0	0	1
Ease of Construction	0.1	1	-1	-1	0	1
Servo Location/Accessibility	0.2	1	0	-1	1	1
Control	0.2	1	1	1	1	-1
Total	1.0	0.7	0.0	-0.3	0.5	0.6

Table 3-2: Empennage Decision Matrix

3.2.5 Wing Characteristics

<u>Sweep Angle</u>: Sweeping the wings aft increases tip loads as well as increases the boundary layer at the tips. This produces wing tip stall patterns, which can lead to reduced aileron effectiveness, loss of control, and shifting in the wing's aerodynamic center. In regards to the contest, by sweeping the wing, the span used in determination of the RAC is reduced. However, this slight reduction in RAC is not worth the drawbacks to sweep in any direction.

Taper Ratio: Assuming that wing area and span are constant, the taper ratio's main effect on score is to increase the maximum chord and the RAC. Overall, the effect of an increase in chord is almost negligible when calculating RAC. For maximum lift, the lift distribution of a wing should be elliptical (this can be done with an elliptical planform), but achieving this distribution is often difficult. This distribution of force reduces the induced drag caused by lift. If the taper ratio is reduced to less than .45, the stall pattern of the wing shifts towards the wing tips. As a result, when the plane begins to stall the ailerons will lose effectiveness and the pilot will lose control of the aircraft. The negative stall patterns of tapering and the fact that increase in chord is a penalty in this year's competition, a rectangular wing proves most attractive.

3.2.5.1 Motor/Propeller Configurations

The propulsion team began the conceptual design process by sketching the various motor/prop configurations that could be possible. The selection process started with a great number of possible combinations for the propulsion system and then began narrowing these down. The three final configurations that were investigated are shown in Table 3-3.

1 engine / 1 prop: For this configuration, three main advantages were discovered. It was definitely lighter in weight, lower in RAC, and is easy to incorporate into any design. However, the disadvantages are possibly less power than a two-engine configuration. Also, motor failure would result in an immediate termination of the flight.

1 engine / 2 props: As an alternative to one engine and one propeller, a configuration of one engine and two propellers was also investigated. One of the advantages that this configuration has is that the torque from a one engine and one propeller setup induces no roll if the propellers are set up to spin in opposing directions. Secondly, the props could be mounted in an area where structural blockage behind the propeller is minimal. One disadvantage of this configuration involves construction difficulties due to gearing the two propellers to the motor. This configuration also has a lower efficiency, increased RAC, and is higher in weight. If one propeller were to fail in this configuration, a large yaw moment would be produced, but flight would still be possible. Failure of the motor would result in immediate termination of the flight.

2 engines / 2 props: The final configuration that was inspected was that of two engines and two propellers. One advantage to using this configuration would possibly be more power available if it is found that there are less electrical losses due to lower current being sent to each motor. This design would also allow the propellers to be mounted in areas could be minimized. Some of the disadvantages are that this configuration would be more costly due to buying two motors and that it would result in a much higher RAC value due to increased aircraft weight and the fact that more propellers and motors are present.

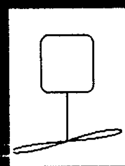
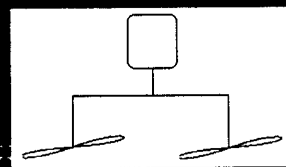
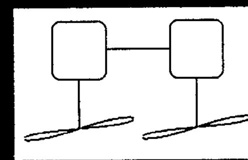
Figures of Merit	Weighting Factor	1 Prop/1 Motor 	2 Props/1 Motor 	2 Props/ 2 Motors 
Construction	0.07	4	1	3
Cost	0.1	5	3	1
Durability	0.1	3.5	2	1.5
Ease of Modification	0.03	3	1	3
Efficiency	0.2	2	3.5	3
RAC	0.35	5	4	1
Weight	0.15	5	3.5	1.5
Total	1	**4.12**	**3.225**	**1.725**

Table 3-3: Propulsion Decision Matrix Geared Drive vs. Direct Drive

Two other options that were investigated for any propulsion configurations were geared motors versus direct drive motors. Gearing a motor would allow the power to be adjusted while using the same motor, but it would also be heavier in weight and a little more costly. The direct drive motor would be a lighter in weight, less expensive and easier to incorporate for analysis and installation, but at the same time it would not allow the motor power to be adjustable.

3.2.6 Material Selection

Table 3-4 is the decision matrix for determining the material used for the aircraft. The highest-ranking method is carbon fiber and foam. The durability of the carbon and foam wing is far greater than other methods.

FOM	Weighting Factor	Carbon Fiber/Foam	Carbon Fiber Ribs/MonoKote	Balsa Ribs/MonoKote
Weight	0.30	2	4	4
Durability	0.35	5	3	2
Cost	0.10	4	3	2
Construction	0.25	3	2	2
Total	1	**3.5**	**3.05**	**2.6**

Table 3-4: Construction Method Decision Matrix

3.2.7 Landing Gear

Five categories are deemed crucial in the pre-design phase of the landing gear configuration. The figures of merit in ranking order are as follows: Interchangeability, Stability, Strength/Rigidity, Weight, and Ease of Construction. The weight of importance assigned to each of these categories is shown in the decision matrix above. Due to unpredictable wind conditions, damage to the aircraft is probable. Interchangeability allows for easy replacement of damaged parts and was considered for a suitable landing gear arrangement. Additionally, in order to avoid damage landing gear must be stable and strong. The landing gear configurations were estimated to have very similar weight. Weight is typically a chief consideration to overall score. However, in this case it would not have a significant influence. Ease of manufacturability would allow for more efficient construction and rigidity would provide a smoother landing.

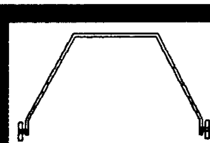
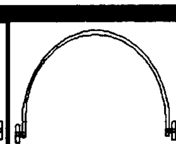
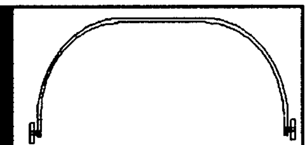
					
Figures of Merit	Weight	One	Two	Three	Four
Interchangeability	0.3	1	0	-1	1
Stability	0.3	1	1	1	1
Strength/Rigidity	0.2	1	1	-1	1
Weight	0.1	0	0	1	0
Buildability	0.1	1	-1	1	-1
Total	1.0	0.9	0.4	0	0.7

Table 3-5: Landing Gear Decision Matrix

As shown in the sketches above: figure one, three and four mount flush with a flat surface. Rounded configurations, like figures two and four, are more difficult to build than designs consisting of flat sides and edges. Figures one, two, and four are fastened to the wing and fuselage through several, breakaway, nylon bolts. This design promotes uncomplicated interchangeability. Alternatively, sketch three would be mounted at 1/2 and 2/3 of the wingspan. This does not allow simple interchangeability and also causes strength issues considering the mount to the wing alone. Therefore sketch one, the bow configuration, was chosen for it's most complete fulfillment of each category.

3.3 Preliminary Ranking

With the initial configurations set, the Score Potential for each was determined using the flight analysis program. Design parameters were assumed and the program flew each configuration through the mission profile, generating the optimum values for each aircraft in terms of RAC and potential score. These values were used to determine the design that would be optimum for the contest. In addition,

other figures of merit were taken into consideration when evaluating the designs. As Table 3-6 shows, no single configuration had a considerable advantage over any other with regards to Score.

Figures of Merit	Weight	Conventional	Canard	Dual-Fuselage	Bi-Wing	Flying Wing
Score/RAC	0.4	0	0	0	0	0
Ease of Construction	0.2	0	0	-1	-1	1
Control	0.2	1	1	1	1	-1
Payload Accessibility	0.2	1	1	0	-1	1
Total	1.0	0.4	0.4	0	-0.2	0.2

Table 3-6: Configuration Decision Matrix

- The bi-wing has a high lift capability while minimizing span, however as span is defined as the sum of all lifting surfaces in the definition of RAC in this year's contest, the bi-wing's advantage is eliminated.
- The dual-fuselage has a means to support a longer wing than a single fuselage. The extra support structure would allow for a slightly higher aspect ratio wing, which would fly the plane at a high efficiency reducing induced drag, permitting flight with less battery, and thus reducing RAC. Conversely, our strategy was to produce an aircraft that was modular; this would not rely on strength of material for durability rather the plane would break into easy to assemble pieces. With a dual-fuselage design modularity would be nearly impossible.
- The canard offers more safety than a conventional plane in that it can be designed so that the canard stalls before the main lifting wing, this would prevent devastating wing stall effects. As a result, the main lifting wing cannot close in on its max C_L. Thus, the canard configuration would sacrifice lift for security. This particular contest does not reward safety, yet increased lift improves the potential score of the aircraft. The canard also does not exploit any RAC benefits.
- The flying wing's main characteristic is that it has no empennage. The combination of a lack of an empennage, a decreased fuselage length, and a shorter wingspan reduces RAC, thus increasing score. The draw back to having no tail is seen in the stability and control of the aircraft. The stability and control analysis would be difficult and the configuration would require near perfect manufacturing and production as well as a high degree of pilot skill. Also, the low lift coefficient at takeoff limits the maximum possible score.
- The conventional configuration is a good all-around design. However, to choose this configuration is to ignore the benefits of the others. It is this reason that a blended configuration evolved.

Since the initial ranking did not yield a design satisfactory to the team, it was decided to take further steps to ensure the optimal configuration.

3.4 Evolution to a Final Design

The process of defining the final configuration began with the goal of utilizing the advantages and removing the disadvantages of each configuration considered. The design evolved from these parameters into the configuration that best exploits the contest rules.

3.4.1 The Final Design

The final design concept is a discretely blended flying wing (Figure 3-5). This effectively modified the flying wing to act as two wings, the main wing and the body wing. By blending the wing discretely construction was made easier. Also, the chord of the body wing is not penalized as it is within the 18-inch payload carriage limit.

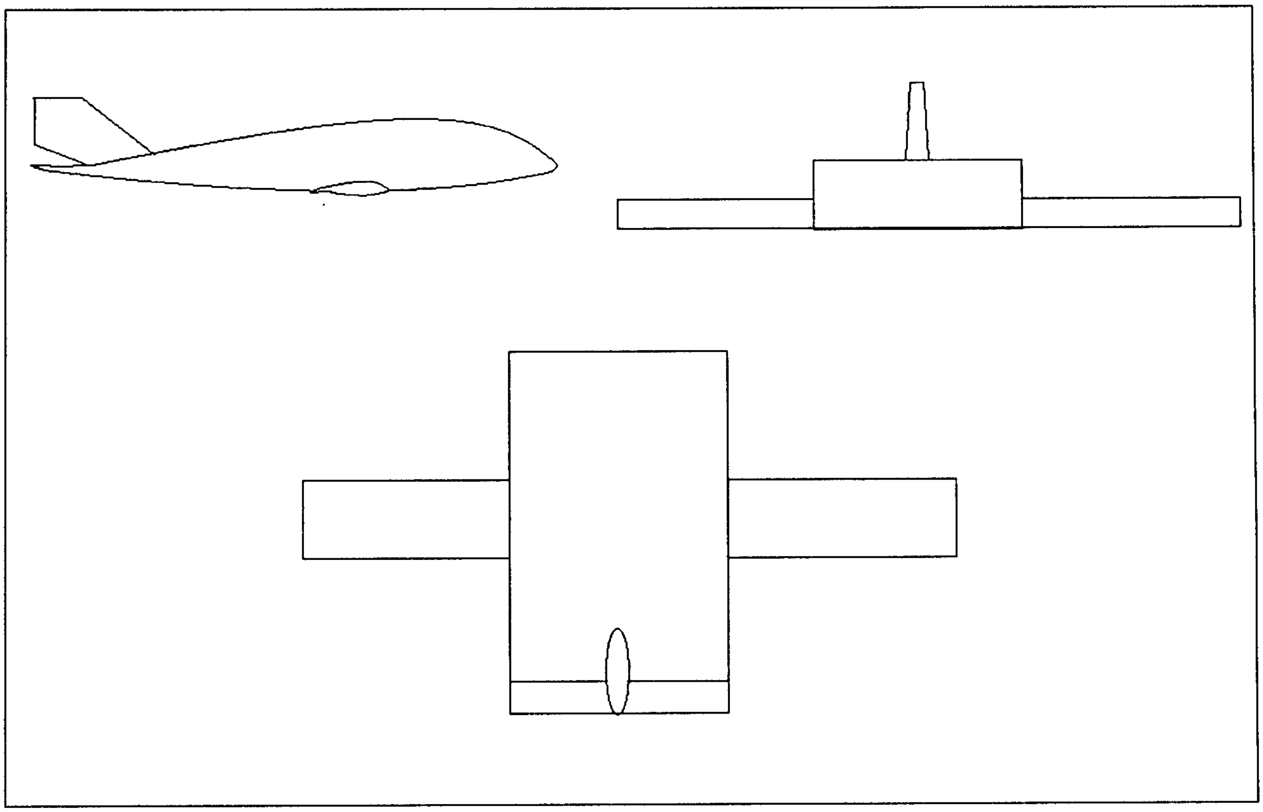


Figure 3-5: Final Configuration Sketch

The body wing is a low aspect ratio wing with a control flap on the trailing edge. This wing is 17 inches in span and is responsible for carrying the payload. The other is a high aspect ratio wing that accounts for the main lifting portion of the aircraft. The lifting wing is aft of the aircraft center of gravity (the center of gravity of the aircraft is at approximately 38% of the body wing chord). With the main wing aft of the

center of gravity and the inherent nature of a cambered lifting airfoil, a nose down pitching moment is produced. Due to the negative lift and drag side effects of tapering and sweep at our Reynolds number, a rectangular wing planform was selected. The body wing's aerodynamic center is ahead of the center of gravity; also the body wing's airfoil has a reflexed trailing edge. As a result, the lifting force on the body wing and the airfoil shape produce a nose up pitching moment. The use of the body wing's flap, or elevon, increases or decreases its nose up pitching moment. It is this mechanism that is used for pitch stability and control; this compensates for the flying wing's poor trim and control characteristics. In an effort to increase the yaw stability and control of a flying wing a vertical control surface is used.

3.4.2 Features That Produced the Final Configuration

- The modified flying or ModWing utilizes the advantage of the flying wing by using a low aspect ratio flying wing for the body. This highly reduces control surfaces and the servos needed to control them.
- The body wing, like any other wing, is streamlined toward the flow in two dimensions while the tips remain flat. This ensures ease of construction. The airfoil shaped body also allows for optimum placement of payload as well as greatly reduces the overall body length (for streamlining in three dimensions addition taper length is necessary).
- The main wing placement on the belly of the body wing allows for modularity in a 'break away' wing.
- The addition of a wing with a reflexed trailing edge onto the flying wing configuration allows for better stability and control without the necessity of a conventional empennage.
- Having a lifting body in place of a traditional fuselage decreases the lift loss of the wing at the peak of the lift distribution thereby shortening the span addition needed to make up the loss.
- After running calculations for the new configuration it is found that the potential score for the final design shows a 14% increase over any other design considered (Figure 3-6). The RAC saw a decrease of approximately 12% from the conventional layout.

3.5 Final Ranking Chart

The final ranking of the investigated configurations can be weighed with the figures of merit important to the competition. The scoring potential of a configuration is very important however it is not the only factor to consider. The design must be constructible within the team's abilities and budget, and it must be designed with the pilot's ability in mind. Score/RAC comparison and the final ranking chart are displayed below.

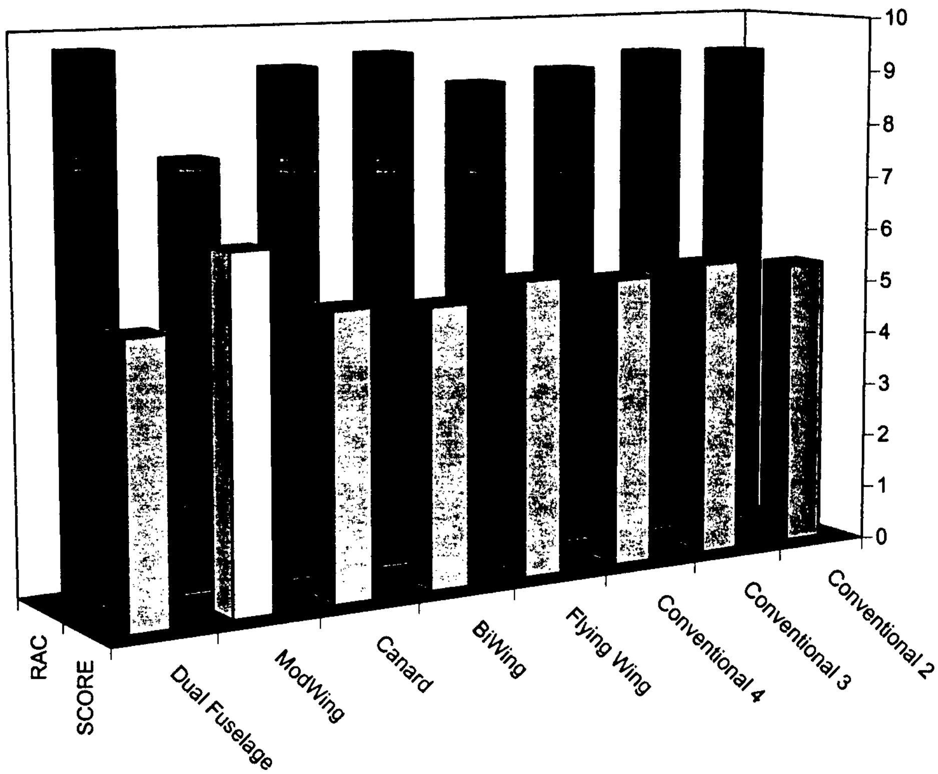


Figure 3-6: RAC and Corresponding Potential Score for Each Configuration

Figures of Merit	Weight	Conventional	Canard	Dual-Fuselage	Bi-Wing	Flying Wing	Modified Flying Wing
Score/RAC	0.4	0	0	0	0	0	1
Ease of Construction	0.2	0	0	-1	-1	1	1
Control	0.2	1	1	1	1	-1	0
Payload Accessibility	0.2	1	1	0	-1	1	1
Total	1.0	0.4	0.4	0	-0.2	0.2	0.8

Table 3-7: Final Ranking Chart

In order to solidify the selection, the team created several hand launched gliders for test purposes. Most of the initial concerns were alleviated and the configuration was given a green light for the preliminary design phase.

4 PRELIMINARY DESIGN

During the Preliminary Design, the simulation code was modified to represent the configuration chosen. The code was enhanced by using improved aircraft performance, weight, and propulsive models. After several iterations of the improved code the design parameters were optimized for this year's competition.

4.1 Design Parameters Investigated

Using the updated simulation code and the trends from the sensitivity analysis, essential characteristics of the aircraft were refined.

Lift: Several trade-offs were considered when judging coefficient of lift. Increased lift allows for the transport of a heavy aircraft or provides short takeoff distances for light aircraft. However, larger control surfaces are required to counteract the large moment about the aerodynamic center characteristic of high lift airfoils. Also, the increase in lift produces more induced drag.

Drag: Parasitic drag was reduced with the selection of a sleek design. However, induced drag relates directly to lift; as lift increases induced drag also increases. As seen in the sensitivity analysis of Figure 3-2, the reduction of drag is critical in maximizing flight score.

Battery Weight: Varying battery weight has numerous consequences. Increased battery weight allows for more takeoff power, resulting in a faster plane with less wing area and span. However the increase in battery weight adds to the overall lift necessary for the plane to takeoff. Again from Figure 3-2, increasing the number of batteries too much will result in a lower flight score due to the high RAC penalty associated with battery weight.

Wing: The important attributes of the wing can be broken into two categories: aspect ratio (AR) and area. As AR increases the efficiency of the wing increases, thus a high AR wing has a higher lift coefficient for a given angle of attack than a low AR wing. An increase in area allows a greater weight to be lifted but also adds weight and form drag to the plane. While a high efficiency, large area wing provides more lift, however the RAC penalties associated with weight, chord and span must also be considered.

All positive flight performance characteristics, associated with a parameter, have negative RAC effects. Therefore, designing to increase either flight performance or RAC independently produces a poor design. Due to the fact that the extremes of design provide the worst scores a peak must exist. The simulation code was critical in locating that peak.

4.2 Figures of Merit

Weight: Weight is an important factor in the design of any aircraft because of issues such as increase in lift requirements, decreases wing and power loading, and also decreases thrust.

Efficiency: Efficiency directs the overall design. The success of the aircraft improves as more useful energy is harnessed from the available energy.

Power Density: The higher the power density of a battery the more power that is receive per unit weight. REP is based off of battery weight this year as opposed to battery power as in previous years.

Cost: Financial cost is a key concern when working with a fixed budget.

Power Output: The power output is an important consideration motor selection. This factor will directly affect take off distance and cruise velocity. These, again, have a direct effect on RAC/Score.

Max Current Load: It is crucial to select a motor that can fulfill the current load requirements of the mission performance specifications.

Availability: Sufficiently testing the intended motor is a wise investment of time. It is important to know the availability and purchasing lead-time of the motor in order to budget enough time for testing purposes.

4.3 Simulation Code Improvement

After the final configuration was decided, the flight analysis program was altered to more accurately reflect the chosen design.

4.3.1 Aerodynamics

As shown in the trade studies, achievement of the correct aerodynamic characteristics is needed to maximize scoring potential. The aerodynamic characteristics investigated were lift and drag of the wing.

4.3.1.1 Airfoil Selection

From the sizing trades completed in the conceptual design, it was known that a high lifting wing with low drag at cruise conditions was desirable.

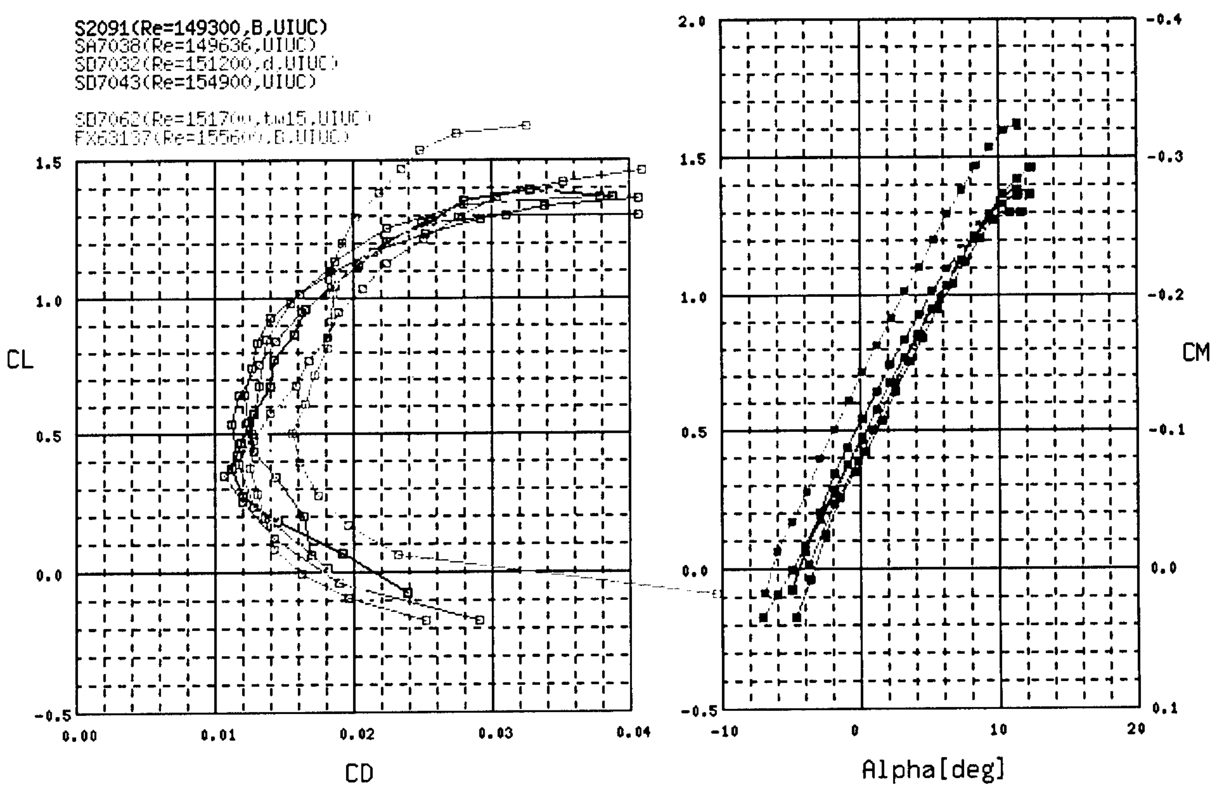


Figure 4-1: Polar Curve Comparison

Main Wing: From the simulation code it was decided that the airfoil of the main wing also needed to possess a high lift coefficient at takeoff conditions. At takeoff, the minimum Reynolds number was defined to be 150,000 in the flight analysis program, due to the fact that undesirable viscous affects occur at lower

Reynolds numbers. Lift and drag information on a number of airfoils was attained using the Nihon University Aerospace Student Group's (NASG) Airfoil Database. The search began with families of airfoils that have the aerodynamic characteristics desired at low Reynolds numbers. These were Eppler, Selig, Selig Donavan, and Wortman's. From these families, data from hundreds of airfoils were available. In an effort to further narrow the field a requirement for coefficient of drag was applied. The drag is an important factor that is most critical in cruise. The simulation code suggested a high lift coefficient. The cruise C_L was 0.3 – 0.7, depending on the payload condition of the aircraft. At cruise, the operating Reynolds number of the wing was near 300,000. Figure 4-1 shows that of the final airfoils investigated.

The lifting capability of each of these airfoils is similar, with the FX63137 capable of the greatest lift. However, going further with the investigation showed that the drag penalty for the FX63137 was much greater than for the other airfoils at cruise conditions. A similar drag polar for cruise conditions was then used to make a final main wing airfoil selection. Using these criteria, the Selig Donavan 7062 was decided upon for the main wing.

Body Wing: The main consideration for the body wing was to balance the moment about the aerodynamic center of the main wing. The main wing inherently produces a nose-down moment on the aircraft due to the camber needed to create lift. To combat this, an airfoil with a reflexed trailing edge was chosen because the reflexed airfoil naturally produces a balancing nose-up pitching moment. The other major concern was fitting the payload inside of the wing. Several airfoils would satisfy the requirement of a reflexed trailing edge, but only a few appeared to be capable of satisfying the payload requirement (namely the Eppler 329 and Eppler 326). The E329 has the ability to store the most payload, but it was decided that the payload could be carried more efficiently in the E326, as it reduced unused payload space.

Vertical Tail: The selection of the airfoil for the vertical tail was not critical. The restrictions placed on the airfoil were that it be symmetric and low drag. From the information on the NASG Airfoil Database the NACA 0009 was chosen.

4.3.1.2 Drag Coefficient

Xfoil is a computer program that has the ability to import and give flight characteristics of an airfoil. It was utilized to give parasitic drag of the main wing, body wing, and tail at a given Reynolds number. The total drag for the aircraft was completely calculated using Xfoil since the aircraft is made up entirely of wings. A buffer of 20% was added to the total drag to account for the landing gear and interference resulting in an overall drag coefficient of 0.024.

4.3.2 Propulsion

4.3.2.1 Battery Weight Capacity

From data acquired in past years by OSU Aerospace Capstone students, it was found that quick charge batteries would be needed to effectively compete in this contest. The type of application that the batteries were designed for was also investigated. The high performance, "racing" batteries were all designed for applications that drained the voltage at a high current, as desired. This choice left eleven high charge/discharge, "racing" batteries to compare (Table 4-1). The top three were chosen on the basis of the energy density of each battery and the overall battery efficiency.

Name	Capacity (mAh)	Mass(g)	Capacity per mass (mAh/g)
RC-1700	1650	57	28.94
N-1300SCR	1500	52	28.84
N-4000DRL	4000	160	25.00
N-1250SCRL	1250	43	29.06
N-3000CR	3000	84	35.71
N-1900SCR	1900	54	35.18
RC-2400	**2400**	**61**	**39.34**
SR-2400	**2400**	**53.86**	**44.55**
P-500DR	5000	145	34.48
P-180SCR	1800	49	36.73
P-280CR	2800	79	35.44
KR-1100	**1100**	**24**	**45.83**

Table 4-1: Batteries Investigated

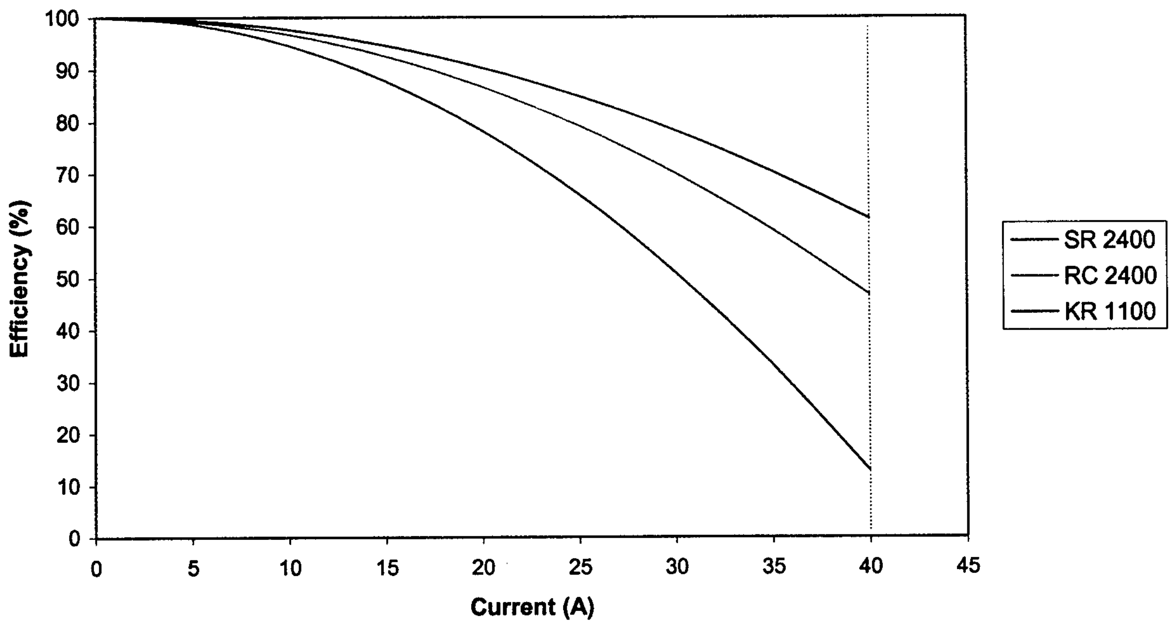


Figure 4-2: Battery Efficiency vs. Current

The high energy density of the SR-2400 as compared to the other types of cells again made it the top choice. The weighted value for each battery is listed next to its FOM's.

Figures of Merit	Weight	RC-2400	SR-2400	KR-1100
Efficiency	.40	1	1	0
Power Density	.45	0	1	1
Cost	.15	0	0	-1
Total	1.00	.40	.85	.30

Table 4-2: Battery Decision Matrix

4.3.2.2 Overall Propulsive Efficiency

Up to this point in the simulation code, the efficiency of the propulsive system was based on historical data. To optimize performance, researching numerous batteries, propellers, gearboxes, and motors maximized the theoretical efficiency of our propulsive system. This information, combined with the results of the simulation, helped to produce the combination of the four components that would generate the best possible efficiency.

The best combination of the propulsion components was determined with an iteration process between the propulsion team and the aerodynamics team. The propulsion team was first given a set of aircraft

flight specifications (such as takeoff thrust, cruise power, flight times, and ideal number of batteries) that was determined by the aerodynamics team assuming a certain total propulsive efficiency. From this information, the propulsion team modeled the aircraft's propulsion system to determine if the needed flight specifications could be reached. The new best possible configuration and total propulsive efficiency were then given back to the aerodynamics team, who used it to determine new flight specifications. This process of iteration was repeated until both groups converged upon the optimal flight specifications and total propulsive efficiency.

4.3.3 Structures

The weight model in the simulation code during the conceptual phase was primitive and did not appropriately penalize poor design configurations. With definite aerodynamic, propulsive, and structural concept configurations decided a more accurate weight model was created.

4.3.3.1 Weight Model

Each onboard component available, relating to the concept configurations, was weighed and the components that were not available were given conservative weight estimates. Using Computer Aided Design (CAD), each component's complex volume and density was determined and its weight was calculated. CAD was also used to find a more accurate weight model for the wing. The cross-sectional area of the wing airfoil changes exponentially with increased chord length, which has a dramatic effect on weight cost for the airplane. Using CAD and the airfoil specified by the aerodynamics team, the wing was generated with several different chord lengths that were within the design range. The exact area of each airfoil was calculated through CAD and graphed versus chord length. A linear interpretation of airfoil perimeter was also calculated versus cord length. With these equations and the thickness and density of the materials to be used, a more proficient weight model was implemented into the scoring program. The improved weight model more accurately penalized the increase in wingspan and chord length. The result was a better estimation of wing sizing and scoring potential.

4.4 Aircraft Sizing

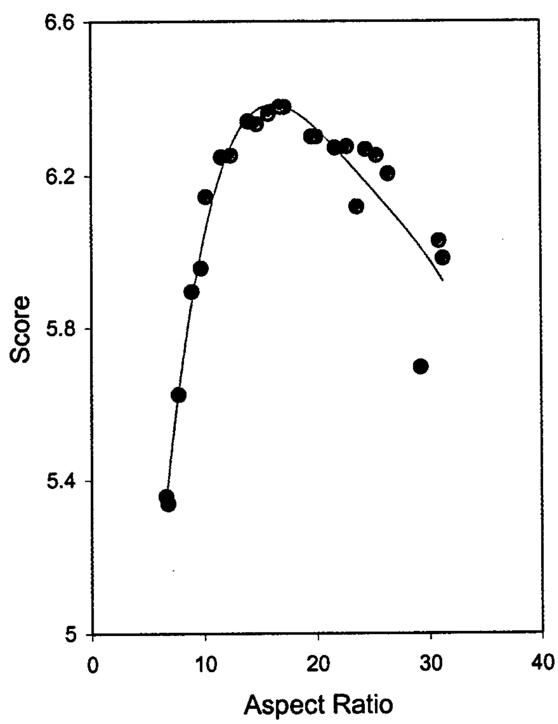
With the performance trades conducted and a more precise weight model, the focus was shifted to finalizing the dimensions of the aircraft. As the code became tailored to the configuration the simulation produced more precise results. From the simulation, flight performance was analyzed and aircraft sizing was completed.

4.4.1 Wing

The first value determined was the size of the body wing. Its size is governed by the payload restrictions given by the contest rules. The maximum width allowed by the contest rules is 18 inches. Therefore, four softballs was the maximum number that would fit abreast within that dimension, giving a width of 16 inches. Another inch was added to give room for fuselage walls and other components to bring the final width to 17 inches. The four abreast configuration meant that the payload volume would be six rows long.

In order to fit this volume of balls within the E326 airfoil, it was found that the body wing's chord must be 42 inches long. This also permits the weight of the softballs to be balanced around the center of gravity of the aircraft.

After inserting the calculated weight for the body wing, the main wing was sized using the flight analysis program. It was decided before the program was executed that a buffer area was not necessary in the selected design due to the lifting force provided by the body wing where the fuselage was normally found. Trade studies showed that an optimal area and aspect ratio existed for the competition (Figure 4-3).



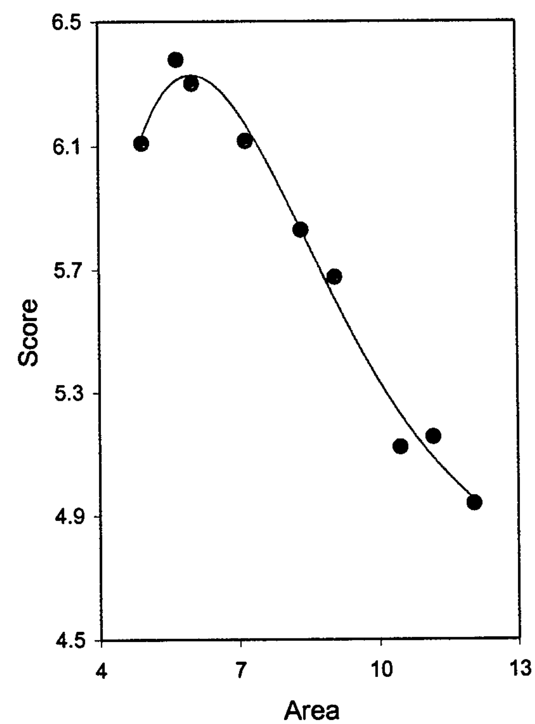


Figure 4-3: Wing Area and Aspect Ratio Trend

However, the trade studies were conducted simulating the conventional design. As such, the actual values due not apply to the final configuration. The final area of 866.8 in^2 and AR of 10.3 were converged upon using the simulation program. These values provide a span of 94.5 inches and a chord length of 9.15 inches.

4.4.2 Ailerons

The ailerons were the next issue to be addressed. There was uncertainty how much roll moment was necessary or required for RC aircraft. Therefore, roll moment was attained using historical data from previous Oklahoma State University DBF planes. Strip theory analysis was used to determine the aileron size and position that produced the desired roll moment. With this information, the decision was made for

the ailerons to be 20% of chord, two inches wide, located from 50% to 90% of each side of the wing. The resulting $Cl_{\delta a}$ was
0.321.

4.4.3 Rudder

The main consideration in sizing the rudder was the ability to conduct landings with the presence of a crosswind. For general aviation aircraft the range of conventional rudders was 25% to 50% of the total vertical tail (Raymer 1999). Initially the rudder was sized at 30%. This allowed the airplane to land parallel to the runway in a crosswind of 15 mph with 20° of rudder deflection. In addition, the stability program showed that a 1° rudder deflection would compensate for adverse yaw at full aileron deflection. The $C_{N\delta r}$ was calculated to be -.012.

4.4.4 Elevon

Pitch control is managed via the elevon located at the aft end of the body wing. When the elevon is deflected it, in effect, changes the camber of the body wing. In doing this, the lift of the body is altered as well as is the moment about the aerodynamic center. Due to the fact that the body is a low lift wing the change in the moment about the aerodynamic center is dominant. The elevon was sized based on its ability to balance pitch. Xfoil was again used to examine hinge position as it relates to moment control. Using slender wing theory to three-dimensionalize the airfoil, it was found that with the elevon at 90% of the body's chord the body had sufficient power to control pitch. This was later confirmed with computational fluid dynamics (CFD). The resulting $CM_{\delta f}$ was -0.304.

4.4.5 Stability

4.4.5.1 Lateral Stability

When considering the winds of Wichita, it was decided to focus greatly on the directional stability of the aircraft. The final plane was designed to be sufficiently stable, but not to a point where the winds would have a drastic effect on the direction of the plane's flight path. Due to the desire for low weathercock stability and the high Aspect Ratio of the main wing, Cn_β, sailplane theory (Nickel and Wohlfahrt 1994) was applied. A Cn_β of 0.04 was used for the tail sizing, which resulted in a tail surface area of 71.2 in^2.

4.4.5.2 Longitudinal Stability

The total pitch stability balance of the aircraft can be broken into three sections, body wing, main wing, and thrust as shown in Figure 4-4. Quantitative values for stability coefficients can be found in Table 4-3 and Table 4-4.

	α_0 (deg)	Cl_α (1/rad)	CM_0	CM_α	CM_{ac}
Wing	4.054	4.989	-0.171	-1.247	-0.0824
Fuselage	-0.004559	1.272	0.064	0.334	0.017

Table 4-3: Longitudinal Stability Coefficients

Component	CnB	ClB
Vertical Tail	0.166	.05
Fuselage-Wing	0	-.013
Total	0.166	-0.08

Table 4-4: Lateral Stability Coefficients

The body wing is a low aspect ratio wing with a positive C_{Mac}. The coefficient of lift versus angle of attack correction for a two dimensional airfoil was completed using slender wing theory. The aerodynamic center of the body was placed in front of the center of gravity of the aircraft; thus, as the lift of the body increases a positive pitching moment is produced. The effect of propeller wash was also considered. The flow of air through the propeller increases the dynamic pressure over the body, thereby increasing its efficiency. These effects were modeled using momentum disk theory (Raymer 1999). Thus, an increase in either angle of attack or thrust increases the nose-up pitching moment contribution from the body wing. The pitch control of the plane is manipulated by deflecting the elevon. As the elevon is deflected the moment about the aerodynamic center of the body wing is varied, which shifts the stability curve for the total aircraft up or down (Figure 4-4).

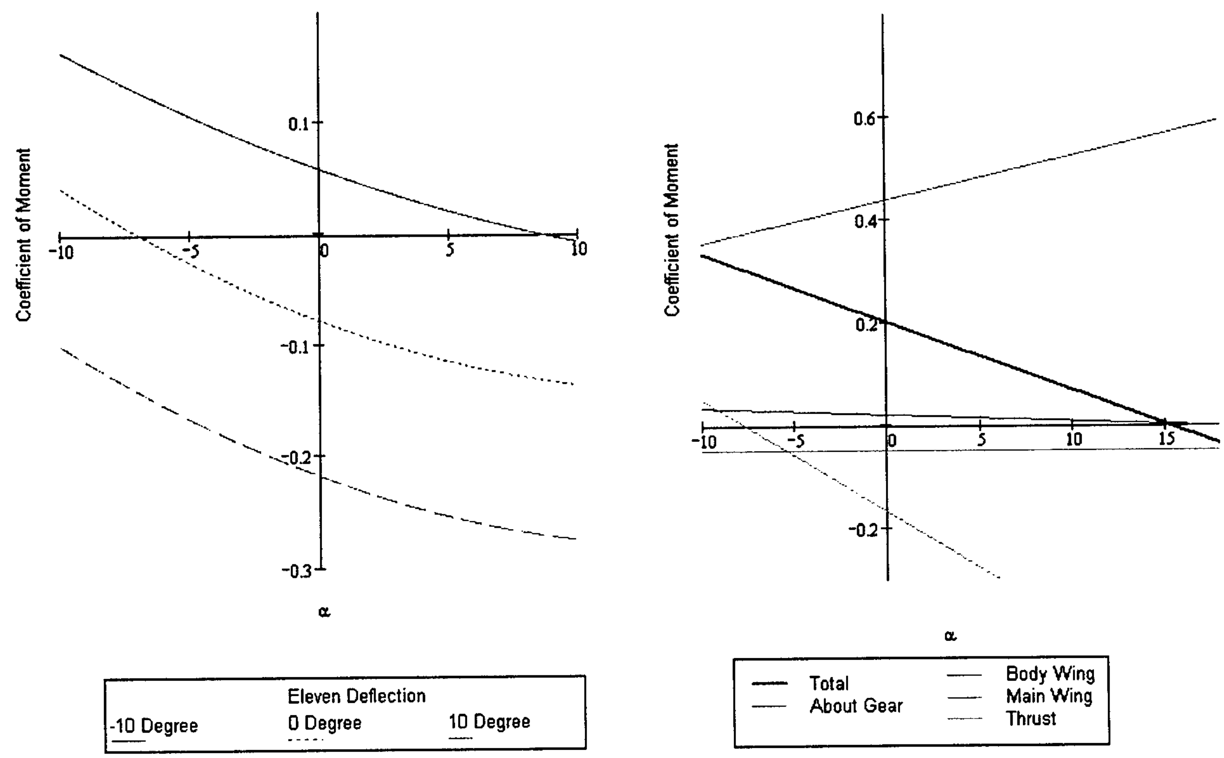


Figure 4-4: Pitch Stability Curves

The main wing is aft of the center of gravity. Therefore, as the lift increases on the wing the aircraft feels an increased negative moment. The main wing is a moderate lift airfoil that has a negative moment about its aerodynamic center. As angle of attack increases, the nose-down pitching moment contribution from the main wing increases.

The propeller thrust line is above the center of gravity of the plane. Angling the propeller downward by 2 degrees increased the distance between this line and the center of gravity. This was done to counteract the prop wash and attempt to restore the pitching moment curve to that of the glide line. The result is a nose-down pitching moment that directly correlates to thrust.

CFD analysis performed on the body of the aircraft was used to validate theoretical equations (Figure 4-5). The simulation was conducted about the longitudinal line of symmetry. From this data, stability and control was estimated and elevon effectiveness was properly attained. The results of the CFD analysis revealed that the aerodynamic center of the body was at 30% of the body's chord. It was also found that the three dimensional correction of airfoil was π multiplied by aspect ratio, double the value predicted by slender body theory.

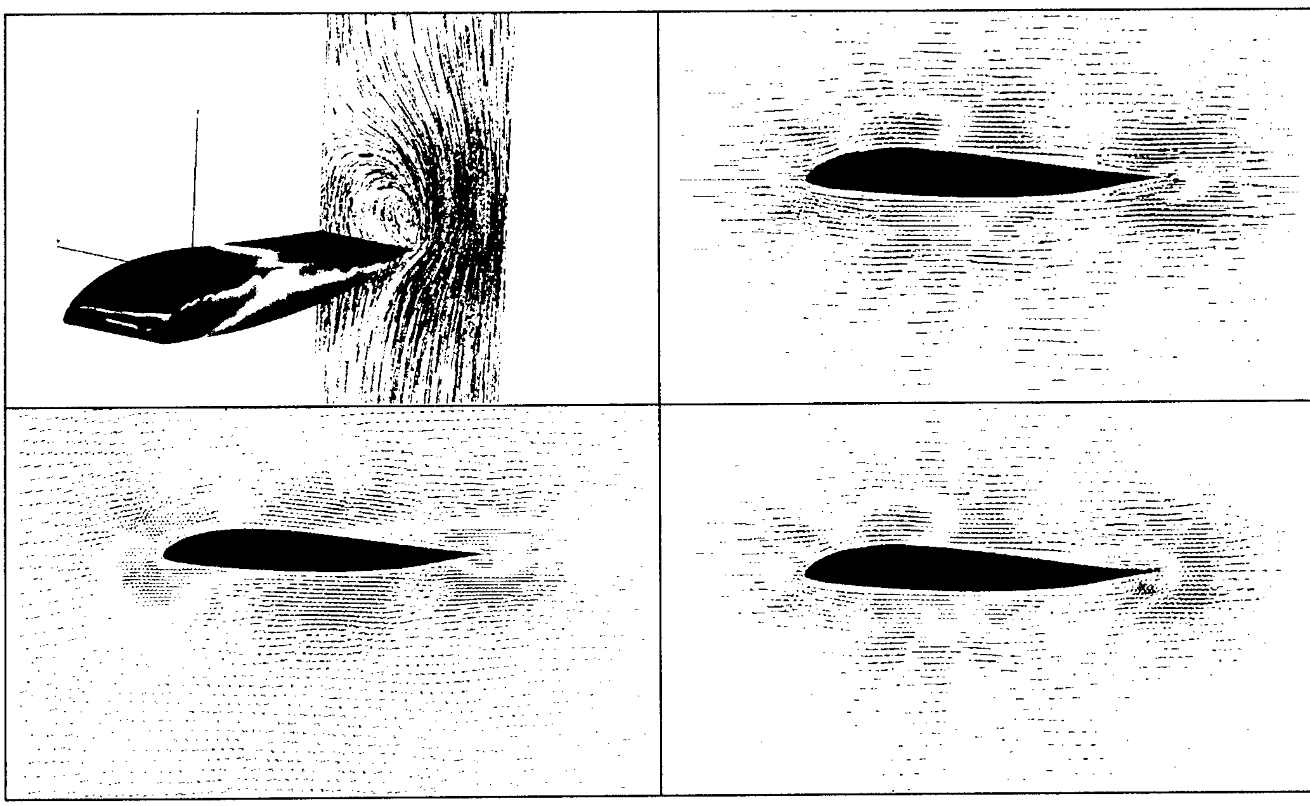
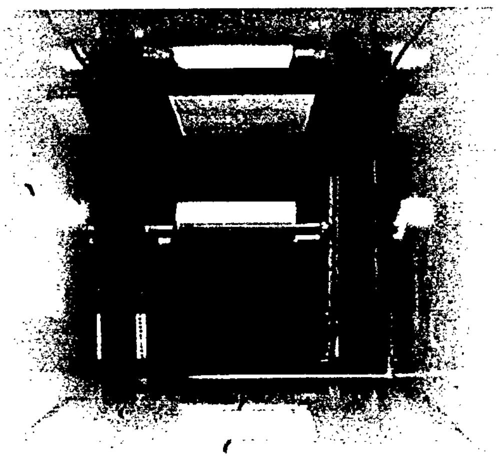


Figure 4-5: a) Velocity Vectors At Trailing Edge, b) 0 Degree Angle of Attack, c) 10 Degree Angle of Attack, d)10 Degree Negative Elevon Deflection

In order to verify the inviscid assumption of the CFD analysis, a full-scale model of the body wing was constructed. The model was allowed to pivot about its center of gravity and measurements of lift force and pitch moment were taken. The tests validated the results of the CFD analysis, as the aerodynamic center of the wing and the rate of change of the lift with respect to angle of attack were nearly identical.

The effectiveness of the elevon was also tested. The coefficient of moment varied as predicted with the theoretical elevon deflection.



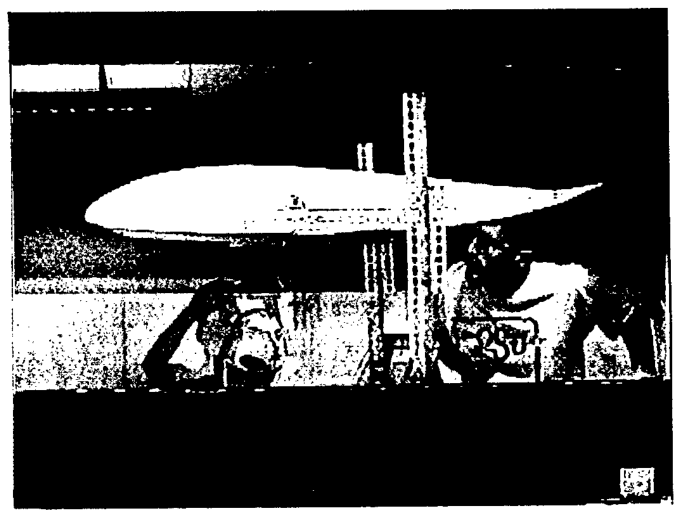


Figure 4-6: Wind Tunnel Testing

4.5 Analytical Methods

Since the longitudinal control methods of the aircraft are of an unconventional nature, further testing was conducted to verify the theoretical results. Fine Grid-Meshing techniques were implemented on the aircraft through the use of STARS CFD program. With this computational tool, the methodology in which to solve the aerodynamic uncertainties became more apparent. Coefficients were generated for lift, pitch moment, yaw moment, roll moment, and the control surfaces. As this process involves an inviscid assumption, a full-scale model of the body wing and the elevon was created to use in the wind tunnel for verification.

4.5.1 CFD Analysis

As mentioned above, the complete dimensions of the aircraft, including the ailerons, rudder, and elevon, were inserted into the STARS Computation Fluid Dynamics Program (Figure 4-7). The program delivered all the coefficients necessary for dynamic flight modeling.

The study confirmed the majority of the assumptions made in the calculations of the aircraft's stability. The pitch moment coefficient behaved much in the same way as predicted by the mathematical modeling. Slight adjustments were needed to the rate of change with respect to angle of attack and trim angle. The reaction of the velocity profile with angle of attack (Figure 4-7), and thus coefficient of lift, was also congruent with theoretical predicted values.

However, several differences were noted. The vertical tail was predicted to have a $C_N\beta$ of 0.04 while the CFD analysis determined that it was 0.17. This increase can be attributed to several reasons. The first is the higher coefficient of pressure across the top of the body wing as opposed to a conventional aircraft.. Secondly, the body wing acts much like an endplate to the vertical tail: the wide surface prevents flow around the body at the base of the vertical tail and adds to its effectiveness (Figure 4-8).

A major flaw in the design was revealed by the CFD analysis. It was assumed that the contributions of the tail and the fuselage to roll stability would be sufficient. However, this was not the case; the anhedral effect of the body wing was too much for the tail to overcome (Figure 4-8). This was corrected by altering the wing to include 2 degrees of dihedral. This correction changed the $C_L\beta$ of the aircraft to -0.008.

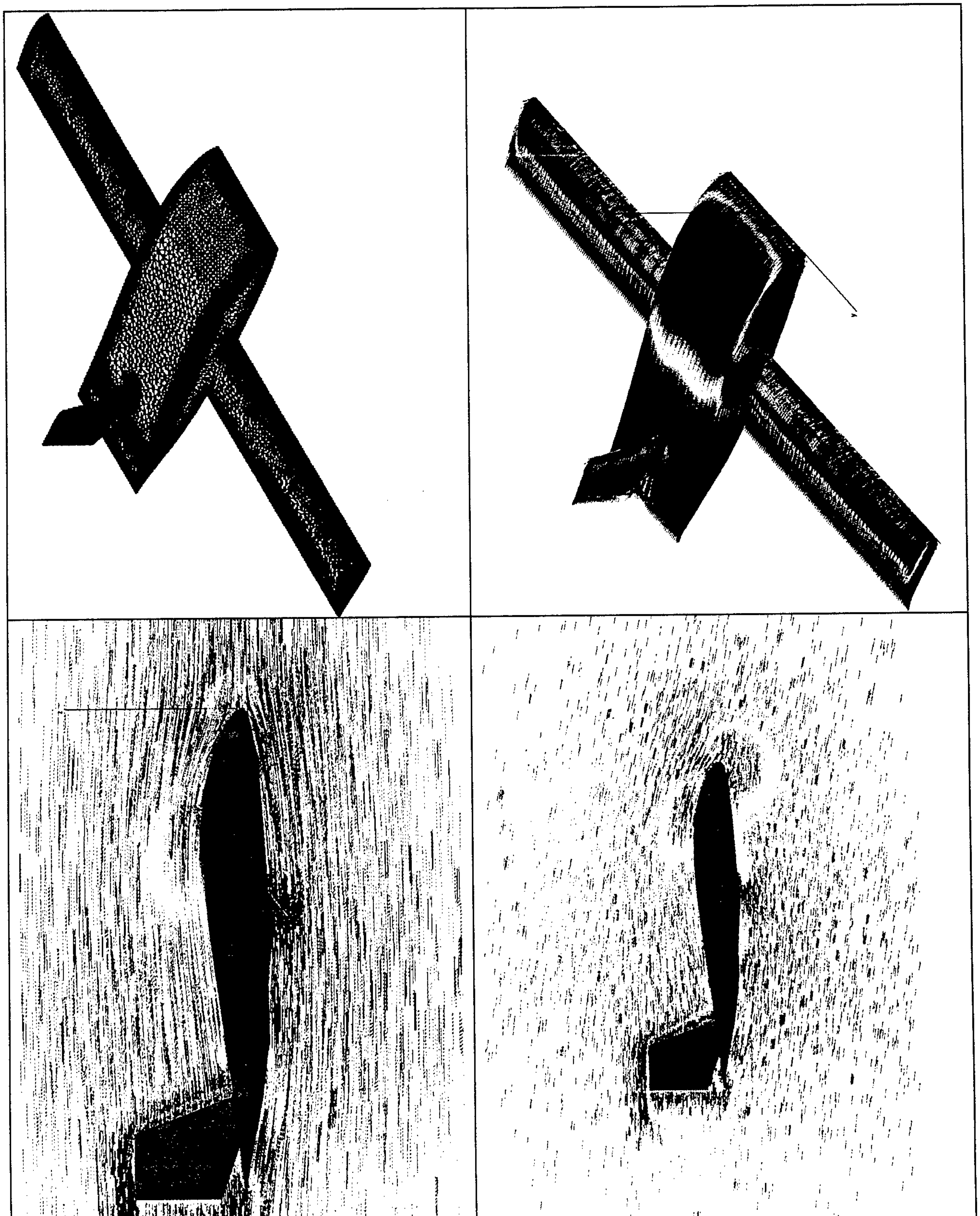


Figure 4-7: CFD Analysis – a) Finite Element Mesh, b) Pressure Distribution at Zero Angle of Attack, c) 0 Degree Angle of Attack, d) 10 Degree Angle of Attack

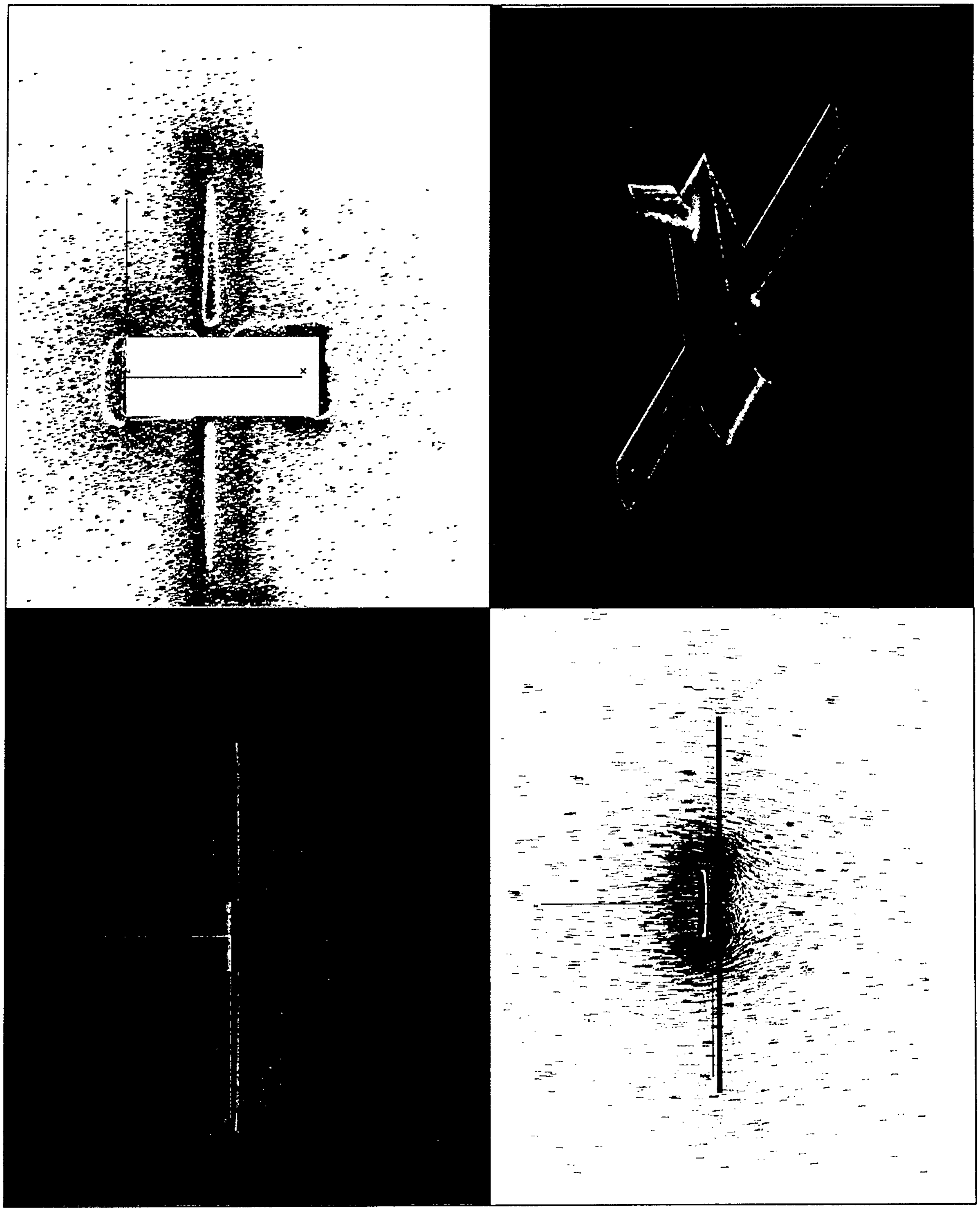


Figure 4-8: CFD Analysis – a)10 Degree Sideslip flow field, b)10 Degree Sideslip Pressure Distribution, c)Quarter Chord Pressure Distribution, d)Velocity Profile at 5 in. From Leading Edge of Body

4.6 Wing and Power Loading Statistics

With the wing completely sized and the propulsive system optimized, the structural aspects of the wing were then analyzed. The aircraft has to meet necessary structural requirements in order to achieve the required aircraft performance needed to achieve the predicted score. These range from supporting weight and thrust at takeoff to the lift, drag coefficient and g-load at cruise. These requirements also change with the presence of payload.

Flight Portion	Parameter	Value
Empty	Wing Loading (lbf/ft2)	1.87
	Power Loading (lbf/hp)	39.76
	Range at Cruise (ft)	24200
	Coefficient Lift Cruise	0.30
Full	Wing Loading (lbf/ft2)	3.37
	Power Loading (lbf/hp)	71.57
	Range at Cruise (ft)	21900
	Coefficient Lift Cruise	0.65

Table 4-5: Wing and Power Loading Stats

4.6.1 Methods of Wing Reinforcement

The wing is one of the most critical aspects in structural design for three reasons. The first is that it is a vital component in the flight of an aircraft, and if damaged or lost can cause loss of control, flight, or even the entire plane. The second reason the wing calculation must be carefully performed is because it is the part of the plane undergoing the highest forces. Lastly, the wing is an anti-structural object, meaning that the shape of the wing does not lend itself to be structurally efficient. It is a very long and slender body with little area moment of inertia in the crucial stress orientation.

Due to the heterogeneous composition of the wing, it has been analyzed using advanced beam theory. This theory uses modulus-weighted equations to determine the actual stress in each different type of material. Because of the complexity of the airfoil, analysis required that the cross-sectional shape be simplified. The simplified cross-section had three main components. The first simplification was a semi-circular leading edge or "D-section". Its diameter was 95 percent of the maximum chamber height. Next the main section was modeled in the shape of a rectangle, again using 95 percent of the max chamber as its height. The length of the rectangle was ½ the chord less the radius of the semi-circle. The final segment of the simplified airfoil was the trailing edge, which tapered down to a point at the end of the wing.

Subsequently the area moment of inertia was calculated. Numerous steps were required to accurately calculate this value due to the fact that the advanced beam theory requires that each type of material be calculated separately. Then, using the reference modulus, all were combined into a weighted area

moment of inertia. With the inertia known, the stress and factor of safety for each section was determined. Thus, given these values, the tip deflection was found. This process was repeated for three important configurations. The first was a foam airfoil cross section wrapped with a single ply of carbon fiber skin. The second is the same type of lay-up, but differs only in the fact that it uses a spar to carry a fraction of the load. The third neglects the use of a spar but instead, uses carbon fiber "doublers" to gain the most moment of inertia, i.e. the farthest from the modulus weighted centroid. A comparison of each type is given in Table 4-6. When comparing the second and third configuration, the additional material for the spar equals the additional material for the doublers.

Configuration	I_{zzstar} (in^4)	(%diff.)	FS_{skin}	(%diff.)	δ_{tip} (in)	(%diff.)	Weight Increase
#1 Skin Alone	0.04033	0.00	10.97	0.00	1.235	0.00	0 lbs
#2 Skin w/ Spar	0.05259	30.4	14.303	30.4	0.947	-23.3	.34 lbs+balsa
#3 Skin w/ Doubler	0.06706	66.3	18.239	66.3	0.743	-39.8	.34 lbs

Table 4-6: Comparison of Construction Methods for Wing

It was determined that the reinforcement would be necessary, either by the spar or doubler addition. The results of this analysis indicated that the optimum configuration was the skin reinforced with doublers.

4.7 Assumptions Made

In this section of the preliminary, many assumptions were made to advance the design process.

- Sail wing theory was used for the sizing of surface. Since the aspect ratio of the wing was greater than 10, the theory applied.
- Momentum Disk theory was used to calculate the increase efficiency of the body wing.
- Doublers were assumed to be sufficient support for the forces exerted on the wing. Experience in previous projects supported this rationalization.
- The flow about the aircraft was assumed to be inviscid by the CFD analysis. This assumption was proven with wind tunnel testing.
- A weight model was derived for the simulation program. This was validated after construction in later phases of design. An error of 5% was found.

4.8 Comparisons of Assumptions from Conceptual Design

The flight analysis program used in conceptual design made multiple assumptions, all of which were validated in this phase of design.

- A possible Cl_{max} was assumed to be attainable. With the selection of an airfoil, this was confirmed.
- The parasitic drag of the aircraft was calculated as 0.024, 67% of the previously assumed value.
- The fuselage length increased from 3.5 in the analysis to an actual value of 3.75. The increase was attributed to the propeller extending from the leading edge.

- A 32% efficiency was assumed to be possible. The aerodynamics and propulsions teams mutually determined that this value was accessible.
- The capacity of the battery was assumed to be 2400 mAh. This was validated by the selection of the SR-2400 battery.

5 DETAIL DESIGN

After the Preliminary Design phase was complete a more detailed analysis of the airplane was required. The Detail Design section accounts for specific component selection, landing gear analysis, g-load study, motor selection, and propeller selection. These areas, above all, required specific attention before they were implemented. This airplane is an aerodynamically dependent design and detailed analysis of the expected mission and dynamic performance are central to this section.

5.1 Expected Mission Performance

With the aircraft completely sized and all components selected, the final estimations for mission performance were generated using the flight analysis program.

5.1.1 Sortie Performance

The subroutines used to develop the plane's characteristics generated data that corresponded to sorties conducted with and without the presence of payload. The addition of the payload requires more runway length for takeoff and more power for the entire flight time. The aircraft also travels slower during this sortie due to the heighten levels of induced drag, as seen in Table 5-1

		Empty (per sortie)	Full	Total
Velocity (ft/s)	Cruise Velocitv	83	75	--
	Take off Velocity	37	55	--
	Climb	44	58	--
Power (watt)	Takeoff	211	211	--
	Cruise	140	140	--
Miscellaneous	Turn Radius (ft)	55	113	--
	g-load	1.89	4.86	--
	Stall Speed (ft/s)	37	49	--
	Takeoff Distance (ft)	34	168	--
Time (sec)	Cruise	44	78	166
	Takeoff	2	9	13
	Landing, Turns, Takeoff	25	34	84
	Total	71	121	263

Table 5-1: Mission Performance

5.1.2 Complete Mission

The total estimated time for a mission is 5.7 minutes. This includes a total airtime of approximately 4.3 minutes. With a payload of 24 balls, a flight score of 5.23 (note that the flight score is not multiplied by 10) results for each mission flown. The propulsive system reserves 10 percent of its maximum power for extreme wind conditions.

5.1.3 Dynamic Performance

Even though the mathematical modeling and CFD analysis illustrated a statically stable aircraft, it is equally important to consider how the airplane performs while actually in the air. The aircraft must be sufficiently stable dynamically to complete the requirements of the competition.

The stability and control coefficients were used to generate the frequency and damping coefficient for the modes of motion. The results are displayed inTable 5-2.

Mode	Damping Ratio	Natural Frequency (Hz)	Period (sec)	Time to Half (sec)	Eiganvalues	
					Real	Imaginary
Short Period	0.128	0.790	1.276	1.087	-0.635	4.931
Phugiod	0.032	0.076	13.094	45.282	-0.015	0.480
Roll	1.000	3.550	0.282	0.031	-22.303	----
Spiral	1.000	0.008	119.474	13.12 (double)	0.053	----
Dutch Roll	0.036	1.010	0.990	3.033	-0.227	6.345

Table 5-2: The Five Modes of Motion

5.1.3.1 Dynamic Simulation

Due to the unconventional nature of the aircraft, it was determined that stability coefficients alone were not satisfactory for guaranteed success. As such an applicable six degrees-of-freedom dynamic modeling program was created that utilized the stability coefficients derived by CFD analysis mentioned in Section Four and modeled by theoretical equations.

The program was designed assuming the aircraft would react with a wind gust in any or all of the three-principle axes. A wind gust of 15 ft/s was applied in all three directions. The initial conditions were assumed to be at straight and level flight. According to Figure 5-1, the aircraft has a maximum horizontal displacement of 240 feet and finishes 10 feet away from its initial position. A maximum vertical displacement of 30 feet is observed. Both motions show visible signs of damping.

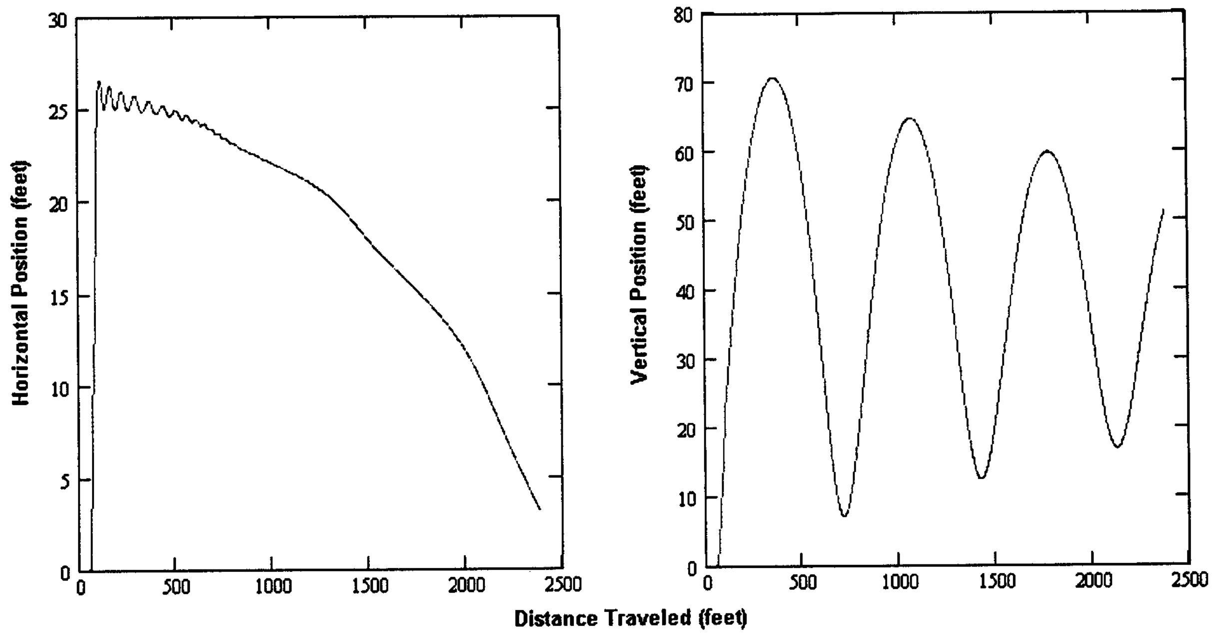


Figure 5-1: Wind Gust Simulation

5.1.4 G-Loads

The design parameters have a direct effect on flight performance. As the aircraft increases in cruise velocity the acceleration in the turns increase. This creates higher g-loads, which in conjunction with the weight of the aircraft, produces an additional force that the aircraft must be structurally prepared for. The quantitative g-load numbers are presented in Table 5-1.

5.2 Weight and Balance Sheet

Table 5-3 is a detailed account of component weights and their respective distances from the origin. This table gives a total loaded weight of 21.56 pounds and a center of gravity along the x-axis of 16 inches from the nose. This table is shown below.

Component	Mass (lbs)	X distance (inches)	Y distance (inches)	Z distance (inches)
Left Wing	1.500	19.000	47.250	-2.000
Right Wing	1.500	19.000	-47.250	-2.000
Left Aileron Servo	0.046	19.000	23.625	-2.000
Right Aileron Servo	0.046	19.000	-23.625	-2.000
Fuselage	3.500	16.000	0.000	0.000
Forward Bulkhead	0.000	0.000	0.000	0.000
Prop	0.132	-1.000	0.000	2.000
Motor	0.875	2.000	0.000	2.000
Battery Pack (7 cells)	0.945	15.000	0.000	-2.000
Battery Pack (8 cells)	1.080	23.000	0.000	-2.000
Rudder Servo	0.046	34.000	-2.000	0.000
Elevon Servo	0.046	34.000	2.000	0.000
Forward Gear	0.357	4.000	0.000	-5.500
Main Gear	0.185	18.000	0.000	-5.500
Watt Meter	0.254	2.000	6.000	-1.000
Speed Controller	0.146	2.000	-6.750	-1.000
Gear Box	0.125	1.000	0.000	2.000
Forward Wheel	0.115	4.000	0.000	-9.000
Main Wheel	0.115	17.000	-8.500	-9.000
Main Wheel	0.115	17.000	8.500	-9.000
Vertical Tail	0.132	34.000	0.000	7.000
Motor Shroud	0.066	2.000	0.000	2.000
Servo Mount	0.066	34.000	0.000	0.000
Tail Plate	0.055	34.000	0.000	1.000
Left Tail Plate Support	0.004	30.000	-2.000	0.000
Right Tail Plate Support	0.002	30.000	2.000	0.000
Receiver	0.090	36.000	-6.000	0.000
Receiver Battery Pack	0.205	36.000	0.000	0.000
Right Brake	0.143	17.000	-8.500	-9.000
Left Brake	0.143	17.000	8.500	-9.000
Brake Controller	0.044	36.000	6.000	0.000
Nose Spinner	0.033	-3.000	0.000	2.000
Nose Gear Mount	0.044	4.000	0.000	0.000
Foam Egg Crate	0.300	16.000	0.000	-1.000
Fuse	0.063	2.000	-8.500	0.000
Rate Gyro	0.044	36.239	6.000	0.000
Softballs	9.000	16.000	0.000	-1.285
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

Table 5-3: Center of Gravity Calculation Using Component Mass and Distance in Relation to Nose of Body

Static Margin is 12.1%. A typical transport aircraft has a static margin of 5% to 10% (Raymer 1999).

5.3 Component Selection and Systems Architecture

Once the aircraft configuration was known the internal component placement for the aircraft was determined. Each onboard component was placed within the aircraft in conceptual drawings. The items were as closely to scale as possible to ensure proper fit. The Drawing Package 5.5 illustrates the layout of all internal components. The batteries can be arranged in a variety of configurations utilizing the spaces between the balls. The ability to move the batteries promotes an adjustable center of gravity. Of the four servos, one was installed on each of the wings to control the ailerons, whereas the remaining two were placed at the tail of the plane. The first one controls the elevon and the other controls the vertical stabilizer and the landing gear. The design includes one motor and one propeller, which are installed at the front of the plane in a motor encasement cylinder. A wattmeter to check the amperage of the batteries and a speed controller are installed at the front of the plane. In accordance with contest rules a fuse is installed at the front. Fifteen batteries are used, which are placed in the fuselage under the softballs. Magnets are laid up within the hatch and fuselage to secure the hatch closed. A hinge was installed at the front end of the fuselage door. A brake controller, receiver batteries, and a receiver were placed at the back end of the plane.

5.3.1 Servo Selection

To select an appropriate servo, an analysis of all possible hinge moments was conducted. The hinge moment due to aerodynamics was present on the elevon at 26.134 oz.-in. The hinge moments on the ailerons and rudder were in the 0 to 7 oz-in. range. The ailerons, the rudder and the nose gear are powered by Hitec 85MG Mighty Micro servos rated at 49 oz-in and the elevon is powered by a Hitec 225MG Mini servo rated at 67 oz-in torque.

5.3.2 Landing Gear Design

The landing gear is a tricycle configuration, with two main wheels underneath the main wing and a nose wheel near the front of the fuselage. There are five categories that relate to landing gear detailed design. The first gives an overall configuration followed by a description of the breaking system that was implemented. The next category analyzes a possible worst-case drop load the gear must sustain and this section is followed by bolt shearing analysis. The final category stresses the importance of takeoff pitching moment.

5.3.2.1 Configuration

In the preliminary phase, an angled "bow" construction with carbon fiber skin and a honeycomb core was decided upon. Placement of the gear was determined by calculating the overturn angle and required pitching moment for takeoff. It was found that the landing gear could vary between 0 to 0.5 inches from the center of gravity. With these numbers the following was obtained:

Distance between nose gear and main gear = 15.27 inches

Distance between nose of airplane and nose gear = 3.31 inches

The height of the main landing gear was determined by the propeller clearance needed. This height was set at 7 inches. The upper horizontal surface of the main gear, which will connect flush to the wing, is 16 inches across. The gear will then angle down and come to a vertical strut that is 2-½ inches long. The angle will taper from 2-½ inches to 1-¼ inches at the strut. The struts will carry the axle and a break sensor that is described in the next section.

For takeoff, aerodynamic analysis determined that it was desirable for the nose gear to be slightly taller than the main gear. An adjustable, two-stroke, strut shock absorbing gear was used for the nose gear. The nose gear is attached to the rudder servo through the use of a push-pull cable.

5.3.2.2 Brake System

Landing and loading time are very important in acquiring the best overall score. A differential breaking system was implemented in order to shorten landing distance. A reflective sensor is mounted to the landing gear above the wheels and a fabricated reflective surface is attached directly to the wheels. The break control circuitry senses the rotational velocity of each wheel on the main gear, maintains equal rotational velocity, and prevents skidding.

5.3.2.3 Drop Load

The updated weight model gives a total aircraft/payload weight of 21.56 pounds. Through the use of a ply-mechanics program, the stress and strain were calculated for each ply and the landing gear as a whole. With this program, the updated weight model, and historical data, it was found that 12 plies of carbon fiber and 1/8 inch honeycomb would suffice for a drop-impact at 60 inches.

5.3.2.4 Bolt Shearing

The braking system creates a force on the tires, which will translate a moment to the bolts. As mentioned previously, the airplane weighs 21.56 pounds and the rubber tires have a coefficient of friction of 0.8. There is a 7-inch moment arm from the bolts to the tires. This moment is decoupled by a 1.5 inch staggered spacing of the bolts. When using three bolts, calculation shows that a 50-pound tensional force to each of the bolts is applied. With a safety-factor of two, the final bolt design must withstand 100 pounds of tension.

5.3.2.5 Takeoff Pitching Moment

While the stability of the aircraft in the air was an important design consideration, the ability of the aircraft to rotate about the main gear while taking off was also extremely important. The placement of the gear was necessary to insure a proper weight distribution of 7 to 10 percent to the front gear. However, by placing the main gear too far forward, the aircraft would not be able to pitch up about the main gear. From calculations, it was determined that the landing gear could be placed between 16.9 and 17.3 inches from the leading edge of the body wing. The final wheel location was selected as 17 inches, putting 7.6

percent on the front gear. In order to achieve this and maintain the same mounting location as the wing, the main landing gear is swept forward 1.8 inches.

5.3.3 Motor Selection

According to the competition rules, motors can be chosen from two companies and must be brush type motors. The companies are Astroflight and Graupner. OSU has a long-standing relationship with Astroflight and receives a substantial discount on their motors. Astroflight is located in California, while Graupner is a German company, so choosing a motor from Graupner would add logistical concerns.

Out of the various series of motors that Astroflight produces, the 25-series and the 40-series motors operated at the planned voltages and currents. When comparing motors, there are two important parameters to consider, the mechanical power out of the motor and the motor efficiency. The mechanical power out, torque, and motor efficiency with the SR-2400 battery are compared in Figure 5-2.

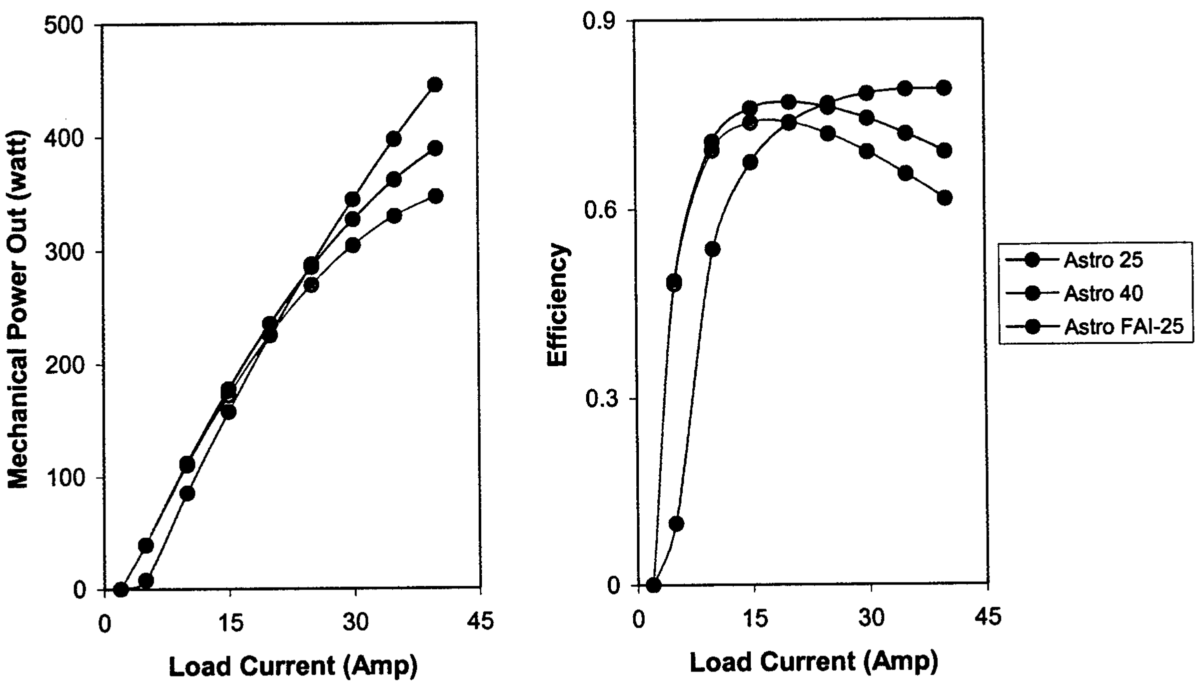


Figure 5-2: Motor Power and Efficiency

From above, it is obvious that the Astro FAI-25 gives better performance with in the range of interest of 20 to 40 amps. Moreover, it is lighter than the Astro 25 and Astro 40. Another advantage over the Astro-25/40 is that it was designed to operate within the number range of cells that were being considered. The following decision matrix, Table 5-4, was formed to help us in our decision of selecting between the Astro FAI-25, Astro-25 and Astro-40.

Figures of Merit	Weight	Astro 25	Astro 40	Astro FAI-25
Power Output	.2	0	0	1
Efficiency	.3	1	0	1
Max current load	.1	1	1	1
Cost	.1	1	1	0
Weight	.2	0	0	1
Availability	.1	1	1	1
Total	1.0	.6	.3	.9

Table 5-4: Motor Decision Matrix

5.3.4 Propeller Selection

The investigation for the best possible propeller was conducted by performing experiments on different prop sizes ranging from a diameter of 12 inches to a diameter of 15 inches with varying pitch sizes. The data from these experiments was used to make the final choice.

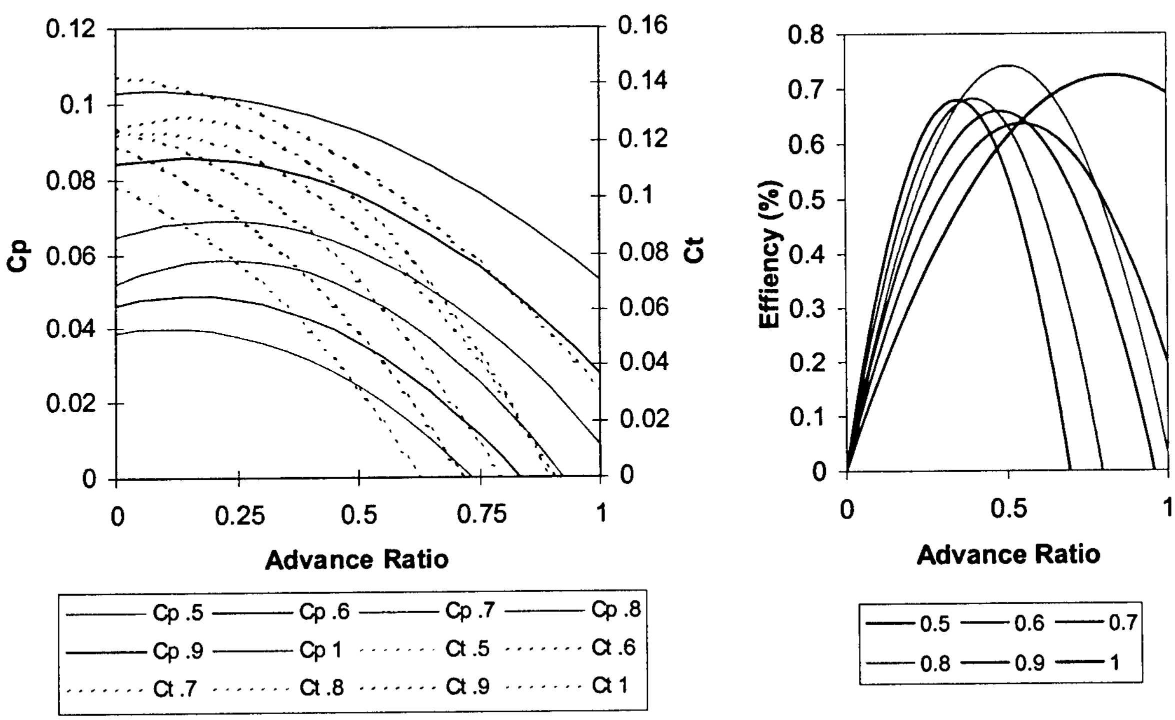
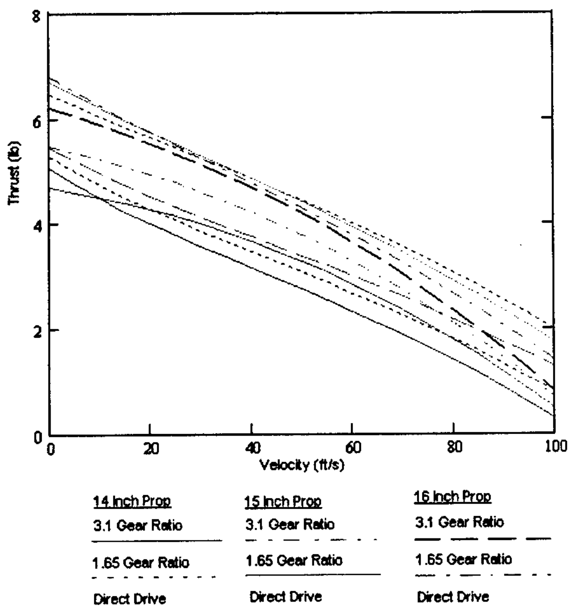


Figure 5-3: Propeller Studies

The primary analysis of the propellers came from performance curves generated for 'ideal' propellers. A program was written that varied propeller pitch to diameter ratio (P/D), gear ratios (G), and propeller diameters then determined the thrust produced, overall efficiency of the battery-motor-propeller system, current drawn, power available from the batteries, and speed of the airplane. All of these parameters were found for both empty and loaded sorties. The graphs in Figure 5-3 show how the available power changes with P/D at different C_T, C_P, and η values. These graphs were used to define the trends of the varying P/D ratios and their affect on the performance of the propulsion system as a whole.



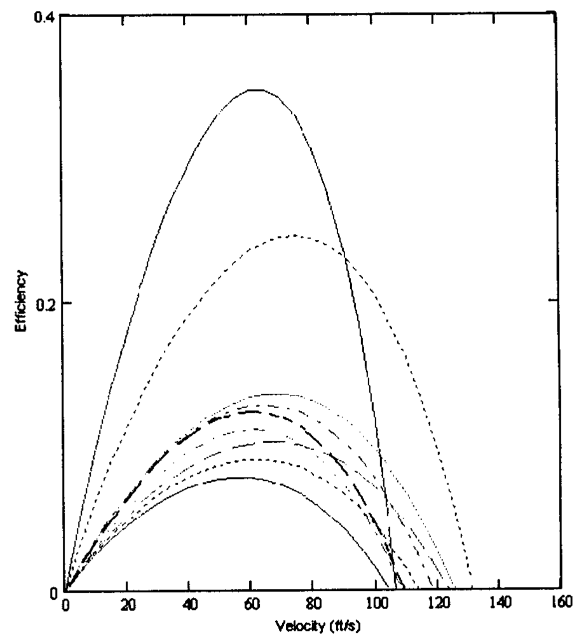


Figure 5-4: Propeller Thrust and Efficiency

The graphs in Figure 5-4 suggested a reasonable range of propellers, which will provide the necessary power and efficiency for the flight profile. Takeoff distance governs the thrust condition and sufficient power is required to takeoff within 200 feet. Likewise, efficiency is important during cruise. Cruise time is increased by an increase in propulsive efficiency. However, as the graphs show, it is not possible to achieve both. A balance must be attained for success in the competition. As the 3.1:1 gear ratio will fulfill the necessary requirements for thrust, it is the most logical choice with its high frequency.

5.4 Assumptions Made

With the design nearing completion, fewer assumptions were needed to progress the design.

- A drop load of 5 feet was assumed to be sufficient for the landing gear.

- Small Disturbance Theory (Nelson 1998) was used for the dynamic aspects of the simulation. This application holds true as long as the aircraft does not enter a stall, spin, or other large amplitude motions.

5.5 Final Characteristics

Table 5-5 gives a detailed list of major airplane characteristics and their respective dimensions. Additionally, Table 5-6 gives a complete summary of RAC for the final airplane design.

Wing	Airfoil	Selig-Donovan 7062
	Area	6.015 ft^2
	Span	7.875 ft
	Chord	9.174 in
	Angle of Incidence	0^o
	Aspect Ratio	10.29
	Aileron Area	0.344 ft^2
Body	Airfoil	Eppler E326
	Length	3.5ft
	Area	4.958 ft^2
	Aspect Ratio	0.405
	Elevon Area	.4958 ft^2
Vertical Tail	Airfoil	NACA 0009
	Area	.494 ft^2
	Root Chord	9.912in
	Span	9 in
	Taper Ratio	0.6
	Rudder Area	.1482ft^2
Propulsion	Battery	SR-2400
	Number of Batteries	15
	Motor	AstroFlight FAI-25
	Propeller Diameter	14 in
Weight	Gross Take Off Weight, no cargo	11.25 lb
	Gross Take Off Weight, payload	20.25 lb.

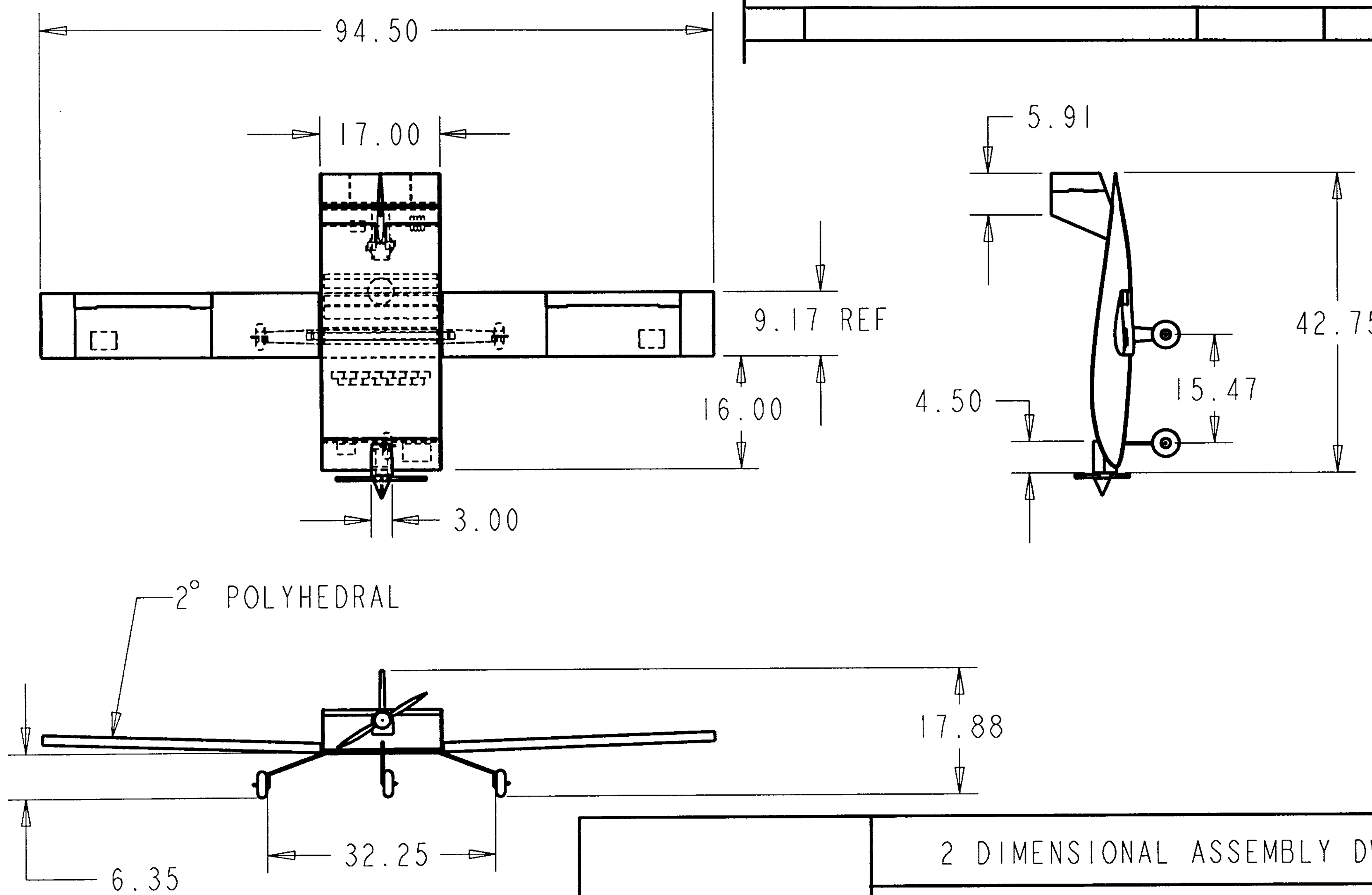
Table 5-5: Final Characteristics

5.6 Drawing Package

The Drawing package follows on the following four pages.

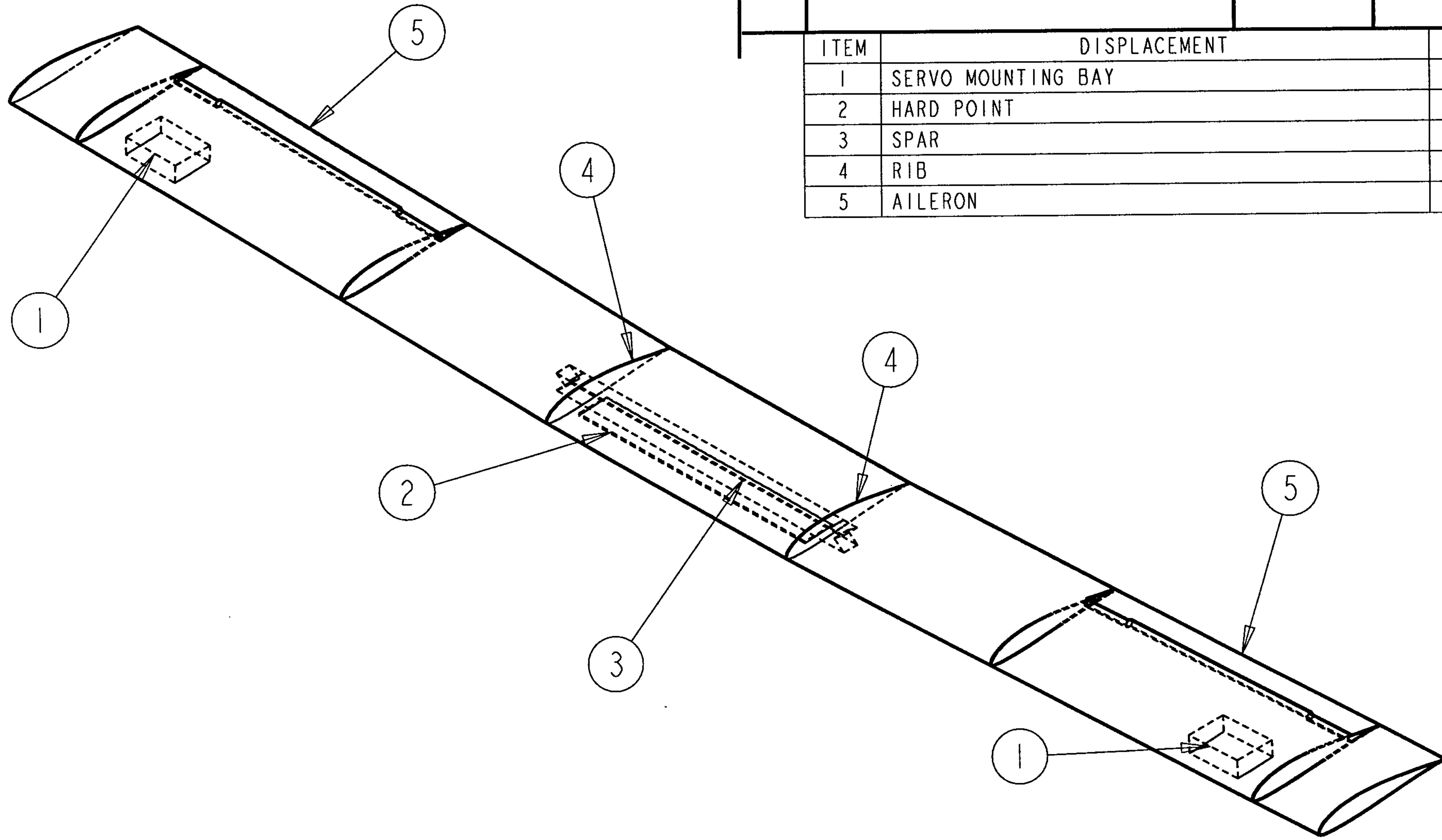
5.7 RAC Worksheet

Description	Supplied Cost Model	Aircraft Parameter	Man Hour Score	Coefficient Cost Multiplier	RAC Contribution ($ thousands)
MEW	Actual airframe weight, lb., without payload	11.525 lb.	--	$100/lb.	1.153
REP	**# Engines** **Battery Weight**	**1 motor** **2.025 lb.**			
	REP = (1+.25*(# Engines - 1))*Battery Weight	**2.025 lb.**	**--**	**$1500/lb.**	**3.038**
MFHR	WBS 1.0 Wing (s)				
	8 hr/ft Span 8 hr/ft Max Chord 3 hr/Control Surface	7.83 ft .77 ft 3 Control Surfaces	62.7 6.1 9		
	Subtotal of Manhours for 1.0 Wing (s)		77.8	$20/ManHr	1.556
	WBS 2.0 Fuselage				
	10 hr/ft of Length	**3.75 ft**	**37.5**		
	Subtotal of Manhours for 2.0 Fuselage		**37.5**	**$20/ManHr**	**0.750**
	WBS 3.0 Empennage				
	5 hr/Vertical 10 hr/Vertical Surface With Control 10 hr/Horizontal Surface	0 Vertical Surfaces 1 Vertical Surface With Control 0 Horizontal Surfaces	0 10 0		
	Subtotal of Manhours for 3.0 Empennage		10	$20/ManHr	0.200
	WBS 4.0 Flight Systems				
	5 hr/Controller **5 hr/Servo**	**1 Controller** **4 Servos**	**5** **20**		
	Subtotal of Manhours for 4.0 Flight Systems		**25**	**$20/ManHr**	**0.500**
	WBS 5.0 Propulsive Systems				
	5 hr/Engine 5 hr/Propeller	1 Motor 1 Propeller	5 5		
	Subtotal of Man hours for 5.0 Propulsion Systems		10	$20/ManHr	0.200
	MFHR = Sum of WBS hours	**Total Manhours:**	**160.3**		
				RAC:	7.396

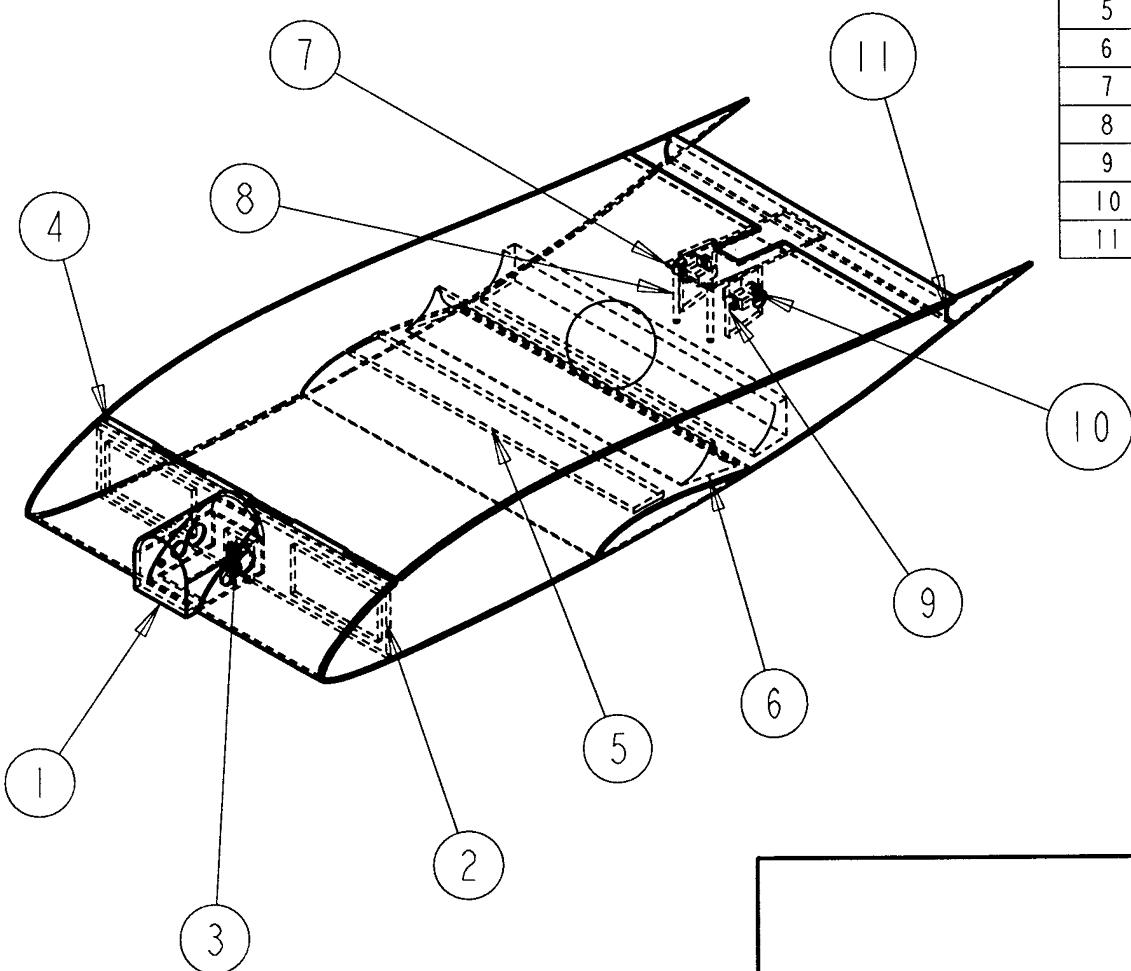


2 DIMENSIONAL ASSEMBLY DWG	
OSU BLACK TEAM	
SCALE 1:20	DRAWN BY: NG
MARCH 12, 2002	SHEE[illegible] OF 4

ITEM	DISPLACEMENT	QTY.
1	SERVO MOUNTING BAY	2
2	HARD POINT	1
3	SPAR	1
4	RIB	6
5	AILERON	2



MAIN WING	
OSU BLACK TEAM	
SCALE 1:8	DRAWN BY: NG
MARCH 12, 2002	SHEET 4 OF 4



ITEM	DESCRIPTION	QTY.
1	MOTOR MOUNT	1
2	FRONT BULKHEAD	1
3	NOSE GEAR MOUNT	1
4	HATCH HINGE	1
5	HARDPOINT	1
6	CARGO RESTRAINING FOAM	1
7	VERTICAL TAIL MOUNT	1
8	MOUNTING ROD	2
9	SERVO MOUNT	2
10	SERVO	2
11	REAR BULKHEAD	1

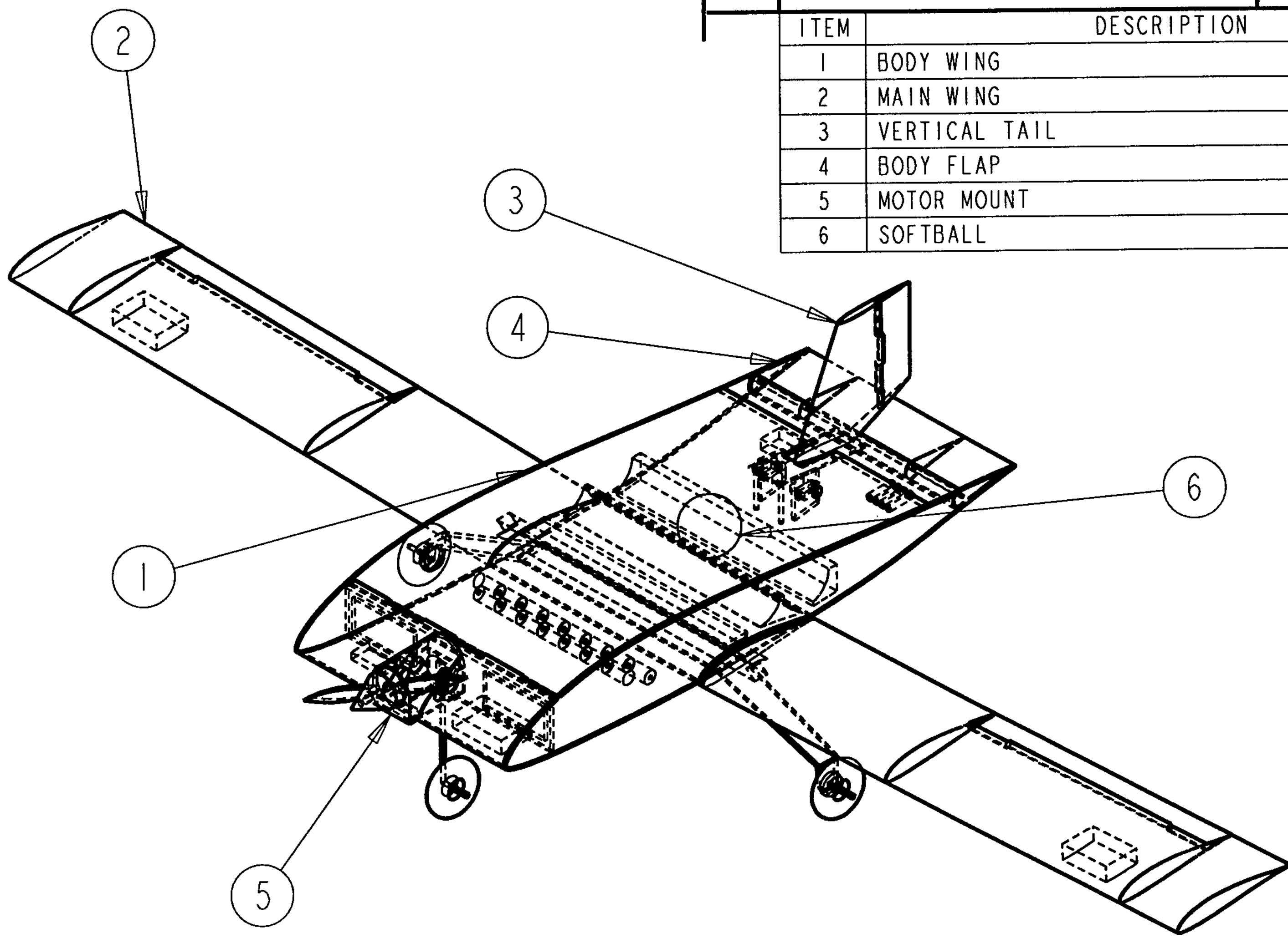
BODYWING

OSU BLACK TEAM

SCALE 1:8 | DRAWING BY: NG

MARCH 12, 2002 | SHEET [illegible] OF 4

ITEM	DESCRIPTION	QTY.
1	BODY WING	1
2	MAIN WING	1
3	VERTICAL TAIL	1
4	BODY FLAP	1
5	MOTOR MOUNT	1
6	SOFTBALL	24



ASSEMBLY DWG		
OSU BLACK TEAM		
SCALE 1:10	DRAWN BY: NG	
MARCH 12, 2002	SHEET 2 OF 4	

6 MANUFACTURING PLAN

Considering that the team's design can only succeed if the aircraft can be precisely fabricated, the structures group spent a large amount of time preparing a thorough manufacturing plan. Before finalizing a plan, the group considered other manufacturing options that are available. After a method was chosen for all the construction processes, a manufacturing schedule was created to help assist the structures group in time management. This schedule can be seen in Figure 6-1FOM's Used and Manufacturing Process Investigated

Because of the decision to construct the entire aircraft out of carbon fiber, the major decision to overcome was the type of mold to use for each major component of the airplane. The types of molds that the team considered included a typical male plug mold and a female mold. For either of these methods, a mold had to be produced that would conserve either the inner mold line (IML) or the outer mold line (OML) of the aircraft. If payload and internal components are tightly packed and, therefore, the internal dimension is most important, a male plug mold should be used to conserve the internal dimension. However, if the outer surface is more important because of aerodynamic concerns such as drag or sizing of major components, then a female mold should be used to conserve the smoothness of the outer surface and the outer dimension.

6.1.1 Figures of Merit Used for Manufacturing Processes

- Cost – Monetary concerns play a role in selection of a mold construction method because we must raise funds used for the project.
- Skill Required – The easier that the plane is to construct, the more time can be spent testing and perfecting the design's fabrication.
- Geometric Tolerance – The ability to get within a certain distance or tolerance of what was needed for either the IML or the OML on the mold.
- Surface Finish – The drag is reduced considerably and the aesthetics are improved if the surface finish is very smooth.
- Availability – The ability to obtain materials or services needed to construct the molds.
- Time Required – With the skills required, the learning curve of learning a new method, and the time lag in obtaining the materials; this is the total time that is expected to finish a mold.
- Ease of Construction – If the design of the mold is easily constructed, the time it takes to finish will be less and the design will likely be constructed well.

In mold design it is crucial to be able to remove the mold after curing it. Proper mold design and release techniques were planned as methods for this reason. For each of the main parts, a separate mold was required, including the upper and lower body wing, the main wing, and the vertical stabilizer. Necessary baking materials like purple release films, breather, bagging films and sealants were all acquired for carbon fiber lay-up.

CNC Machining: The benefits of using CNC machining to create our molds were that exceptional geometry tolerance and surface finish would be obtained. The equipment needed for this process was not available to the team however. This drawback increased the needed budget greatly since subcontracting the work out would be necessary.

Hot Wire Foam Cutting : OSU has extensive experience with wire cutting foam and making the corresponding templates needed to use this method. The wire cutting tools and knowledge have been passed down from previous OSU teams making it a relatively inexpensive process and a skill level match for this year's team. With the diameter of the wire used accounted for in the template, fairly good geometry tolerances were obtained in the foam cuts. Also, various foam sheets were readily available through local suppliers. After comparing the methods for mold fabrication through use of the decision matrix seen in Table 6-1, hot-wire foam cutting was chosen over its alternatives.

Figure of Merit	Weight Factor	CNC	Wire Cutting / Foam
Cost	0.1	-1	1
Skill Required	0.2	-1	1
Geometric Tolerance	0.1	1	0
Surface Finish	0.2	1	-1
Availability	0.2	-1	1
Time Required	0.2	0	1
Total	1.0	-0.2	0.5

Table 6-1: Construction Method Decision Matrix

Male Mold: As stated above, there were two main ideas for carbon fiber lay-up. The male plug idea involves cutting the foam out according to the IML, laying fiberglass and epoxy over it, and sanding it down. Then a sandwich of two plies of carbon fiber and 1/8-inch honeycomb is laid on the outside of the male plug along with a layer purple release on the outer surface of the top carbon fiber layer. This does not produce a nice outer surface finish.

Female Mold: The female mold was made by cutting the foam out using a template that conserves the OML and then baking multiple fiberglass layers over the two female pieces created by the foam cut. Then the fiberglass was polished and mold sealant and releasing agent was used in order to disengage the carbon fiber once baked. This does produce a nice outer surface finish, which reduces drag and adds visual appeal. The OML female is the industry standard for many parts. An OML female mold required additional strength in order for the mold to hold its dimension under vacuum. After comparing the two alternatives in the decision matrix found in Table 6-2, the female mold was decided upon.

Figure of Merit	Weight Factor	Male Mold	Female Mold
Geometric Tolerance	0.3	1	1
Ease of Construction	0.3	1	0
Surface Finish	0.4	-1	1
Total	1.0	0.2	0.7

Table 6-2: Mold Method Decision Matrix

6.2 Process Selected for Major Component Manufacture

For the prototype plane, a carbon fiber construction with a honeycomb core was used. The body was constructed of a carbon fiber-honeycomb sandwich while the wings and vertical stabilizer were made of carbon fiber around a blue foam core. The carbon fiber used was a combination of LTM24 and LTM25. A bake time of 12 hours at 140 degrees Fahrenheit with a ramp time of 2 hours at 120 degrees Fahrenheit was used to cure the structures.

6.3 Individual Processes Investigated

Each major component needed a slightly different type of mold and lay-up method as described below.

6.3.1 Body wing

The body wing mold includes both the upper and lower surfaces of the body wing from the foam cut. These are to be laid up separately on polished female molds. The lay-up for all body wing components is to be 1/8-inch honeycomb between two carbon fiber skins. The LTM24 was pre-impregnated only on one side of the carbon fiber, so this side was bonded to the honeycomb on the inside and the dry side was post-cured with a wet lay-up bake on the body wing. To create a guaranteed fit between body wing and the hatch, the hatch will be cut out after baking. Plastic hinges for the hatches will be put in place after lay-up. Quarter-inch diameter rare earth magnets were glued with epoxy inside the body wing to hold the hatches in place during flight. The sides of the body wing and the bulkheads were laid up on glass to ensure a good finish. The body wing and sides were secondarily bonded together after baking. Bulkheads were added in the same manner.

6.3.2 Hatch

The hatch by which the payload is to be accessed was hinged at the top, front of the body wing so that nearly the entire top of the body wing was hinged up as the hatch. This large hatch allows easy access to the payload. The hatch fastens in the closed position using rare earth magnets that were glued onto the inside of the hatch.

6.3.3 Wing

The wing was designed to use the foam itself for the airfoil core inside the wing with one ply of carbon fiber surrounding it. When cutting these foam cores, female molds were naturally cut. The female molds were then used to cradle the carbon fiber through use of thick Mylar on both the top and bottom surfaces of the wing. Using the Mylar, a smooth, shiny surface was obtained on the wing. A two-inch thick ply of carbon fiber was used as a doubler on top of the wing in place of spars.

An experiment was performed to determine the amount of resin flow in composite pieces that are cured at an incline because in order to fit the whole wing into the OSU oven, the wing must be turned on its side. It was determined that this was not a factor, so the main wing for our aircraft was laid up as one piece and cured in the existing oven. This saved weight compared to a wing that is laid up as two halves and

bonded together with a wet lay-up after baking. Laying up the wing as one piece saved over a day in construction of the prototype.

Doors for the servo compartments were laid up into the wing but the foam was not removed until after the baking process to ensure the airfoil shape. Balsa ribs were included near the servo door to provide a mounting surface for the aileron servos. Ailerons were cut from the wing foam before baking and were laid up with the wing after being separated from surrounding carbon fiber with purple release.

6.3.4 Vertical Stabilizer

The vertical stabilizer was attached using two supporting pieces made of a sandwich of carbon fiber/honeycomb. The supports were epoxied to the rear bulkhead and also attached to the inside of the body wing with two aluminum arrow shafts. Two nylon bolts were passed through the supports and hard points in the stabilizer to secure the stabilizer. This method allowed the vertical stabilizer to be completely modular.

6.3.5 Landing gear

The landing gear was baked on a mold made of foam also. Because of the pitch needed for takeoff about the main gear wheels, the landing gear was constructed with a forward sweep. The landing gear bolts will pass through the wing and into the body wing, thus securing the wing to the aircraft. The threaded brass inserts were epoxied into the landing gear after baking and drilling appropriately sized holes.

6.3.6 Motor Encasement

The motor mount was baked on a foam mold. The motor mount encasement was fabricated individually and added to the body wing with epoxy after a section was cut out of the pre-baked body wing. Holes were then drilled for motor mounting and for motor cooling.

6.3.7 Payload Restraint

To make sure the softballs do not move around in the cargo bay area, foam softball seats were cut to secure the balls during flight. A thin layer of spongy foam was epoxied to the inside of the hatch in order to secure the balls from the top.

6.4 Analytic Methods Including Cost, Skill Matrix, and Scheduling

Each of the discussed methods had different costs and skills required as discussed above. The time allotted for construction of the prototype was only about three weeks. Every hour is critical to successfully fabricate a working prototype. To CNC machine the molds would require too much time and would be too costly. This time could be used working with the carbon fiber and the secondary bonds. Making the molds with tooling dough would require a higher skill than the team had available and would cost more also. Foam cutting requires some time and necessary tools, but fortunately OSU owns the tools needed and the time is less than that of the alternatives. A skill matrix showing the number of team members with certain skills is shown in Table 6-3.

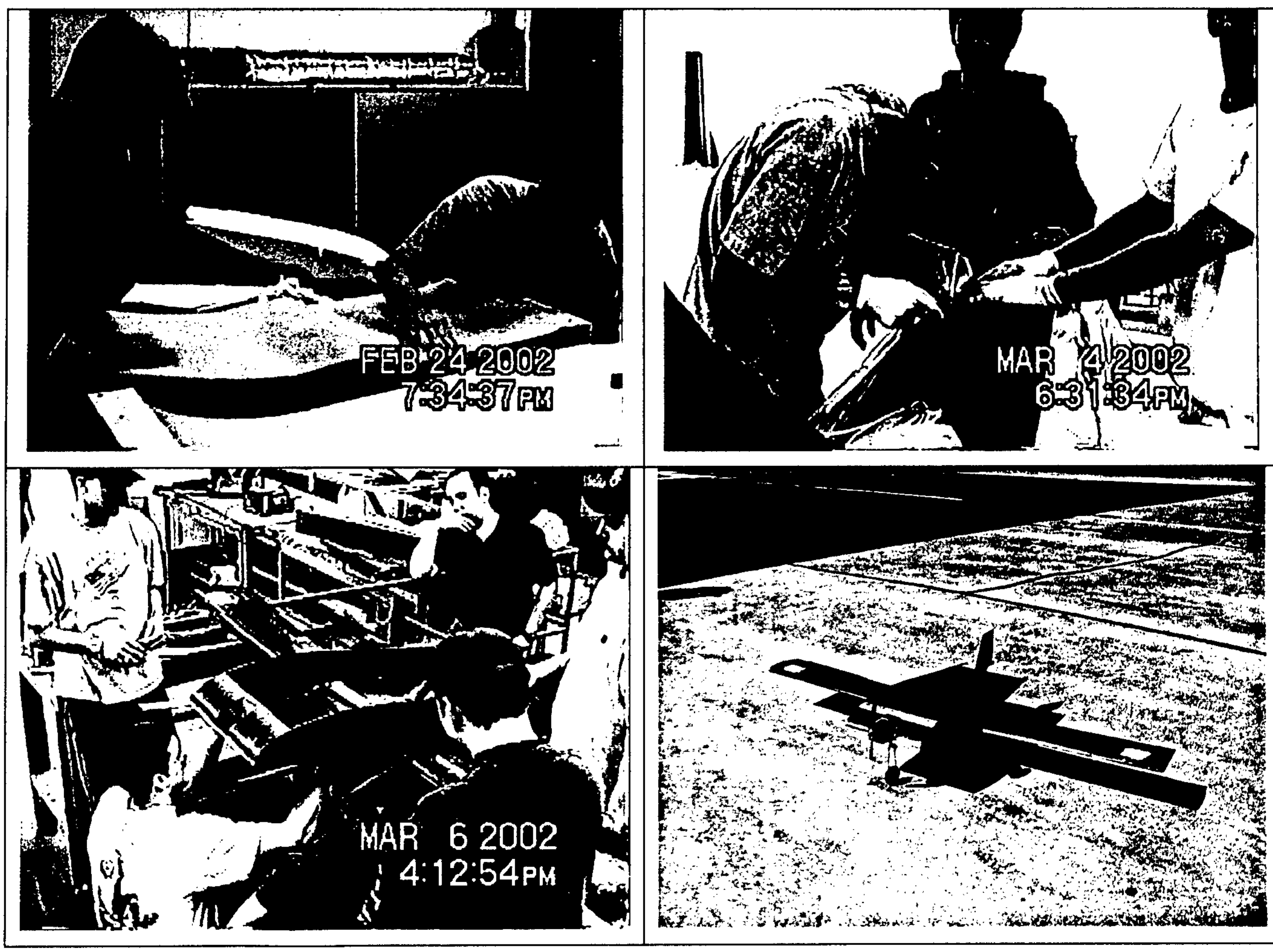
Available Skills Within Team	CNC Machining	Lathe/Mill Operation	Wood Working	Carbon Fiber Layup	Hot Wire Foam Cutting	Fiberglass work	MonoKote Application
Number of Team Members	0	18	18	16	6	5	4

Table 6-3: Skill Matrix

6.5 Aluminum Sheeting Mold

The Structures Team used a foam mold, which was hardened with epoxy for the wing and fuselage of the prototype. This type of mold left a dull finish on the carbon fiber. When carbon fiber is laid up on glass the finish results in a translucent surface that resembles smooth black glass. This was the desired finish for the entire airplane, but because of curved surfaces, glass could not be used. As a replacement, sixteen thousandth thick aluminum sheeting was placed over the foam mold. The aluminum would give the glossy look to curved surfaces, which greatly improves the aesthetics of the airplane.

6.6 Prototype Construction



ID	Task Name	Start	Finish
1	**Body Wing**	**Tue 2/19/02**	**Wed 2/27/02**
2	Body Wing Templates	Tue 2/19/02	Tue 2/19/02
3	Body Wing Templates (Actual)	Thu 2/21/02	Thu 2/21/02
4	Body Foam Cuts	Wed 2/20/02	Wed 2/20/02
5	Body Foam Cuts (Actual)	Sat 2/23/02	Sun 2/24/02
6	Body & Elevon Lay-up and Bake	Thu 2/21/02	Sat 2/23/02
7	Body & Elevon Lay-up and Bake (Actual)	Mon 2/25/02	Wed 2/27/02
8	**Main Wing**	**Tue 2/19/02**	**Fri 3/1/02**
9	Main Wing Templates	Tue 2/19/02	Tue 2/19/02
10	Main Wing Templates (Actual)	Thu 2/21/02	Thu 2/21/02
11	Wing Foam & Aileron Cuts	Tue 2/19/02	Tue 2/19/02
12	Wing Foam & Aileron Cuts (Actual)	Fri 2/22/02	Sat 2/23/02
13	Wing Lay-up and Bake	Mon 2/25/02	Mon 2/25/02
14	Wing Lay-up and Bake (Actual)	Thu 2/28/02	Fri 3/1/02
15	**Vertical Stab**	**Tue 2/19/02**	**Thu 2/28/02**
16	Vertical Stab Templates	Tue 2/19/02	Tue 2/19/02
17	Vertical Stab Templates (Actual)	Wed 2/20/02	Wed 2/20/02
18	Vertical Stab Foam & Rudder Cuts	Wed 2/20/02	Wed 2/20/02
19	Vertical Stab Foam & Rudder Cuts (Actual)	Sat 2/23/02	Sun 2/24/02
20	Vertical Stab Lay-up and Bake	Mon 2/25/02	Mon 2/25/02
21	Vertical Stab Lay-up and Bake (Actual)	Wed 2/27/02	Thu 2/28/02
22	**Landing Gear**	**Tue 2/19/02**	**Thu 2/28/02**
23	Create Landing Gear Molds	Tue 2/19/02	Tue 2/19/02
24	Create Landing Gear Molds (Actual)	Mon 2/25/02	Mon 2/25/02
25	Landing Gear Lay-up and Bake	Thu 2/21/02	Sat 2/23/02
26	Landing Gear Lay-up and Bake (Actual)	Wed 2/27/02	Thu 2/28/02
27	**Structural Components**	**Thu 2/21/02**	**Thu 3/7/02**
28	Bulkhead Lay-up and Bake	Thu 2/21/02	Sat 2/23/02
29	Bulkhead Lay-up and Bake (Actual)	Mon 2/25/02	Tue 2/26/02
30	Edge Trimming	Wed 2/27/02	Wed 2/27/02
31	Edge Trimming (Actual)	Sat 3/2/02	Mon 3/4/02
32	Secondary Bonding	Thu 2/28/02	Thu 2/28/02
33	Secondary Bonding (Actual)	Tue 3/5/02	Thu 3/7/02
34	**Electronics**	**Sat 3/2/02**	**Sun 3/10/02**
35	Integrate servos, reciever, batteries, linkages	Sat 3/2/02	Wed 3/6/02
36	Integrate electronics (Actual)	Fri 3/8/02	Sun 3/10/02

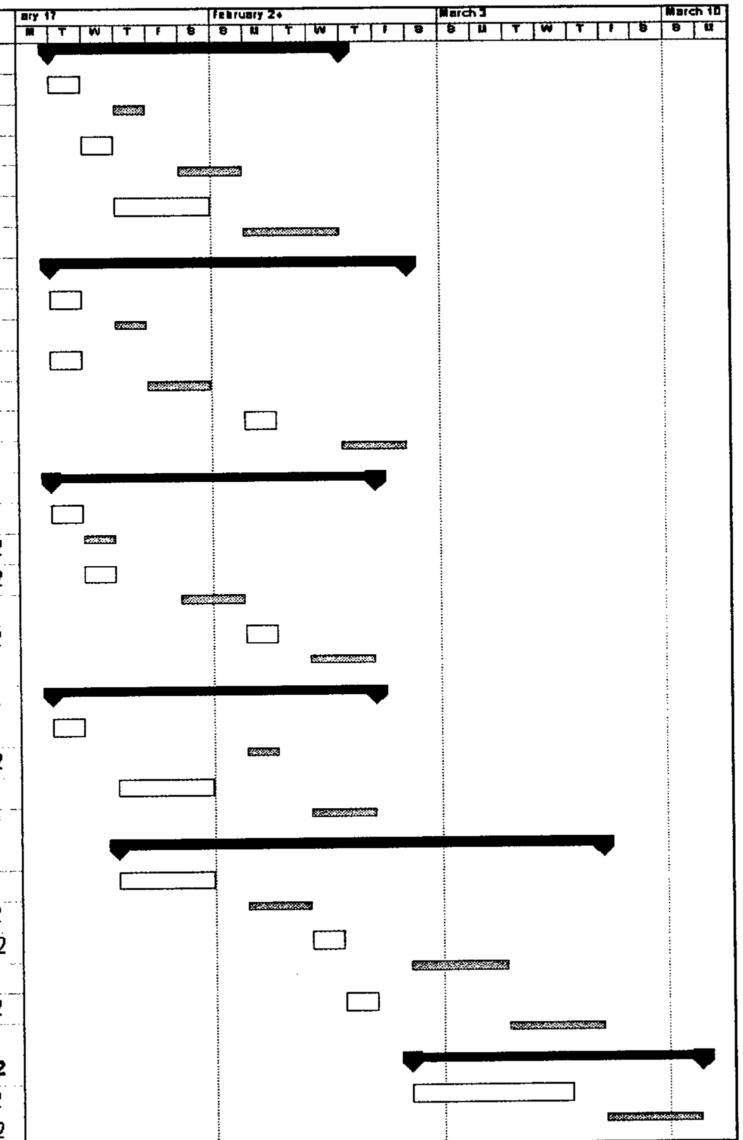


Figure 6-1: Construction Milestone Chart

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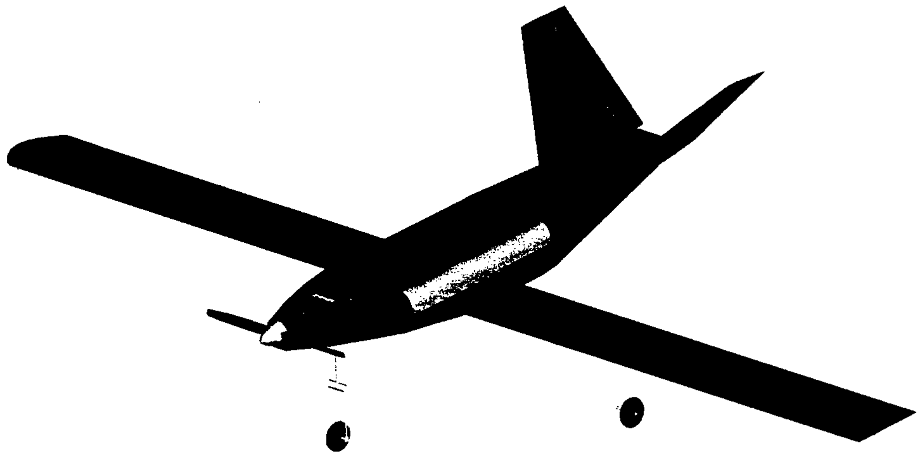

Oklahoma State University
Orange Team

2002 Cessna/ONR
Design Build Fly Competition
Design Report

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1.0 Executive Summary

This report outlines the approach taken by the OSU Orange Team to design and construct an aircraft that best meets the goals set forth in the AIAA 2001/2002 Design/Build/Fly (DBF) Competition. The goal of the competition is to design a propeller driven, unmanned aircraft that performs at the optimum balance between payload capacity, propulsive efficiency, and rated aircraft cost around a closed course of 6 laps. The maximum payload allowed is twenty-four softballs. The total score awarded to the design is a function of the report score, flight score, and Rated Aircraft Cost.

1.1 Conceptual Design

The first step of the conceptual design was to analyze the competition requirements in terms of the three technical groups: aerodynamics, propulsion, and structures. Each group performed sensitivity studies in their respective areas, scrutinizing every aspect of the competition to determine design requirements and to identify the major contributors to overall scoring potential. Then each group generated design concepts in their respective technical areas to meet the design requirements. The next step was to perform a design optimization analysis. Results from the analysis were used to create figures of merit to compare various concepts and evaluate different approaches to achieve the highest possible score. After several iterations, attention was focused upon the optimal region as described by the optimization analysis. Concepts were narrowed down to the most promising ideas, which exhibited the greatest scoring potential and took into consideration all aspects of the competition. Each group refined their respective concepts into their final selections of aircraft configurations, construction methods, materials, and propulsive schemes.

1.1.1 Conceptual Design Alternatives

The aerodynamics group divided the aircraft configuration into modular sections. Different combinations of five fuselage shapes, five wing configurations, and six tail configurations were evaluated for advantages and disadvantages. Different structural alternatives were concurrently evaluated for the components that made up the different configurations. Ease of construction, the ability to produce a lightweight structure, and Rated Aircraft Cost were used to compare alternatives structurally. Three propulsion system alternatives were generated from combinations of different battery configurations, number of engines, and propeller configurations. Propulsive schemes were evaluated in combination with aerodynamic and structural configurations to weigh into the overall aircraft configuration in terms of scoring potential

1.1.2 Conceptual Design Tools

Several tools were employed to evaluate scoring potential of each configuration. Scoring potential was judged as a combination of Rated Aircraft Cost (RAC) and flight performance. Score sensitivities were determined using an optimization code developed from modeling all aspects of the mission. The models represented the weight and drag of the airplane, aerodynamic and propulsive effects of various

components, and a Rated Aircraft Cost (RAC) model. Each model effected the time and RAC and in turn adjusted the overall score of a particular configuration. The performance program adjusted six parameters using random variables and found local maximums in the performance of the plane. Weighted decision matrices were constructed using the sensitivities studies produced from these models. Finally configurations were evaluated in the weighted decision matrices for scoring potential.

1.1.3 Conceptual Results

At the conclusion of the conceptual design phase, the design with the highest scoring potential was selection from all alternatives. The aircraft was a low wing monoplane with a V-tail. The fuselage was composite sandwich, monocoque construction. Both the wing and tail sections were constructed of foam core with composite skin. The optimum propulsive scheme was determined to be a single engine tractor configuration with batteries connected in series.

1.2 Preliminary Design

With the basic configuration of the aircraft determined different approaches to aircraft and control surface sizing, structural design, and propulsion alternatives were analyzed. The aerodynamics group focused on the component and control surface sizing and static stability of the aircraft. The structures group performed stress analysis and selected the materials for the primary components. The propulsion group determined the combination of motor and batteries and propellers to be tested.

1.2.1 Preliminary Design Alternatives

The preliminary design phase consisted of evaluating component types, sizes, and quantities against scoring potential. The aerodynamic studies consisted of wing, tail and fuselage sizing combinations and affects on the overall score. Once preliminary sizes were decided, structural analysis was performed on primary components. Material selection, and construction methods for primary components were then finalized. Finally, the propulsion group evaluated different motors, propellers, and battery configurations to narrow down the propulsive system components.

1.2.2 Preliminary Design Tools

Preliminary modeling of the aircraft was initiated by constructing a two-dimensional station chart using yarn and construction paper mounted on a pegboard. The model allowed the entire team to develop a feel for initial component sizing and placement early in the design. The next step was developing a three-dimension model using CAD software to provide more detailed information for structural analysis. Finally the structural information was inserted into spreadsheets developed to perform stress analysis on primary components. The aerodynamics group used information from the developing structural model for the development of computer code to analyze the requirements and performance of each aerodynamic surface. Information from both the structural and aerodynamic models was used to set design

requirements for the propulsion group. The requirements were feed into a computer code designed to predict the performance of different combinations of motors, batteries, and propellers

1.2.3 Preliminary Results

At the conclusion of the preliminary design phase, the type, sizes, and construction methods of the primary aircraft components were finalized. The aerodynamics group determined the static stability and provided dimensions and tolerances on wing, tail, fuselage, and control surfaces to the structures group, and gave a propulsion mission profile to the propulsion group. The propulsion group used the data to narrow down component alternatives to the most promising choices for further testing. The structures group completed stress analysis on primary structures using mission loads. The final dimensions and makeup of each primary structural component were available for the detail design phase.

1.3 Detail Design

The detail design was generated from components sized during the preliminary design phase. The aerodynamics groups performed dynamic stability analysis and determined handling characteristics and control system requirements. The propulsion group finalized the combination of motor, propeller, and batteries to make up the propulsion system. The structures group completed a three dimensional model and detailed drawings of the surfaces, structures, and subsystems that comprised the final design.

1.3.1 Detail Design Alternatives

Detail design alternatives in each group focused on secondary components and process used in the construction and operation of the aircraft. The system of using figures of merit was once again implemented to analyze the impact of each decision on score. Structural alternatives such as the placement of interior components, system integration methods, wire routing, and hatch design were finalized. Different methods and hardware alternatives for mating the aircraft components were considered. Manufacturing processes were examined for each component. The propulsion group tested the performance of different propellers in combination with different battery and motor types. The propulsion group also evaluated several different types of fuse holders and methods for charging batteries. The aerodynamics group determined control system alternatives in the form of servo requirements and control characteristics.

1.3.2 Detail Design Tools

The aerodynamics group used a flight dynamics code along with an airfoil performance program to determine the dynamic stability of the aircraft, control characteristics, and servo requirements. The stability code was also adapted to generate the stability derivatives. The structures group further developed the CAD model to organize the component design and to predict the weight and center of gravity of the aircraft. A scheduling program was implemented to develop a manufacturing process schedule. Structural analysis on secondary structures and component mating hardware was performed

using information in the spreadsheets and weight models. The propulsion group employed a dynamometer to compare actual system performance to analytical predictions.

1.3.3 Detail Design Result

At the conclusion of the detail design phase all aircraft dimensions, system components, and materials had been selected. The aircraft performance capabilities and control characteristics had been determined. Manufacturing processes had been developed. Finally detailed production drawings were prepared.

1.4 Manufacturing and Prototype Testing

Information gained during the construction and manufacturing of the prototype aircraft was useful to improve the design for the final aircraft. Several changes were made to the final aircraft design. During the construction of the prototype, wing difficulties were encountered when trying to produce the leading edge of the airfoil shape. To correct the problem adjustments were made to the female cradles used in the process to help keep the cross section shape. Difficulties with the construction of the fuselage female mold were also encountered. Several different methods and compounds were tested to find a process that could create the desired fuselage finish. During the flight-testing phase of the prototype difficulties were encountered with roll stability and problems were identified with aircraft visibility. The problems in conjunction with gusty wind conditions led to a crash, from which the problems were identified. To correct the problem, dihedral was added to the wing and reflective tape was applied to the top of the fuselage. After the crash, the modular design of the aircraft was discovered to have assisted in the preservation of the major aircraft components, which made possible continued flight-testing three days later.

2.0 Management Summary

2.1 Team Architecture and Personnel Assignment Areas

The OSU Orange Team was divided into three technical groups: aerodynamics, propulsion, and structures. Each technical group was comprised of a lead engineer and component specialists; each lead engineer answered to the chief engineer. Figure 1 outlines the architectural structure of the team. The chief engineer was responsible for overall team direction and performance as well as ensuring that each group had the tools and information necessary to complete assigned tasks. The lead engineer of a group was responsible for coordinating the efforts of the component specialists to meet the overall goals of that technical group, as set forth by the chief engineer. Together, the chief and lead engineers assured that the design process followed a logical progression from day to day and ensured that every aspect was properly considered.

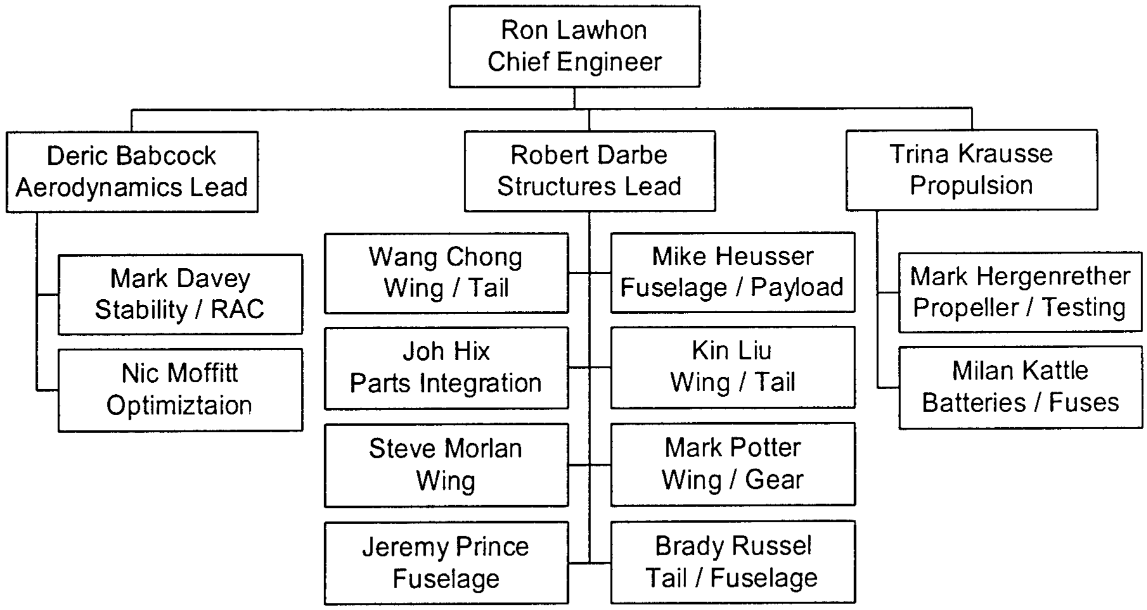


Figure 1: OSU Orange Team Architecture Chart with Design Personnel and Assignment Areas.

2.2 Responsibilities of Each Technical Group

Ultimately the configuration of the aircraft was a combination of the major components submitted by each technical group. The nature of this collaboration was such that every decision made by one group affected the constraints placed on the conceptual designs of the other groups. Each group agreed upon a single concept before further progress was made. The responsibilities of each group were divided among the members of that group. Individual contributions are displayed in Figure 1. Other team members such as the pilot, graduate students, and underclassmen were not assigned to a specific group.

2.2.1 Aerodynamics / Stability and Control Group

The first task for the aerodynamics group was to oversee the mathematical modeling process and incorporate the competition regulations. Using the resulting data, the group generated rough estimates of performance parameters for each group as a base for conceptual studies. The group then progressed to analyzing different aircraft configurations by scoring potential. Figures of merit were defined to further rate the scoring potential of each concept. The end result of the aerodynamic conceptual study was the final plane configuration. The aerodynamics group continued to develop the mathematical model during the preliminary design phase. The model was refined in areas to better represent the conceptual configuration. The results of the mathematical model were focused on the general aerodynamic sizes of the wing, tail, and fuselage. During the preliminary design phase, the aerodynamics group focused on the stability and control of the aircraft. The tail and control sizing resulted from the preliminary phase, along with the servo sizes required to create control movement.

2.2.2 Propulsion Group

The propulsion group was responsible for the design and evaluation of different motor, propeller, and battery combinations. The concepts were judged on the effective use of available power, ability to produce the desired thrust, and effect on Rated Aircraft Cost. During the conceptual phase, the propulsion group decided how many motor and propellers would be required to produce the thrusts calculated by the mathematical models. The group also decided whether a pusher or tractor configuration would be implemented. The propulsion group developed analytical and experimental means by which the wide selection of motors and propellers was narrowed to a few combinations to be prototype tested. The group continued analysis of the components during the detail design phase by checking the validity of preliminary and historical data. Prototype analysis allowed the group to further refine the combination of propulsion components to best met the mission requirements and help minimize the Rated Aircraft Cost.

2.2.3 Structures Group

The structures group was responsible for the structural design and integration of components. Considerations ranged from payload handling methods and structural analysis to material selection and manufacturing processes. During the conceptual phase, the structures group examined several different methods of building the general structures required by all aircraft. The group examined all construction materials available and considered the figures of merit of each. Further brainstorming developed component buildups for secondary structures. During the preliminary design phase, the structures group used loading parameters given by the aerodynamics and propulsion groups to decide on the quantity and size of primary structural components in the wing, fuselage, and tail. Landing gear configurations and loading were also considered so that the gear could be completely designed. During the detail design phase, the landing gear and primary structures were tested for compliance to preliminary parameters. Further analysis resulted in the sizing of the secondary structures. The structures group also considered all tooling and construction methods to be used during the manufacturing phase. Final drawings and implementation of the manufacturing process were the responsibility of the structures group.

2.3 Scheduling, Document, and Configuration Control

To meet the competition deadline, the design process had to progress at a rapid pace. During the course of the design several milestones were identified and implemented as deadlines for different phases of the process. The design process was divided into the conceptual design, preliminary design, detail design, and manufacturing phases. Dates were set for the completion of the ingredients for each of these phases. The design report was developed concurrently with the design process and marked an important milestone that had to be accomplished. Report sections were due along with respective design phase completion. Figure 2 is a milestone chart developed for the design process.

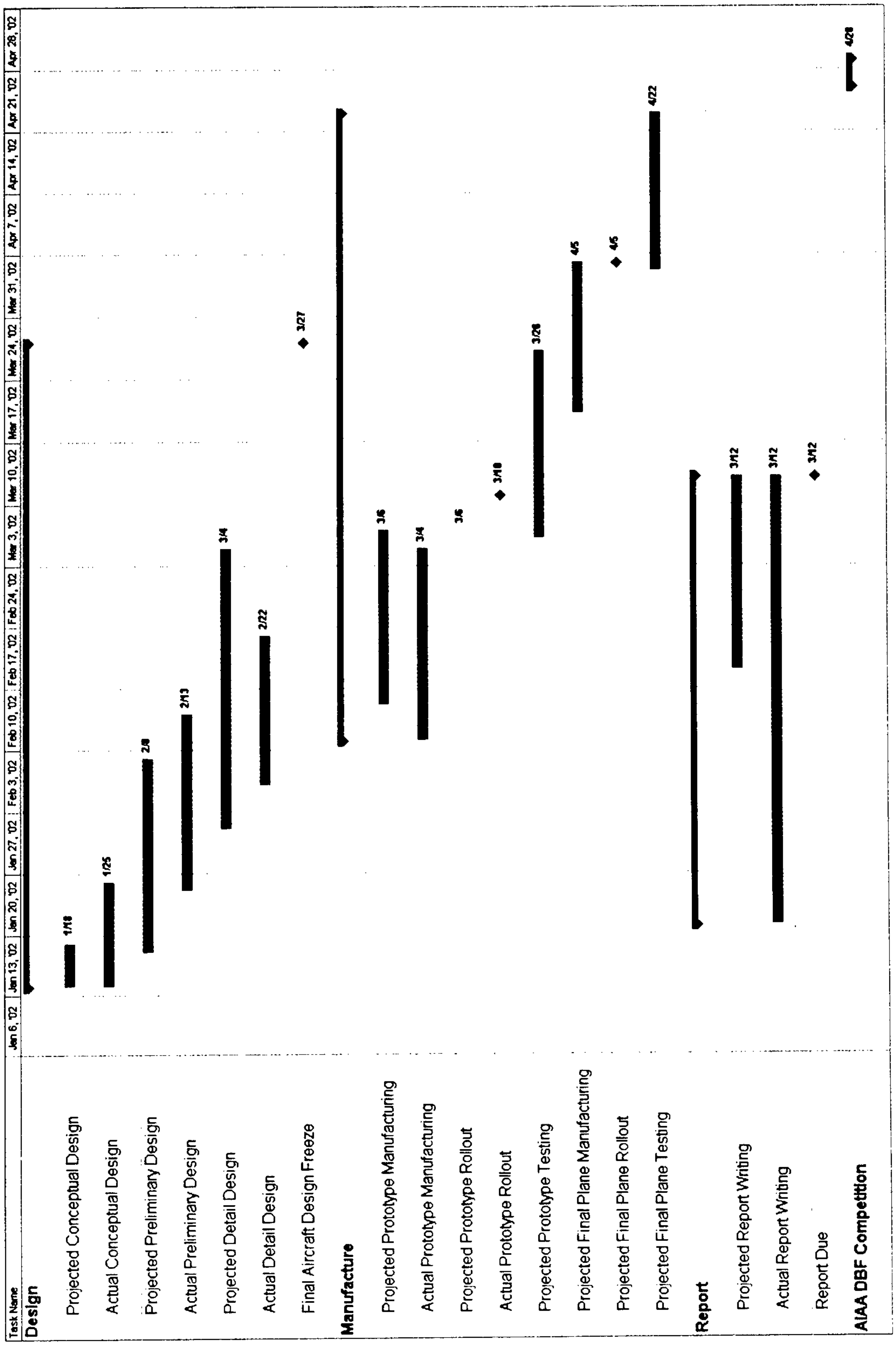


Figure 2: OSU Orange Team Milestone Chart. Red lines represent projected dates; green lines represent actual dates. Black diamonds represent beginning and completion milestones.

Configuration control involved the processing of new information into the current state of the design. Continued updating of the design was important in order to insure that the design iterations were not performed on outdated data. In order to implement the system, control files were created where the latest information was made available. The control files evolved from hanging folders, at the beginning of the design process, to computer files and finally to CAD drawing packages.

3.0 Conceptual Design

The first step in the concept selection process was to perform sensitivity studies on the competition in the areas of aerodynamics, propulsion, and structures. Next, ideas for accomplishing all aspects of the competition were generated. After generating ideas, each concept was discussed to determine the benefits and penalties of implementation. Ideas were combined and evolved until a final concept was reached. The final configuration chosen was best suited for the contest rules and mission, allowing for the most scoring potential.

3.1 Design Parameters

A performance program was written to investigate the design parameters of the plane over the entire mission. The program was used to perform sensitivity analysis on the design parameters. Once the sensitivity to each design parameter was known, an overall strategy was developed and implemented during the three design phases.

3.1.1 Primary Design Parameters

The performance program considered all phases of competition flight and Rated Aircraft Cost. Six primary parameters were chosen as major contributing factors in the development of the plane: weight of batteries, weight of balls, power used in cruise, power used in takeoff, planform area of the wing, and span of the wing.

The six factors were used to maximize the scoring potential of the design. The program found that the highest scoring plane configurations carried the limit of 24 softballs. Takeoff distance was limited to two hundred feet. The full 200 feet was also used by every high scoring plane configuration. Another limiting factor was the number of batteries. Each high scoring configuration used 99% to 100% of the energy capacity of the batteries. These tendencies showed the plane needed to fly efficiently in cruise, while still taking off under the limit with a payload of 24 balls. Figure 3 summarizes the sensitivity analysis results.

The power and wing parameters established areas in which optimal performance occurs; battery and ball weight tended in the largest number of balls with the fewest batteries that could be carried. Takeoff power was found to be between 850 and 1000 watts, while cruise power was between 600 and 700 watts. The wing planform ranged from 5 and 7 square feet with a span of 7 to 9 feet. The above parameters were given to the propulsion and structures group so that refined analysis could be made during the conceptual design phase.

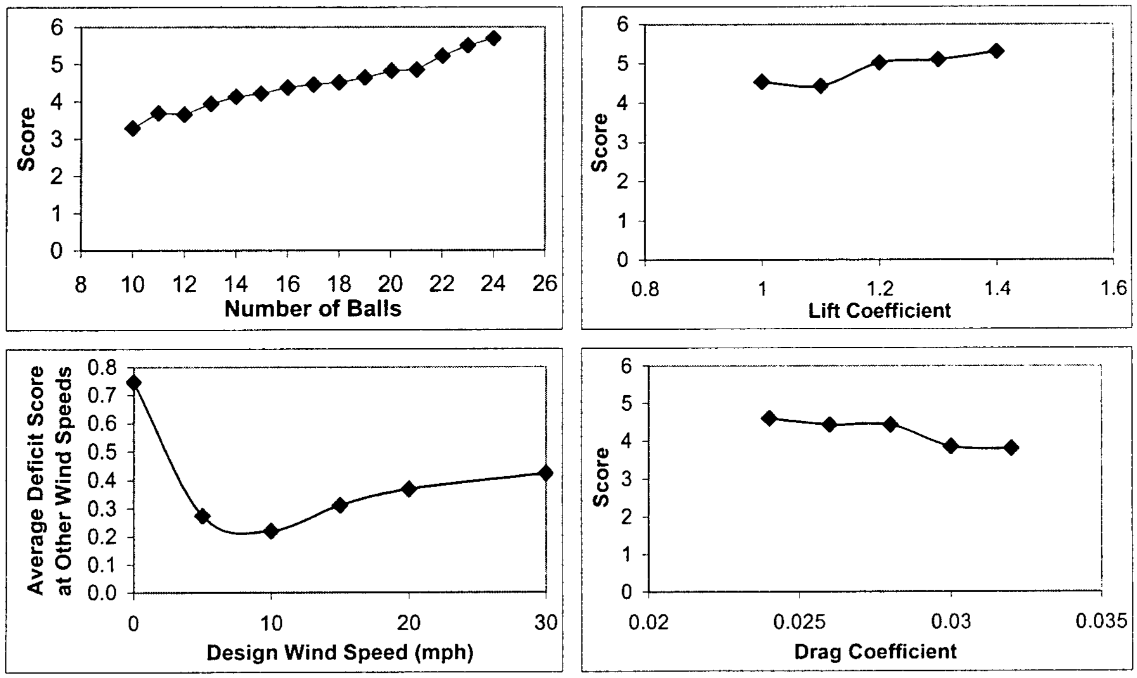


Figure 3: Sensitivity Analysis Graphics. From top left clockwise: Sensitivity to number of balls, lift coefficient sensitivity, drag coefficient sensitivity, and wind sensitivity.

3.1.2 Design Parameter Sensitivity Analysis

Sensitivity checks were made regarding other major parameters. The parameters consisted of the following: weight model of plane, lift and drag coefficients, ball configuration width, fuselage length, time of the ground, and wind sensitivity. The parameters were not flexible within the program and therefore could drastically affect the outcome if used improperly. The sensitive of the seven major parameters helped to optimize the plane and determine the desired values of the parameters within the program.

- Weight Model of Plane: Weight models for the plane were developed from historical data. These models included foam and graphite composite, foam and fiberglass composite, and wood. The weight models, when analyzed through the optimization, showed that the performance of the plane was very sensitive to the overall structural weight. Although, material could not be chosen from this modeling system, the weight model sensitivity did show that construction material selection would be very important to the final design.
- Ball Configuration Width: Ball width was modeled so that the ball configurations could be narrowed down into families of 2, 3, and 4 balls wide. One and five balls wide were discretely eliminated by the rules. Sensitivity to ball configuration proved to have little effect on scoring ability, but the three ball family had a slight advantage and was used in the rest of the analysis.
- Fuselage Length: Fuselage length was investigated for sensitivity by dividing the length into spaces needed to carry the payload and additional length for the nose and tail structures. A basic

drag model was developed to analyze the skin friction on the fuselage and an appropriately sized horizontal tail. The results concluded that the fuselage needed to be as short as possible, even if the result was a huge tail. The limiting factors were the forward and aft rake angles. The total length, taking into consideration the rake angles and three ball family, was analyzed using a combined aerodynamic and structures model.

- <u>Wind Sensitivity</u>: Wind sensitivity was investigated so that the performance could be optimized in preliminary design. To accomplish this goal, the six major parameters were optimized at various wind speeds and compared to other winds. Figure 3 shows an average score deficit for wind speeds from 0 to 30 mph. While each plane performed better in higher winds, softballs had to be dropped to get planes off the ground at lower wind speeds. The plane optimized at 10 mph was found to have the smallest average deficit. Preliminary optimization occurred at this speed.
- <u>Lift and Drag Coefficients</u>: Lift and drag coefficients were also analyzed. Analysis was accomplished by comparing the three optimal scoring capabilities of several lift and drag coefficients. Fewer batteries were required for higher coefficients of lift. The better performance prompted an airfoil study for higher lift coefficients, ranging from 1.2 to 1.4. The number of balls was limited by higher drag coefficients. Drag polars for several airfoils were found to match the lift range and lowest drag possible. Figure 3 also shows the sensitivity of score to lift and drag.
- <u>Time on Ground</u>: The final sensitivity analysis was the ground time. Results showed that ground time was a major factor, and that decreasing ground time would drastically boost the score.

3.1.3 Rated Aircraft Cost Analysis

Another concern was the impact that Rated Aircraft Cost (RAC) had on the design. To evaluate the RAC, a computer program was written to calculate the effects of the RAC parameters. The program was designed to calculate the RAC from a few configuration inputs. The major factor limiting configurations through the RAC was the total battery weight. The battery weight was counted in both the Rated Engine Power and in the Manufactures Empty Weight. Additionally, the Rated Engine Power Multiplier was the largest of the multiplier values. All three factors combined to give the weight of the batteries the greatest influence over the design. Other important factors were the wingspan, empty weight of the aircraft, and body length. To give an idea of the relative importance of these three components, the battery weight had a greater impact on RAC than all three of the other factors combined. Other factors, while not directly having a large effect on the RAC, greatly increased the importance of certain components. For example, the number of engines did not significantly increase the Manufacturing Man-Hours; however, when the power for the extra motors and the Rated Engine Power were considered, the impact of adding an additional motor became large. Using the RAC program, the rating of each configuration could be immediately evaluated.

3.2 Aircraft Configuration

Once the primary design parameters were identified and investigated, the various aircraft component alternatives were evaluated. For each major component, the Figures of Merit (FOM) were outlined and used to judge the various alternatives through a weighted decision matrix. The matrices determined the alternative that best meet the Figures of Merit.

3.2.1 Alternative Fuselage and Tail Configuration

Figures of Merit (FOM) in the analysis of the fuselage and tail configurations included:

- Rated Aircraft Cost: Since RAC strongly affects the overall score of the aircraft, RAC was heavily weighted.
- Take off and landing: From the design parameter studies, the takeoff lift coefficient and takeoff roll were key to producing a high scoring airplane. In the landing phase, ground effects would cause the plane to drift instead of landing quickly.
- Handling Qualities: Handling qualities were important due to the competition location. To fly an efficient circuit, the plane must be able to withstand gusty conditions and be moderately stable.
- Drag Performance: With battery weight counting twice in the RAC, a successful design must make efficient use of the available power and complete the mission quickly.

Once the design parameters and figures of merit for the fuselage and tail were defined, numerous configurations and combinations of configurations were considered. After the concept generation phase, the most promising alternatives were evaluated further. For further analysis, several assumptions were made including: The fuselage could be sized to hold the ball configuration; the propulsive efficiencies were approximately equivalent; and the difference in structural weight was negligible. The parameters could be considered constant during conceptual design because changes in the value of the parameters did not effect the conceptual configuration. Given the assumptions and the figures of merit above, the alternatives were evaluated as follows:

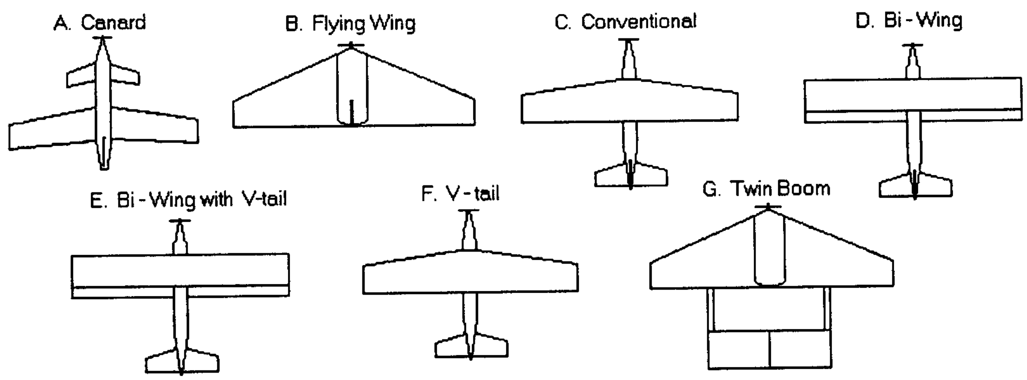
- Canard: The canard configuration had excellent stall characteristics; however, by preventing the main wing from stalling, this design limits the maximum lift at takeoff. The canard configuration had the added benefit of flexible motor setups, both tractor and pusher.
- Flying Wing: The flying wing had an obvious advantage in the RAC by removing the tail and tail control surfaces. The flying wing had some problems, namely extended takeoff roll and poor handling qualities in gusty winds due to a lower wing loading.
- Conventional: The conventional fuselage with a standard tail was set as a standard for comparing the various configurations. The conventional design was relatively straightforward with well-documented performance characteristics.
- Bi-wing: The biplane configuration was able to produce large amounts of lift with a smaller span.

However, the RAC counted the span of each wing and the chord penalty twice.

- Bi-wing, V-tail: A biplane configuration with a V-tail was also considered. The only significant difference would be the RAC advantage of the V-tail over the conventional tail, which resulted from few surfaces.
- Conventional, V-tail: A conventional fuselage with a V- tail had the same performance as the conventional tail alternative, but had the RAC advantage previously mentioned.
- Twin-boom: The twin-boom inverted V-tail had the same RAC benefit as the V-tail and could use either a tractor or pusher propulsion systems. The design would reduce the rigidity of the tail section and allow the booms to twist under loading reducing the handling qualities of the design.

In order to evaluate the Rated Aircraft Cost figure of merit for the various configurations, the RAC of each alternative was calculated. The results of the analysis are listed below with configuration, RAC value, and then the RAC benefits of that configuration:

- Flying Wing - RAC of 8.47 - does not have a horizontal tail surface or elevator servos
- Conventional Fuselage with V-tail - RAC of 8.77 - has passive vertical surface instead of active
- Twin Boom with invert V-tail - RAC of 8.77 - has passive vertical surface instead of active
- Conventional Fuselage with Standard tail - RAC of 8.87 - none, used as standard of comparison
- Canard - RAC of 8.87 - does not have an RAC benefit, equal to standard case
- Bi-wing with V-tail - RAC of 8.88 - has passive vertical surface instead of active, two wing penalty
- Bi-wing with conventional tail - RAC of 8.98 - no RAC benefit, two wing penalty



		Configurations						
Figures of Merit	Weighting Factor	A	B	C	D	E	F	G
Rated Aircraft Cost	0.4	0	1	0	-1	-0.66	0.66	0.66
Take Off and Landing	0.25	-1	-1	0	0	0	0	0
Handling Qualities	0.2	0	-1	0	0	0	0	-1
Drag	0.15	-1	1	0	-1	-1	0	0
Score	-	-0.40	0.10	0.00	-0.55	-0.41	0.26	0.06

Figure 4: Fuselage and Tail Configuration Weighted Decision Matrix

A weighted decision matrix was used to evaluate the alternatives according to the figures of merit to determine the configuration with the highest scoring potential. Weighting factors were assigned with different magnitudes according to the defined mission sensitivities. While RAC is the dominating factor, it is not the only consideration. For instance, while the flying wing has the lowest RAC, the poor performance of the design with regard to the other figures of merit limited its overall scoring potential. The results of the decision matrix can be seen in Figure 4. The highest-ranking fuselage and tail configuration is the conventional fuselage with a V-tail. This arrangement best met the figures of merit because of an improved RAC advantage and average performance qualities.

3.2.2 Alternative Wing Configuration

Figures of merit in the analysis of the wing configurations included:

- Payload Interference: In order to carry the maximum number of balls as the design studies indicated, while still minimizing internal volume and drag, the wing carry through structure should not consume a large portion of space allotted for the payload.
- Loading Payload: From analysis of the flight performance score, a minimum of time should be spent loading and unloading the payload. Therefore, wing locations that reduced the time of or interference with the loading process were awarded a higher magnitude in the decision matrix.
- Construction: The alternative selected must be within the team's construction abilities or the wing cannot be manufactured. The relative time and effort involved were estimated and compared.
- Landing Gear Interface: The gear interface was mainly a structural and construction issue. Low mounted wings provided a broader base for attaching the gear.
- Stability: The stability contributions of the wing location were estimated from historical data. High mounted wings typically provide roll stability, while low mounted wings can produce instability.

The three basic alternatives for wing location were considered: low, middle, and high wing attachment. For analysis the following assumptions were made: the taper ratio was one for better stall characteristics and ease of construction; the sweep angle was zero for ease of construction, low Reynolds number effect, and more efficient lift production; and the wing was sized to produce the necessary lift efficiently and proper wing loading for gust stability.

- High Wing: The high wing did not significantly interfere with payload volume but decreased the symmetric loading of the payload unless multiple hatches were built. The high wing produced roll stability without adding dihedral into the wings.
- Mid Wing: The mid-wing design would prove to be difficult to construct. Blending the body and wing was easier with the mid-wing design than the others considered, but the wing carry through structure would cause much more interference with the payload.
- Low Wing: The low wing alternative has less roll stability, which might require dihedral. The main advantages of the low wing were the light structural requirements and the minimal interference of

the wing with the payload configuration and handling.

Figures of Merit	Weighting Factor	High	Mid	Low
Stability	0.05	1	0	-1
Payloading	0.30	-1	0	1
Construction	0.20	0	-1	0
Payload Interference	0.30	1	-1	1
Gear Interface	0.15	0	0	1
Score	-	0.05	-0.5	0.7

Table 1: Wing Configuration Weighted Decision Matrix

The weighting factors of the wing decision matrix in Table 1 shows the relative importance of each figure of merit in the wing selection. While the payload and construction played a large part in the decision matrix, the stability and landing gear issues were secondary effects. Since the low mounted wing better addressed more of the important figures of merit than the other alternatives, the low wing rated the highest and was selected as the wing configuration for the aircraft.

3.3 Structural Configuration

3.3.1 Figures of Merit

A crucial area of the structural conceptual design was the development of figures of merit to represent mission objectives for each component to perform. The mission objectives chosen were designed to represent crucial areas within the competition that were affected by different structural components. The figures of merit which were developed for the conceptual structural design were:

- Scoring Potential: Each aircraft component had a direct effect on the ability the design to achieve a high score. Since the size, shape, and capability of each component can affect both aircraft performance and Rated Aircraft Cost, the design of each component is important to perform the mission in the most efficient way in order to maximize scoring potential this figure of merit is the overall result of all figures of merit analyzed.
- Strength and Weight: The weight of each component influences both Rated Aircraft Cost and aircraft performance in every phase of flight. Therefore the proper balance between the strength of a component and the weight of the component was necessary to determine.
- Formability: The ability to produce the desired component shape with the least amount of man-hours and components has an affect on the quality of the component, the overall weight of the structure, and the Rated Aircraft Cost
- Ease of construction: Certain construction techniques required more time and skill to produce complex shapes such as high camber airfoils and streamlined bodies. The ability to produce such shapes directly affected the capabilities of the aircraft in all phases of flight
- Repairability: During the flight-testing phase and the competition, minor to moderate damage was expected. On site repair of the damage was desired to reduce amount of time.

- Cost: While a score efficient structure was desired, the cost of certain construction methods and materials have a drastic affect on the validity of some concepts.
- Durability: During flight operations, the aircraft was expected to experience sharp winds gusts and hard landings. In order to be competitive, the aircraft had to withstand the conditions and operations of the competition through many cycles.

While the figures of merit were valid for all structures, several specialized figures of merit were developed for individual components. These figures included:

- Ground Tracking: The landing gear must be capable of precise aircraft maneuvering while the aircraft is on the ground. The mission advantage was to provide quick turnaround times between different phases of the mission sortie.
- Landing Performance: Also contributing to the aircraft score was time spent in the landing phase of the sortie. The ability for the landing gear to both endure hard landings and damp out the landing force with out springing the aircraft back into the air was important to achieving a time efficient landing maneuver.
- Chalked Fuselage Attitude: The attitude of the aircraft while static on the ground has the ability to facilitate or hinder payload-handling operations, which has a direct affect on ground time and thereby overall score.
- Braking: During the landing phase a quick, precise braking action reduced the time spent on the ground drastically. Importance was placed on using the braking power in the most efficient way.

3.3.2 Assumptions Made and Design Parameters Investigated

During the conceptual design phase assumptions were made to allow an unbiased look at different alternatives. For the four primary components analyzed, loading conditions were assumed based on simple models and historical data from previous competitions. Simple weight models were produced from available material samples. Previous experience with different materials and simple analysis validated strength estimates for conceptual studies.

During the conceptual design, parameters that had the most influence on score were identified for each primary component. The parameters influenced not only the structural design but also the conceptual studies of the aerodynamics and propulsion groups. The design parameters investigated for each structural component included scoring potential, efficient use of structural weight, strength, dimensional alternatives of the component, different material and construction alternatives, and effects on Rated Aircraft Cost.

3.3.3 Alternative Wing Structures

The figures of merit used to investigate wing alternatives were strength to weight ratio, formability, ease of construction, durability, repairability, and cost. Additional constraints were imposed on the wing design as a result of the functions of the wing: to generate lift and transfer that lifting force to the rest of the aircraft. To create lift, the wing had to be capable of holding an airfoil shape. For the high aspect ratio indicated by the mathematical model, the wing was required to be rigid enough to prevent a decrease in the angle of attack at the wing tips due to the aerodynamic loads. Due to initial estimates, the wing design was expected have an area of approximately 6 feet square and to experience a 3.4 g-load during a turn at maximum gross weight. Both the functions of the wing must be met on a weight budget for the wing to be efficient. Material alternatives and construction methods were considered. The three types were conventional buildup of lightweight wood skeleton and monocote skin, conventional buildup of carbon fiber skeleton and skin, and composite skin with foam core. Using the OSU historical database, a weight comparison was performed on the three concepts, seen in Figure 5.

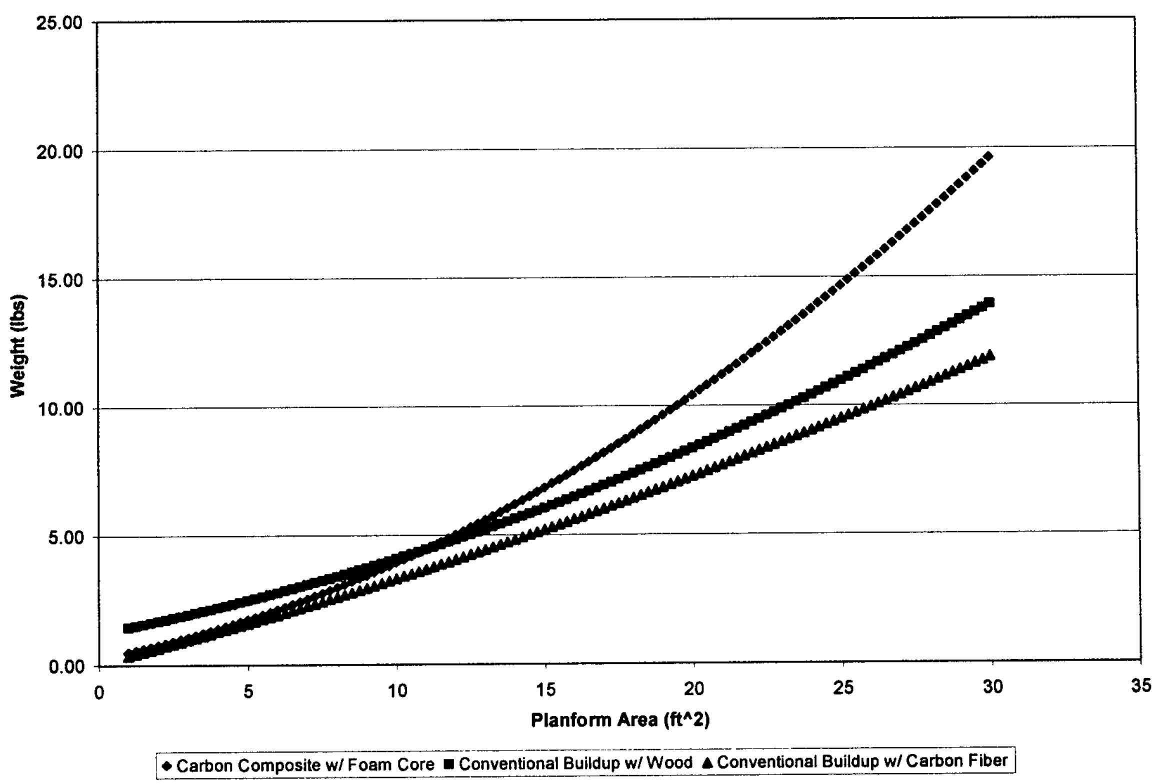


Figure 5: Wing and Tail Structural Material Comparison

Based on the models, the data concluded that composite construction with foam core had the greatest potential to produce the estimated wing area at the lightest weight possible. Composite construction using a foam core was advantageous because the foam allows for accurate construction and holds the shape through out the life of the wing. Table 2 is the weighted decision matrix used to evaluate the

different construction methods. Based on the decision matrix the wing would be of composite skin with foam core construction.

Figures of Merit	Weighting Factor	Conventional Buildup with Wood	Conventional Buildup with Carbon Fiber	Composite Buildup with Foam Core
Strength to Weight Ratio	0.3	-1	1	0
Formability	0.25	-1	0	1
Ease of Construction	0.15	0	0	1
Durability	0.15	0	1	1
Repairability	0.1	-1	0	0
Cost	0.05	1	-1	-1
Score	-	-0.6	0.4	0.5

Table 2: Wing and Tail Structural Decision Matrix

In addition to the structural build up of the wing, conceptual alternatives for internal system components of the wing were generated. These included servos, wire routing, and how the wing will mate and transfer loads to the rest of the fuselage. A final conceptual sketch of the wing structure can be seen in Figure 6.

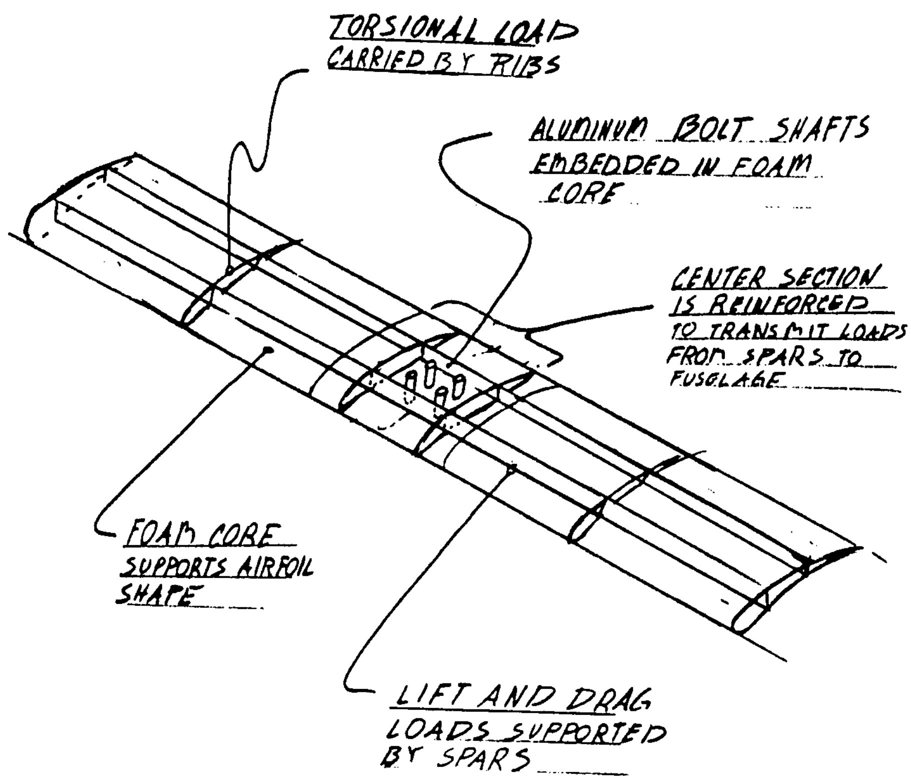


Figure 6: Final Conceptual Sketch of Wing Structure

3.3.4 Alternative Tail Structures

Similar considerations as those used on the wing were made for the tail. The figures of merit used to investigate wing alternatives were strength to weight ratio, formability, ease of construction, durability, reparability, and cost. The three types of tail construction alternatives were conventional build up of lightweight wood skeleton and monocote skin, conventional build up of carbon fiber skeleton and skin, and composite skin with foam core.

The function of the tail was to apply longitudinal and laterally stabilizing moments in order to provide control in aircraft pitch and yaw. The tail unit needed to be lightweight, durable, and able to adjust the incidence angle. Referring to Figure 5, composite construction with foam core showed to have the greatest potential to construct a rigid lightweight tail. Table 2 is the weight decision matrix used to evaluate the different construction methods. (The wing and tail decision matrices were created independently, but reached the same conclusions.) Based on the decision matrix composite buildup with foam core construction will be used to construct the tail. A final conceptual sketch of the tail structure can be seen in Figure 7.

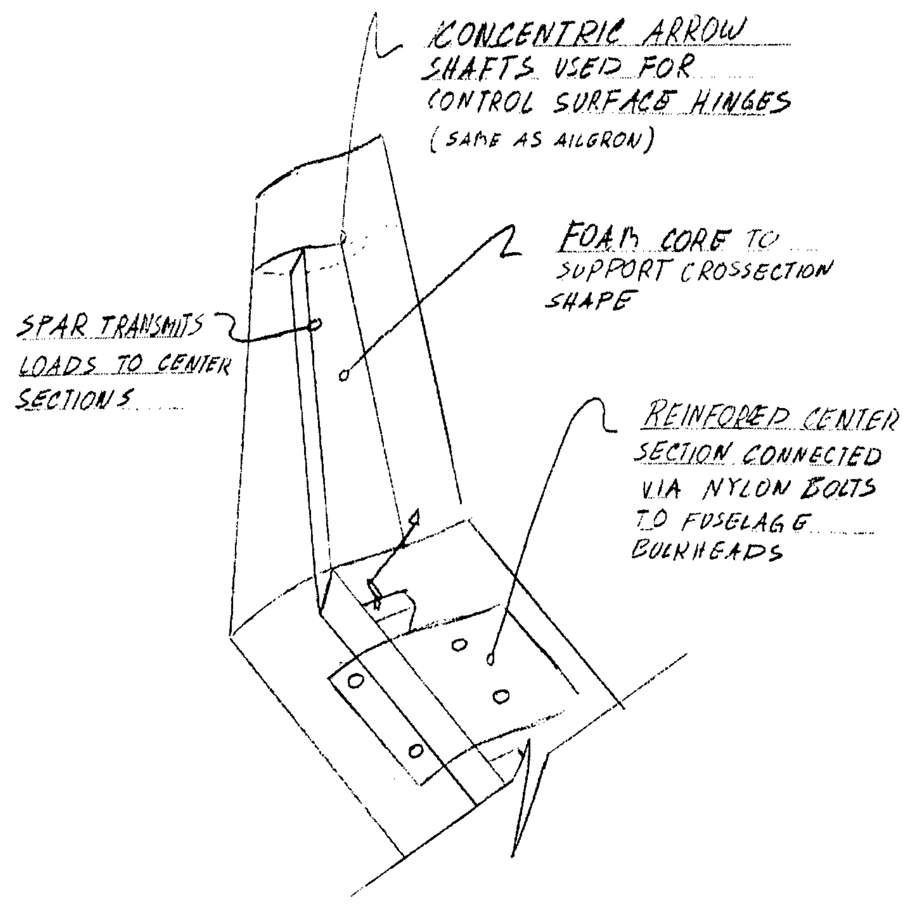


Figure 7: Final Conceptual Sketch of the Tail Structure

3.3.5 Alternative Fuselage Structures

The fuselage concept was developed as a combination of requirements. The figures of merit involved in the fuselage concept were: strength, formability, ease of construction, durability, reparability, and cost. The fuselage must be capable of handling the optimum amount of payload with the lowest drag possible while carrying all radio equipment, batteries, and the propulsion system. The fuselage also serves as the convergence point of all other aircraft components supporting the weight of the entire plane and transferring moments and loads between components. Several different construction techniques were evaluated, including convention buildup with wood, conventional build up with carbon fiber, solid foam construction, and monocoque.

- <u>Conventional buildup with wood</u> consisted of frames and stringer to carry loads and maintain shapes. The framework was covered with a slick monocote that has no load bearing capability. The method featured very lightweight construction but had the potential to lose the advantage if the structure was required to be very rigid. The method suffered from limited usable interior volume capacity due to the framework necessary to maintain a rigid shape. Complex contours were also difficult to achieve with the conventional buildup with wood and monocote method.
- <u>Conventional buildup with carbon fiber</u> was built up of the same frame work as conventional build up with wood however the use of carbon fiber skin allows loads to be transferred throughout the skin thereby eliminating the need for stringers. Frames were also replaced with bulkheads to support the payload deck as well as to stiffen the structure where needed. Complex contours were achieved with relative ease. The conventional buildup with carbon fiber method allowed a lightweight, sleek, and rigid body
- <u>Solid foam construction</u> consisted of a solid fuselage of foam coated with epoxy hollowed out for payload and equipment. The solid foam method afforded a very lightweight and shapeable fuselage. Drawbacks included component mating issues, reparability, and structural deficiencies.
- <u>Monocoque</u> fuselage consisted of a fully load bearing outer shell made of composite material built up on foam molds. Monocoque provided a very lightweight rigid structure that can be formed into almost any shape. Hatch lines were made very smooth and contours can be easily created to blend the fuselage into the different aircraft components. Bulkheads were still necessary to support internal structures but fewer were needed than with conventional buildup methods. Speed of construction as well as repeatability and reparability were all features of monocoque construction.

In order to evaluate the different fuselage construction methods, a decision matrix was formed based on the figures of merit: strength, durability, reparability, formability, ease of construction, and cost.

Table 3 illustrates the comparison of the four methods against the figures of merit.

Figures of Merit	Weighting Factor	Conventional Buildup with Wood	Conventional Buildup with Carbon Fiber	Solid Foam Buildup	Monocoque
Strength to Weight Ratio	0.35	-1	0	-1	0
Formability	0.25	0	1	-1	0
Ease of Construction	0.15	-1	-1	1	1
Durability	0.15	-1	-1	1	1
Repairability	0.1	-1	0	-1	1
Cost	0.05	1	-1	-1	0
Score	-	-0.65	-0.1	-0.4	0.4

Table 3: Fuselage Structural Concept Decision Matrix

Based on the figures of merit, monocoque construction showed the most potential for a lightweight, sleek, fuselage capable of carrying the required loads. The decision to use monocoque limited the fuselage material to carbon fiber and fiberglass. Preliminary design work was required to determine the material that provided the best balance between strength and weight.

3.3.6 Alternative Landing Gear Configurations

The aircraft landing gear system served several purposes and played an important role in payload handling and aircraft performance. The figures of merit used to evaluate different landing gear configurations were: ground tracking, landing performance, chalked fuselage attitude, braking, weight, ease of construction, aerodynamic drag, component mating, and cost. The different approaches to landing gear were first evaluated by overall configuration. Four landing gear configurations were considered: bicycle, tricycle, quad, and tail wheel (or tail dragger). The figures of merit considered for landing gear configuration included ground tracking, landing performance, rotated fuselage attitude, weight, ease of construction, drag, component mating, braking, component mating and cost. The decision matrix shown in Table 4 was used to evaluate the different landing gear configurations. Based on the decision matrix tricycle configuration was chosen as the best balance of all the figures of merit considered.

Figures of Merit	Weighting Factor	Tricycle	Quad	Bicycle	Tail Wheel
Ground Tracking	0.20	1	-1	-1	-1
Landing Performance	0.15	0	1	0	-1
Chaulked Fuselage Attitude	0.15	1	1	-1	-1
Braking	0.15	0	1	1	-1
Weight	0.10	0	-1	1	0
Ease of Construction	0.07	0	-1	-1	0
Drag	0.06	0	-1	1	0
Component Mating	0.06	1	-1	-1	1
Cost	0.06	0	0	0	0
Score	-	0.41	-0.04	-0.17	-0.59

Table 4: Landing Gear Concept Decision Matrix

Once the landing gear configuration was determined, three types of main gear were considered. The function of the main gear was to absorb energy during landing, to provide ground tracking, and to provide proper propeller clearance. The main gear needed to be lightweight, durable, and have low drag. Figures of merit used to evaluate the main gear included landing performance, weight, ease of construction, drag, component mating, and damping effects. Table 5 is the decision matrix used to evaluate the different types of main gear. Based on the decision matrix bow gear and leaf spring scored equally. These two designs were further tested in the preliminary design phase.

Figures of Merit	Weighting Factor	Bow	OLEO Strut	Leaf Spring
Landing Performance	0.25	0	-1	0
Weight	0.25	0	1	0
Component Mating	0.25	0	-1	0
Damping	0.15	0	0	1
Ease of Construction	0.07	1	-1	0
Drag	0.03	0	1	0
Score	-	0.07	-0.29	**0.15**

Table 5: Main Landing Gear Conceptual Decision Matrix

The nose gear of an aircraft was the primary steering mechanism on the ground. The nose gear also provided support for approximately ten percent of the total aircraft weight. Although ground weight loads were low, the loads placed upon the nose gear can be high during the landing phase. The nose gear provided accurate steering while under simultaneous axial and bending loads. Three types of nose gear were considered: Mouse trap (common dual strut nose gear), spring strut, and OLEO strut. The figures of merit considered for the nose gear included ground tracking, weight, ease of construction, cost, and lateral damping. Table 6 is the decision matrix used to evaluate the different types of nose gear. Based on the decision matrix the "mouse trap" style nose gear provided the best performance during the critical takeoff and landing phase when the highest loads are applied and steering was the most crucial.

Figures of Merit	Weighting Factor	Mouse Trap	Spring Strut	OLEO Strut
Ground Tracking	0.30	0	0	0
Longitudinal Damping	0.25	1	-1	-1
Ease of Construction	0.20	1	0	-1
Weight	0.15	1	0	-1
Cost	0.10	1	-1	-1
Score	-	**0.7**	-0.35	-0.7

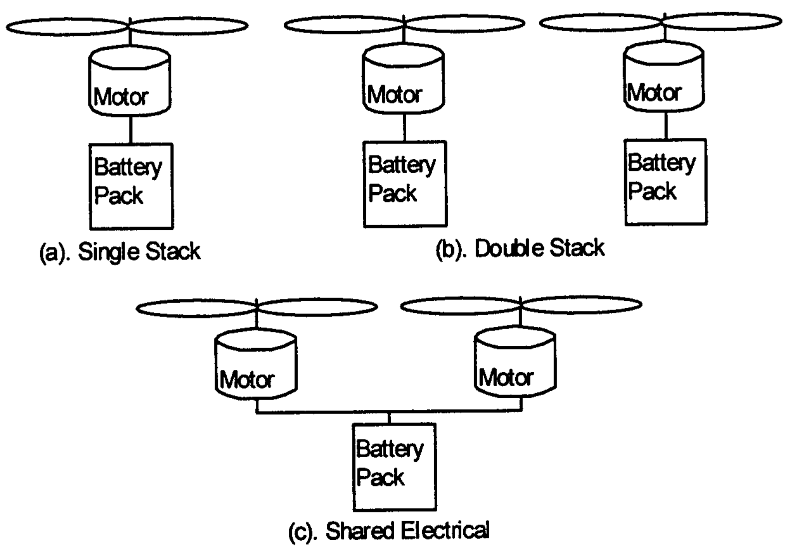
Table 6: Nose Gear Conceptual Decision Matrix

3.4 Alternative Power Plant Configurations

The propulsion system had three main components: batteries, motor, and propeller. The rules limited the variety of components. Three major configuration ideas were considered: single stack, double stack, and

shared electrical. Figures of merit (FOM) were developed to evaluate the three configurations. These included:

- Score Effects: The propulsion system had influence in every variable of the score function.
- Rated Aircraft Cost (RAC): The propulsion system is very influential in the RAC function.
- Power Produced: The power produced by the motor has direct effect on aircraft performance. Power also requires battery performance, which affects the weight of the aircraft and RAC.
- Weight: The battery weight was determined to consume the majority of the overall RAC
- Price: Although the highest performance possible was important, the cost of components limited the level of components used.
- Efficiency: The ability to produce the required performance with the lowest number of batteries was important in both aircraft flight performance and Rated Aircraft Cost.



Figures of Merit	Weighting Factor	Single Stack	Double Stack	Shared Electric
RAC	0.35	1	-1	-1
Power Produced	0.25	-1	1	1
Weight	0.2	1	-1	0
Efficiency	0.15	1	-1	0
Total Number of Fuses	0.05	1	-1	1
Score	-	0.5	-0.5	-0.05

Figure 8: Battery-Motor-Propeller Alternatives and Propulsion Weighted Decision Matrix

Figure 8 shows the propulsion alternatives and decision matrix used to evaluate the different propulsion alternatives. From the decision matrix, the single stack was found to be the best option. The single stack

had the lowest RAC and while meeting the required thrusts and speeds to achieve takeoff and climb. The single stack could provided enough power, thus eliminating the need for two motors. Another draw back to using two motors was the loss in efficiency. In the RAC equation, the battery weight was the most detrimental. Anything that can be done to make the system more efficient was desirable. In order to increase the efficiency of the engine, several cooling concepts were considered. The concepts investigated included air-cooled, water-cooled, and ice-cooled systems. An air-cooled system was chosen as the best concept based on cost, weight, and complexity. The battery packs were to moveable to adjust the center of gravity.

3.5 Analytical Tools

3.5.1 Optimization Program Architecture

The performance program was written to help analyze each phase of the mission individually as a contribution to the overall score. The program contained mathematical models for the structural weight, propulsive efficiency, and aerodynamic characteristics. The models were written into the program so that refinement and expansion could be done quickly and simply. The structural weight and propulsive models were constructed from OSU historical data, as calculated from component buildup. The aerodynamic model was constructed using a simple drag buildup method suggested by Jensen (1990) and the coefficient of lift corrected for aspect ratio of the wing. Characteristics, such as wind speed, takeoff lift coefficient, taper ratio, and sweep angle, were held constant and studied for sensitivity.

After the flight characteristics of the plane were calculated from the four models, the performance characteristics for each phase were calculated:

- The time and energy consumption required to takeoff under the 200-foot limit.
- Time, energy consumption, and power required to climb at a vertical speed of 3 fps.
- Time, distance and g-loading during turning were calculated from weight and stall characteristics.
- Time and distance to slow down from cruise velocity to stopping (without the use of brakes).
- Time in cruise given the selected power settings.

Finally, the times, distances, and energy consumption were summed and compared. The overall values of the three parameters were used to calculate total flight time, flight path distance, and total battery usage. Rated Aircraft Cost was further calculated, and the Flight Score was calculated using both flight time and RAC.

Score = Total Flight Score x Report Score / Rated Aircraft Cost

Further, limitations were placed upon several characteristics so to restrict the possibilities. The span was limited to 12 feet to keep the wing within reasonable structural parameters. Takeoff distance was limited to 200 feet, and battery usage to 100% of available. The limitations kept the performance characteristics within reason during conceptual calculations.

3.5.2 Other Methods Used

During the conceptual design phase, the various groups used weighted decision matrices to decide upon important decisions. Decision matrices were used so that the opinions of group members did not bias the decision, but rather the best configuration was chosen in a more numerical form. The graph of weight and volume effects along with RAC calculator were used to access parts of the factors input into the decision matrices. Strength to weight ratios found in historical data sheets were also used by the structures group to optimize the materials used.

3.6 Final Aircraft Configurations

When the final ranking phase was completed the design that stood out on top was the conventional fuselage with a low wing and V-tail, shown in Figure 9. The configuration had good handling qualities in heavy winds and a reasonable take-off distance. With the low mounted wing, access to the cargo bay from above allowed for fast handling of softballs. The V-tail produces a lower RAC value than a standard tail while still providing adequate control of the plane. The propulsion system was composed of a single propeller tractor setup. The wing and tail were to be constructed of a composite fiber and foam buildup. The fuselage was decided to be a monocoque design, and mousetrap and bow gear were found to be the best gear. The final configuration was the best compromise between all of the ranking factors.

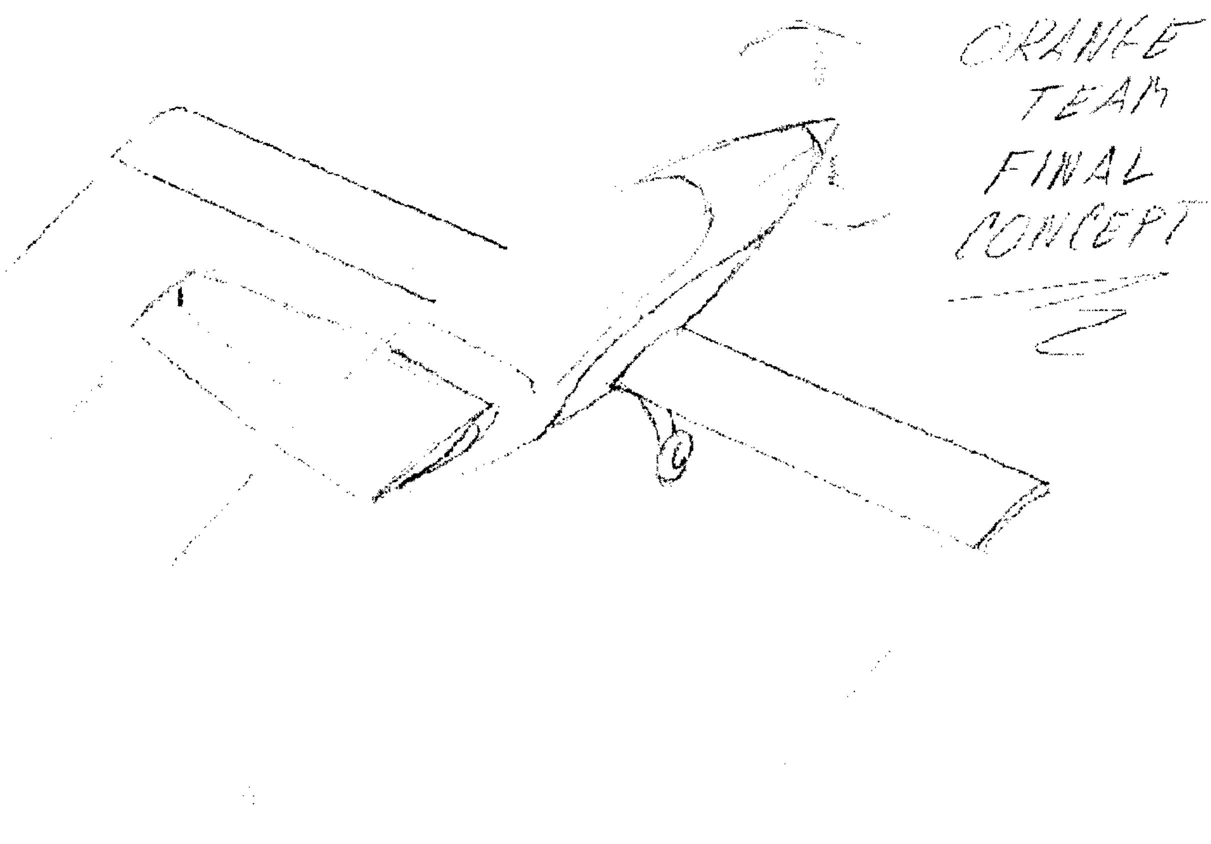


Figure 9: Final Plane Configuration

4.0 Preliminary Design

The preliminary design phase began as soon as the conceptual configuration was determined. The optimization program was refined with preliminary models and used to find parameters to begin the preliminary phase. The parameters were used as a guide to size the aerodynamic and propulsive systems. The wing and fuselage were sized using these parameters; then both were used to size the tail. The dimensions and tolerances for the components were passed on to the structures group so that the trade studies and structural analysis could be performed on primary structures. Finally, the propulsion group narrowed down the possible combinations for testing in the detail design phase.

4.1 Initial Trade Study Results

After the conceptual phase was completed, the weight, drag, propulsive, and Rated Aircraft Cost models in the optimization were updated. The previous models were considered conceptual simulations because the previous versions were developed from the OSU historical database, and the drag model was developed from a basic buildup method. The new performance program was enhanced so that the scoring potential could be maximized. The new program optimized features of the plane and helped develop trade studies for each parameter in relationship to final score. The optimization program was only able to output the local maximums for plane performance; the global maximum was solved by graphical means.

4.1.1 Study of Number of Batteries Required

The number of batteries was explored by considering the effects of battery weight on the aerodynamic performance and propulsive efficiency. Figure 10 shows the relationship of the number of batteries to the optimized score. The graph resembles an umbrella, as does the relationship of other parameters to score. The umbrella relationship shows a definite maximum value. The parameters near the top of the umbrella range from 15 to 20 batteries. The optimal number of batteries was found to be 18 cells with a weight of 2.43 pounds. The five-battery range gave the propulsive group a cushion of two or three batteries in either direction during analysis. Battery selection within the range did not affect the overall plane configuration or scoring potential.

4.1.2 Study of Take Off Battery Power Requirements

The power required from the batteries was not the propulsive throttle position or propulsive power, but the raw power that could be produced by the batteries given the base voltage and a maximum current of 40 amps. The graph of score versus takeoff power showed a similar umbrella trend that was seen in number of batteries. A band of the best values ranged from 625 to 880 Watts with an optimal value of 792 Watts.

Since both the number of batteries and the power required from the batteries during take off showed an umbrella trend, the relationship of takeoff power and batteries was investigated. Figure 10 illustrates the

trend found during this investigation. The trend shows that score increased as the take off power reached the maximum power available in the batteries.

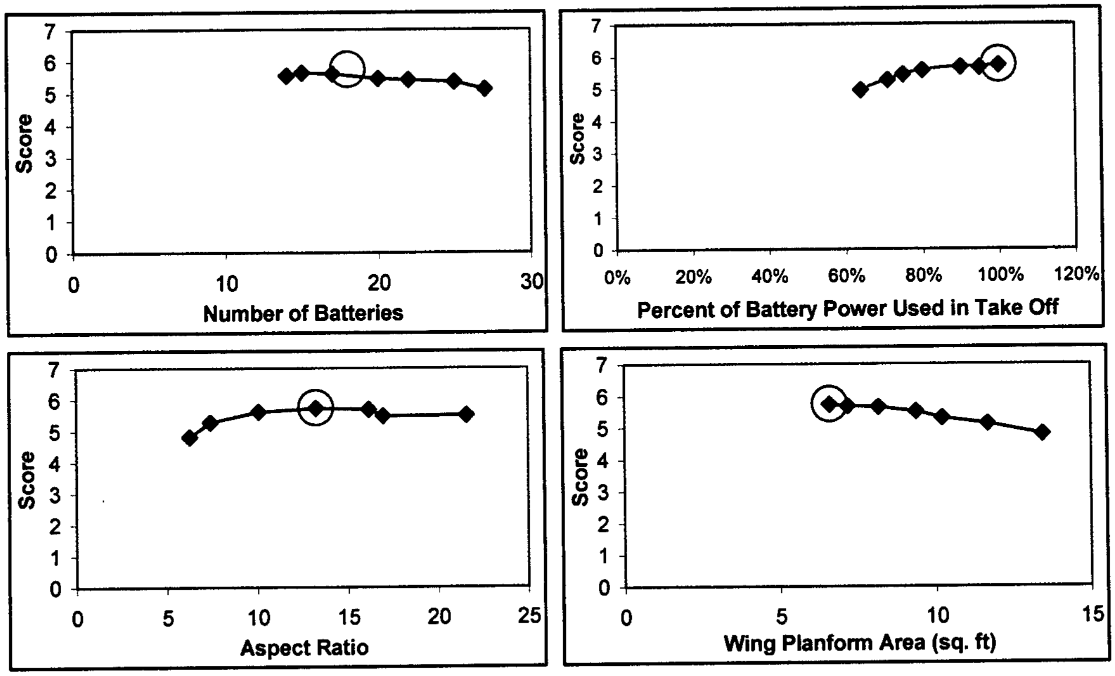


Figure 10: Tradeoff Analysis Results. From top left clockwise: Number of batteries required, takeoff power required as a percent of available power, wing planform area requirement, and aspect ratio requirement. Red circle represents the optimal value found by the program.

4.1.3 Study of Cruise Battery Power Requirements

Similar to takeoff power, the battery power required during cruise was also examined. The score comparison has maximums ranging from 355 to 580 Watts and an optimal value of 494 Watts. Again, the broad range gives a margin of safety for the propulsion group to work within.

4.1.4 Study of Wing Planform Area Requirements

The wing planform area was shown to have a linear relationship to score. The relationship existed on one side of the optimal value and dropped off sharply on the other side. The sharp drop was probably due to the added propulsive requirements to get the plane to takeoff under the limit of 200 feet. Adding battery weight added to the RAC and decreased ability to perform at cruise. Up to the optimal, the linear relationship seemed to represent the plane well because increasing the planform decreases cruise efficiency. All of the above features of the linear trend can be seen in Figure 10. The top 20 values were found to range from 6.25 to 8.2 square feet; the optimal was found to be 6.6 square feet.

4.1.5 Study of Wingspan Requirements

Wingspan was shown to have a rectangular relationship with optimal score. The rectangular scatter illustrated that many different spans can be considered optimal, but given other characteristic

combinations. An optimal range of 8.5 to 10.75 feet was found with an optimal peaking at 9.5 feet. The values gave a good range to oversize or undersize the wing, which would be better explored through prototype experiments.

4.1.6 Study of Aspect Ratio Requirements

Since the optimization tended toward an efficient propulsive system, the optimal scores were graphed versus corresponding aspect ratios to illustrate the aerodynamic trends toward efficient planes. To some extents, aerodynamic efficiency was shown to be the trend. The optimal range of aspect ratios was found to lie at the top of another umbrella shape. The range of these values was from 10 to 17. The optimal configuration was found to have an aspect ratio of 13.25. The trend and optimal value can be seen in Figure 10.

4.1.7 Study of Plane Structural Weight Effects

Plane structural weight was found to be a large factor in the overall performance of the plane during the conceptual phase. The structural weight was again found to have a linear relationship with score. The relationship peaks within a range of 12.25 to 12.85 pounds; optimal being 12.73 pounds. Given the linear trend, any decrease in the overall structural weight of the plane increased the overall performance and score of the plane.

4.1.8 Study of Rated Aircraft Cost Effects

A final study of the effects of Rated Aircraft Cost on performance and score was completed during the preliminary phase. The study found that RAC was considered to be more of a factor than performance, to a point, when optimizing scoring potential. The optimization found a balance that minimized RAC while still keeping the plane within good performance parameters. The optimal plane had a RAC value of 8.91, within a range of 8.50 to 9.25 for the top 20 configurations.

4.2 Aerodynamic Considerations

Using the design parameters determined during the preliminary phase, further analysis of the aircraft was performed to refine the design. The areas examined included: wing area, tail sizing, fuselage lofts and stability and control. The concepts for each aspect were evaluated and then filtered.

4.2.1 Wing Figures of Merit, Assumptions, and Analysis

The figures of merit for the wing analysis included:

- Total Flight Score: Since the score is used to rank the aircraft for the competition, total flight score was set as the primary figure of merit. The wing sizes that produced the greatest scoring potential ranked highest. The flight score included the RAC effects of the configuration.
- Performance: Performance quantifies the effects not considered in the optimization; namely, tip strike, strength, and deflection. The performance merits tended to shorten the wing to ensure

that the wing could realistically achieve the characteristic calculated in the optimization. The wing was modeled in the optimization program to provide estimates of the flight score based on the wing area and span required for the mission profile. On the highest scoring configurations, the chord varied from 7.5 to 9.5 inches, and the wingspan ranged between 9 and 12 feet. The optimization also included other factors involved in the analysis of the wing dimensions, such as the takeoff power and cruise power. The power necessary for takeoff and cruise was related to the wing area through calculations of lift and thrust. Therefore, a study of the motor performance was necessary in the analysis of the wing dimensions. Another aspect of sizing the wing was deciding whether a greater chord length or a longer wingspan best achieved an increase in planform area. The wing span length was penalized heavily in the RAC calculations, while the chord was only a minor factor. Increasing the wing chord produced more wing area, but decreased the aspect ratio. Higher aspect ratios produced drag efficient lift production as predicted by the conceptual assumptions. The optimization program altered the constraints to find the best scoring configuration. The optimized wing was then slightly oversized to allow for both construction and aerodynamic margins of safety. The wingspan and chord were 9.5 feet and 8.625 inches, giving a wing planform of 6.83 sq. ft.

The coefficient of lift necessary for takeoff predicted by the optimization program was 1.2. Using the assumption of 80% of C_{lmax} at takeoff, the wing airfoil required a C_{lmax} around 1.5. A search on the NASG database produced several airfoils meeting the requirements. The Selig-Donovan 7062 and SG6043 were the most promising airfoils. When comparing these two airfoils, the coefficient of drag at both takeoff and cruise were examined. The SD7062 had a lower C_D at cruise, while the SG6043 had a lower C_D at takeoff. Since more time is spent at cruise, the lower C_D at cruise was better. Therefore, the SD7062 was selected as the airfoil for the wing; the lift and drag performance can be seen in Figure 11.

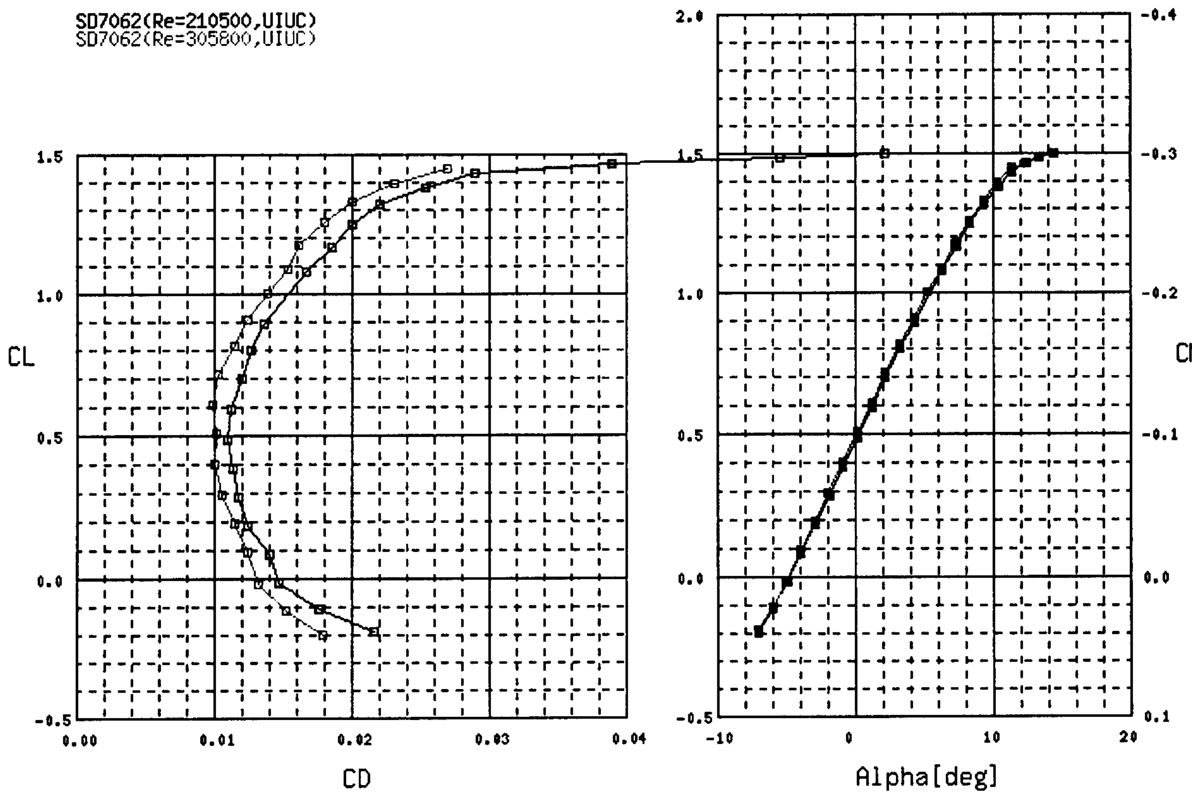


Figure 11: Drag Polar and Lift Curve for SD7062, Wing Airfoil. Cruise Reynolds number of 350,000 in red. Takeoff Reynolds number of 250,000 in black.

4.2.2 Fuselage Figures of Merit, Assumptions, and Analysis

The Figures of Merit for the fuselage analysis included:

- RAC: The main RAC affect of the fuselage was the body length. In order to reduce the RAC, the body length was minimized but still contained all components.
- Drag: The drag on the fuselage was minimized in order to reduce the thrust required ultimately decreasing the battery weight. The factors contribute to a higher score.

Further studies into the effects of the fuselage were performed to obtain better drag models and stability characteristics. During the conceptual design phase, the optimum number of softballs was found to be 24 balls. The payload configuration was determined by calculating drag effects from various fuselage cross-sections. Using the data gathered, the number and configuration of the balls governed the fuselage shape. The fuselage was broken into three sections: the propulsion compartment, cargo bay and tail section. The necessary area for each portion was evaluated independently. Propulsion equipment, such as the motor and speed controller, needed to fit within the confines of the forward compartment. The length of cargo bay was defined by the space necessary to hold the 24 softballs. In conceptual design, the length of the cargo bay and therefore the overall fuselage length were governed by the number and configuration of softballs. The governing factor in the tail section dimensioning was the rake angle for

optimum performance. The propulsion equipment was arranged to fit within a 9 inch tapered nose section. The distance necessary for the cargo bay to hold the payload was 34 inches. A rake angle of 15 degrees was selected to minimize the drag due to flow separation while still considering weight and length. The angle was obtained from historical data. With a 15 degrees rake angle, the tail section is 14 inches in length. The three parts combine to form the total fuselage length of 57 inches. By designing the fuselage around the components, rather than vise versa, the length of the fuselage was minimized.

4.2.3 Tail Figures of Merit, Assumptions, and Analysis

The figures of merit for the tail analysis included:

- Performance: Two major performance factors were considered. The tail had to trim the airplane in cruise and produce the necessary pitching moment at takeoff.
- RAC: If the horizontal tail span exceeded a quarter of the wingspan, the tail was classified as a wing. The RAC effects must be investigated to determine which produced the smallest RAC.

Once the moments at cruise due to the wing and fuselage had been calculated, the moment required by the tail became a function of airfoil and root chord of the tail. The Selig 6063 and the NACA 0009 were the leading possibilities for the tail airfoil. The S6063 was selected for low drag and small lift coefficient at zero angle of attack. The S6063 airfoil required only minor adjustments of the control surfaces to trim at cruise resulting in less drag. Before the final sizing of the tail, the takeoff pitching moment was calculated and compared to the moment produced by the tail. The dimensions were then adjusted so that the tail produced the takeoff pitch moment. The parameters found for the tail can be seen in Table 8 on page 41. Note the area used in the calculations was smaller than the actual area. Some planform area was lost due to the taper of the fuselage. The conservative estimates used in the above analysis were expected to reduce the amount of deflection

Another factor influencing the tail decision included the RAC. If the projected horizontal tail span exceeded a quarter of the wingspan, the tail was classified as a wing by the RAC. Both alternatives were calculated to determine if the tail would score better if classified as a wing or V-tail. Once the tail was classified as a wing, the span and chord penalized the RAC and overall score. Therefore the RAC calculations proved that the tail span needed to stay within the quarter wing span limit. Conceptual design considerations assumed that the V-tail configuration would help to reduce RAC and boost scoring potential. Therefore the RAC calculations proved that the tail span needed to stay within the quarter wing span limit.

4.3 Structural Analysis

4.3.1 Figures of Merit

During the preliminary design phase several figures of merit were used to represent different mission features for structural design.

- Scoring Potential: Every aspect of each component had the potential to influence score through the ability to efficiently perform a task
- Weight: The weight of each component played an important role in both aircraft performance and rated aircraft cost.
- Ease of construction: The methods required to produce complex contours and shapes limited the validity of a design due to the required skill and materials
- Aerodynamic considerations: The design of the fuselage wing, tail, and landing gear influenced lift, drag, and aircraft stability. The effects that different components have on these aerodynamic considerations were examined.
- Allows easy loading and unloading of payload: In order to minimize time spent on the ground during sorties, the structural design needed not interfere with payload handling operations.
- Cost: The cost of certain materials and methods prohibited the validity of designs requiring them.

4.3.2 Design Parameters and Trade Studies Investigated

During the preliminary design phase parameters for primary components were identified for structural analysis. The design parameters investigated included:

- Material selection for the primary components
- Weights for primary components
- The number of spars required for the tail and wing
- The skin thickness of the wing and tail
- Spar shape and lay-up for both the wing and the tail
- Landing gear dimensions and spring rate
- Fuselage skin thickness and composite lay-up

During the structural analysis, different approaches to the design of each parameter were investigated to identify tradeoffs. The trade studies compared structural alternatives in the areas of material selection for each component, strength vs. weight tradeoffs, composite lay-up orientations, and component deformation and aerodynamic tradeoffs. Using information from the studies assisted in the design of each parameter to achieve each components highest scoring potential.

4.3.3 Wing Assumptions and Structural Analysis

While the wing was to be a single component that bolted directly to the fuselage, the analysis model of the wing was a cantilever beam that extended from the edge of the fuselage to the wing tip. For simplicity of the analysis, the lift and drag loads were assumed evenly distributed along the lengths of the wing portion. Since the bending stresses in the wing caused by the lift will be the dominant factors contributing to failure, an evenly distributed load assumption will provide a more conservative analysis. When the distributed load was resolved into a resultant point load, the load was applied further out on the wing compared to a resolved elliptically distribution. Therefore the calculated bending moment will be larger.

The same reasoning was used for justification for the drag load analysis. For buckling analysis the wing skins met the criteria necessary to be modeled as an Euler Column with one fixed and one free end.

A trade study on three spar cross-sections, a solid rectangular, a sandwich rectangular, and a solid C-channel was analyzed to determine which shape would provide the needed structural integrity with the least amount of material and weight. The analysis helped to decide whether one or two spars were necessary to carry and transmit the expected loads. The investigation included calculating the bending, shear, and axially tensile and compressive stresses that the spar would encounter under conditions of a fully loaded aircraft at a 3.4 g-load. At gross weight, the conditions resulted in a vertical wing loading of 0.08 pounds per square inch. The results of the spar analysis showed that one main spar located at the quarter chord of the airfoil would be sufficient to support the expected loads. The sandwich rectangular spar was the lightest spar configuration to carry the loads. The bending and shear stresses of the sandwich spar were significantly lower than the other two spars due to the larger moment of inertia obtained from the sandwich design. The axial stress was larger because the stress was calculated using only the cross sectional area of the outer shell of the sandwich, which was significantly lower than the areas of the other two spars.

The shear stress on the rib was calculated from the aerodynamic moment on the wing. The shear stress showed to be extremely low compared to the material properties of the carbon fiber. Only a two-ply rib was required to carry the load, and the weight of the ribs was almost negligible, at 0.08 ounces per rib. The maximum shear expected in the rib was calculated at 1391 psi.

The outer skin was modeled as a cantilever beam. The bending, shear, and axial tension and compression stresses were calculated using computer generated section properties for both the skin as a shell and the skin with a foam core. The results showed that due to the material properties of the carbon fiber, the wing skin withstood all of the expected loads without any internal structure, with the exception of the torsional stresses. The structural analysis was performed with the assumption that the carbon fiber lay-up was balanced and symmetrical, using a balanced weave prepreg material with no plies oriented in the 45 degree direction. The maximum bending and torsion stresses expected for the wing assembly were calculated to be 2283 psi and 1265 psi respectively. The critical load for skin buckling was estimated at 3693 lb.

For a carbon fiber lay-up of 4 plies on the spar and one ply for the skins, the wing failed in bending at the root. The factor of safety in bending was calculated at 5. The wing assembly structural weight was estimated at 1.9 pounds. In reference to the weight material comparison shown in conceptual design, the wing weight was found to be within 18% of the conceptual estimate.

4.3.4 Tail Assumptions and Structural Analysis

The moments created by the tail for longitudinal stability were a function of both the airfoil used for the cross section and the control surface deflection angle. The loading cases assumed for the structural design of the tail was the loads of a 20-degree ruddervator deflection at a flight speed of 85 fps. The loading case corresponded to an emergency control reaction at our highest cruise speed. Loads were also modeled from the aerodynamic moments and forces created by the airfoil. The stress analysis was performed in a similar manner to the wing analysis. All loads were assumed to be carried from the tail to the fuselage via a single carry through spar of rectangular cross section. The analysis was conservative because the moment of inertia provided by the carbon fiber skins of the tail was not accounted for.

For structural analysis the spar was modeled as a simple beam. The highest stress encountered was located at the root of the tail fin. The maximum bending and shear stresses were calculated to be 19000psi and 360 psi respectively. The deflection at the tip of the tail fin was calculated to be -0.43 inches. The angle of twist caused by the aerodynamic moment was calculated to be 0.086 degrees. The factor of safety was determined to be 4.2 under these loading conditions. The preliminary weight estimate for the tail was 0.98 pounds, 30% difference from conceptual design.

Estimates of the carbon fiber tail skins were sufficient to handle the loads with a reasonable factor of safety. Although the spar added weight, a tail spar was deemed necessary in order to provide a suitable load path into the fuselage. Figure 12 shows a transparency of the tail section to illustrate the spar design. The same flight scenario was used to determine the maximum static load applied to the graphite arrow shaft used for the hinges on the ailerons and the ruddervator. Analysis determined that a single graphite arrow shaft would deflect 0.15 inches. Testing proved that deflection still allowed smooth movement of the control surface.

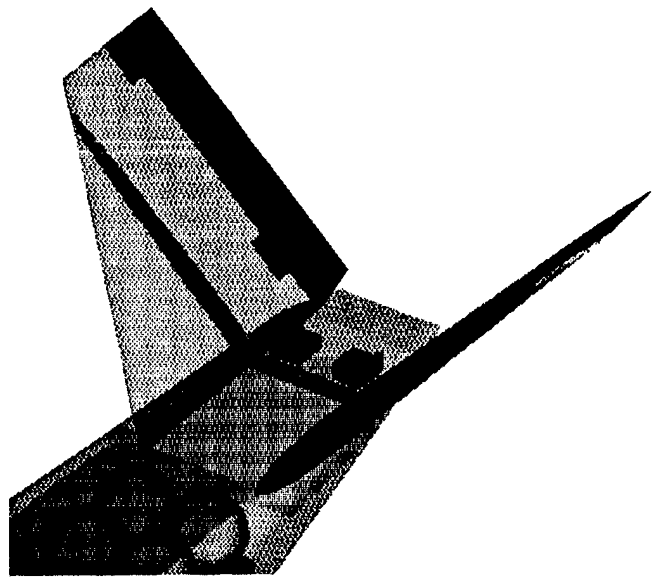


Figure 12: Transparent V-Tail Cut Away from the Rest of the Plane

4.3.5 Fuselage Assumptions and Structural Analysis

In modeling the fuselage some basic assumptions were made: Static analysis could be used when the fuselage of the aircraft was modeled as two cantilevered beams fixed at the center of gravity. The beams were modeled to represent the bottom half of the fuselage because the top half of the fuselage will be a non-load carrying payload access hatch. The center of gravity was fixed such that no rotation or translation was allowed. Dynamic tail forces were not included in the fuselage strength analysis because the aircraft was assumed to be in level flight. When applying loads to the fuselage, the following assumptions were made. All loads where modeled as point loads with a factor of safety of 1.25. The loads were developed as if the aircraft was fully loaded with 24 softballs and 20 batteries and experiencing a 3.4 g loading. Engine torque was also analyzed at maximum power settings. These assumptions were based on information from the preliminary design aerodynamic and propulsive studies and historical data.

The fuselage was broken into two sections in order to determine different inertia values at the relative location of the loads. Using singularity functions, the maximum bending stress was calculated to be 54,000 psi. Classical laminated plate theory with two plies of prepreg carbon fiber orientated at an angle of 45° relative to the axial direction of the fuselage resulted in a margin of safety of 72%.

4.3.6 Main Gear Assumptions and Structural Analysis

The analysis for the solid spring bow gear was modeled as a cantilever beam attached at the center of the fuselage. The general loading cases for the landing gear were developed based on information obtained from Raymer (1999).

The analysis began by establishing a wing tip scrape angle and a maximum gear height. The wing tip scrape angle was specified to be 13°, and the maximum gear height was set at 8 inches. The wing tip scrape angle was determined by a combination of historical data and research performed on aircraft with similar aspect ratios. Based on the above constraints and the length of the wing, the overall gear track width was calculated to be 38.70 inches with a fully extended angle of 28.3 degrees. The gear stroke length was calculated for a number of empty and loaded static loading cases. Tests were performed on two-wheel touchdown empty and loaded, and a one-wheel touchdown empty and loaded. Using the gear loading information, a gear load factor of 3 was assumed for normal two wheel landings. The value was found in Raymer (1999). The gear load factor caused the landing weight to increase by a multiple of 1.5 per wheel in two-wheel landing conditions. The landing weight was doubled for the one wheel landing analysis.

The stresses calculated included vertical bending stress and direct shear stress due to landing. The braking forces included horizontal bending stress and torsional shear stress due to wheel spin-up or braking. A horizontal acceleration of 10 ft/s^2 was used in the calculations. Because the gear was assumed to be made of a heterogeneous material and had a complex three-dimensional loading condition, advanced beam theory was used to determine the maximum stress at four key points in the main gear cross section. The upper and lower corners of the leading and trailing edge of the main gear were examined for failure. Angle of twist was calculated to finalize the spin-up and braking analysis. All calculated stresses were compared to their appropriate strengths in the form of factors of safety being calculated as well. The maximum bending and shear stress was calculated to be 17900 psi and 2000 psi respectively.

When the stress analysis equations were developed, attention turned to the design of the composite lay-up, which carries the various loads. Based on the nature of the torsional loads on spin-up or braking, 45 degree plies were needed in the lay-up. After studying several combinations of composite layers, a lay-up containing equal numbers of 90 degree and 45 degree plies proved to be the best in carrying the loads. Building the main gear entirely out of carbon fiber prepreg without sandwich construction requires 26 plies, weighing approximately 0.92 pounds. The factor of safety with this lay-up was estimated at 4.6

4.4 Propulsion Analysis

The preliminary design phase for the propulsion system involved construction of a computer program for optimizing the major components: batteries, motor, and propeller. The program simulated the complete propulsion system from the power used by the batteries to the power produced by the propeller in flight. Components were allowed to vary in such a way as to match and/or exceed the output values estimated for thrust, current, power, and efficiency for the overall propulsion system. Estimated values were provided from the optimized aerodynamic aspects of the plane.

Four propulsive components were varied in the optimization program: battery cell count, propeller dimensions, motor-size, and gearbox ratio. Battery cells were considered based on the cells that produced the best energy density (amp-hour/weight). The SR 2400 mah was determined to have the best energy density. The dimensions for the propeller were based on the relationship between diameter and pitch. A range of propellers was used to vary the performance characteristics. Six motor series were examined for the program: the Astro Cobalt 25, 40, and 60 and Astro FAI 25, 40 p/n 642, and 40 p/n 643 series. For simplicity of the design, the gearboxes were based on those available for AstroFlight motors. The gear ratios chosen for analysis were 1:1, 1.68:1 and 3.1:1. The pitch/diameter (p/d) ratio of the propeller was first varied. For higher p/d ratios, the speed and efficiency at cruise were better than the lower p/d propellers. Significant increase in the number of batteries was required to produce the desired speeds with lower p/d ratios. The gear ratios, battery cells, and propeller ranges were combined with the six motors to form a variety of configurations. Analysis was performed for the optimum configuration. The Cobalt series motors were compared to the FAI series motors. The analysis of these comparisons is shown below.

4.4.1 Figures of Merit

Figures of merit were developed to evaluate the design parameters. The figures of merit represented the relation between efficient propulsion system and the competition goals and regulations. The figures of merit used were:

- <u>Score</u>: In all aspects of the design, score was the ultimate figure of merit. All aspects of every design decision had the potential to influence the overall scoring potential of the aircraft
- <u>Efficiency</u>: The propulsion system affected score in both flight performance and Rated Aircraft Cost. Efficiency of each part was optimized to offer good performance in both areas
- <u>Weight</u>: The weight of the propulsion system was determined to be a large player in the Rated Aircraft Cost because battery weight was heavily penalized. Therefore reducing the weight of the batteries as well as other propulsion system components can provide a dramatic score improvement
- <u>Current</u>: The current draw on the batteries was a limiting factor set forth by the competition regulations
- <u>K_V Values</u>: The K_V Value of a motor was a ratio that relates RPM to voltage. Selecting a motor with the proper K_V Value allowed thrust to be produced at acceptable current levels
- <u>Energy Density</u>: The energy density of the batteries was crucial to determining which batteries provided the most energy with the lowest increase in battery weight.
- <u>Historical Data</u>: Historical data was invaluable in predicting efficiencies and performance during the trade studies performed on the different propulsion components.

4.4.2 Design Parameters and Trade Studies Investigated

Several design parameters were involved in the design of the propulsion system. Studies on the tradeoffs between the parameters were used to evaluate different combination of batteries, motors, and propellers.

- P/D ratios: By fine-tuning the ratio of propeller pitch to propeller diameter, it was possible to increase the efficiency of the motor and thereby reduce the number of batteries required.
- Propeller Diameters: The propeller diameter has a direct affect on the amount of current required to maintain an RPM setting. Since the propulsion system is limited to 40 amps of current the propeller diameter was a limiting design parameter.
- Number and type of batteries: Overall aircraft efficiency was a crucial parameter on the competition. An efficient propulsion system would find the optimum balance between power produced and batteries required. Balance was crucial to reach the aircrafts maximum scoring potential.
- Size and type of motor: Different motors were optimized for operation in specific ranges. Motor selection was important with the most potential to operate efficiently under the predicted flight constraints

4.4.3 Weighted Decision of Motors

The results of a weighted decision matrix were analyzed for the Cobalt series motors and the FAI motors. For the Cobalt 25 and 60 series motors, the optimal configurations produced thrust for takeoff due to much current or the number of battery cells required to produce the thrust was out of range of interest. Therefore, the motors compared in Table 7 were the Cobalt 40 and the FAI series motors. From the decision matrix, the efficiency for the Cobalt was much lower than the FAI series. Furthermore, the Cobalt series motor was determined to require more battery cells than the FAI motors. Results from the decision matrix show that the FAI motor series score were all fairly close in relationship with each other. To determine which motor was best for the design, other parameters must be considered.

Figure of Merit	Weight	Cobalt 40	FAI 25	FAI 40 / 642	FAI 40 / 643
Number of Batteries	0.3	0.7	0.9	1	1
Current	0.25	1	0.8	1	1
Takeoff Thrust	0.25	1	1	1	1
Cruise Power	0.1	1	1	1	1
Efficiency	0.1	0.5	0.9	0.8	1
Score		0.86	0.91	0.98	1

Table 7: Motor Weighted Decision Matrix

4.4.4 Investigation of Astro FAI Series Motors

To determine which FAI motors would be more desirable, the following analysis was performed on the motors. Each motor was analyzed individually to find the optimal gear ratio and propeller sizes. Each motor was then graphed vs. airspeed velocity for the following parameters: current, thrust, efficiency, and

power. The results were analyzed using a constant number of battery cells since the range for the FAI was determined to be similar. Figure 13 displays the current verses velocity. The FAI 642 drew the highest current load for the optimum conditions while the other two motors drew similar current loads. Similarly, analysis of the power drawn from batteries was lower for the 25 and 643 series motors since drawn current was lower, shown in Figure 13.

Thrust analysis can also be seen in Figure 13. The thrust for the 642 was higher than the 25 and the 643. All three motors had a broad efficiency band around the designed airspeed. The best efficiency around designed airspeeds was determined to be the 643, with a broad band but peaked out near the designed airspeed.

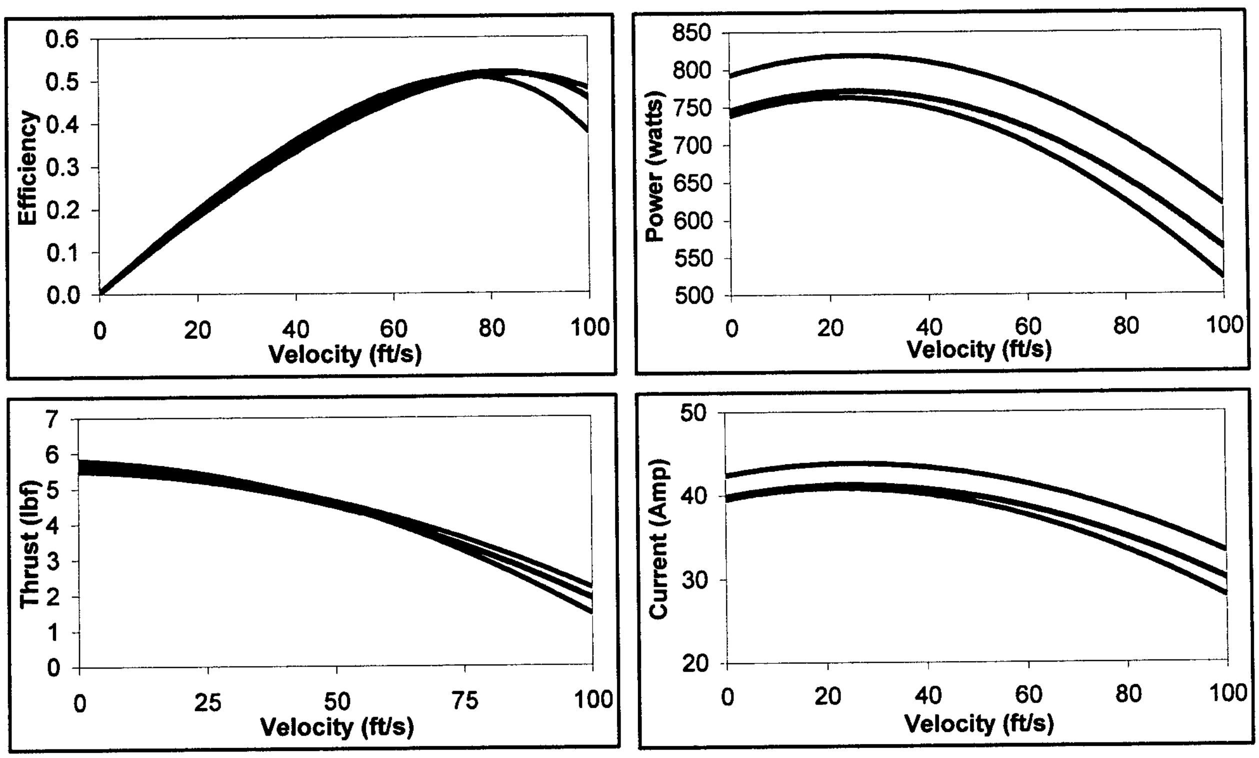


Figure 13: FAI Motor Performance Data. From top left clockwise: Efficiency, power available from the batteries, current, and takeoff thrust. Blue represents the FAI 25 motor, red represents the FAI 40 p/n 642 motor, **and black represents the FAI 40 p/n 643 motor.**

4.4.5 Motor Selection

The Cobalt motors had lower efficiencies for the designed airspeeds and required currents than the FAI motors. The required number of cells for the Cobalt motors was also higher then the FAI motors. Because of the trend between motors, the FAI series were chosen. All of the FAI motors had similar characteristics. The efficiencies for all three motors were fairly close. The FAI motors seemed to perform best at a p/d of 0.8. The FAI 40 p/n 642 was not chosen because the 643 had a better airspeed efficiency

while keeping every thing else constant. The FAI 25 and 643 had similar efficiencies and takeoff thrusts. The FAI 642 required a current greater then 40 amps for takeoff, while the 643 required a slightly smaller current. The simulation was done for a specific head wind. For different head winds, a larger range of propellers could be used on the 643 than on the FAI 25 without drawing too much current for an extended period of time. Based on historical data, propellers with a p/d ratio of approximately 0.8 and a diameter of 13.5-15 was determined. Based on these factors, the FAI 643 motor was selected with a gear ration of 1:3.1, a battery pack size of 17 cells, and propeller size of 13.5-15.

4.5 Analytical Tools

4.5.1 Optimization Program Architecture

After the conceptual phase was completed, the weight, drag, propulsive, and Rated Aircraft Cost models in the optimization were updated. The previous models were considered conceptual simulations because the previous versions were developed from the Oklahoma State historical database, and the drag model was developed from a drag buildup method. The following models describe the updates to the program:

- Weight Model: The weight model was reconstructed according to the structural and propulsive assessments presented earlier in the conceptual design phase. The weight model was checked against the OSU historical data and the previous weight model.
- Drag Model: The drag model was reconstructed according to the drag build-up method found in Raymer (1999). The method gave a closer estimate of drag without requiring the parameters developed in the preliminary and detail design phases. The drag model also included the drag polar and lift curve for the chosen airfoil. Different values were used for both cruise and take off Reynolds numbers. Conservative margins of safety were also placed on the model (approximately 25%).
- Propulsive Model: The propulsive model was compared to the analytical model used by the propulsion group in their analysis. The model showed that the conceptual model had been too conservative. The new model was reconstructed to remain conservative yet represent the propulsive system better than the conceptual model.
- Rated Aircraft Cost Model: The RAC was also updated to meet all the RAC requirements, and better represented the features chosen in the conceptual design phase.

The drag, weight, propulsive, and RAC models were placed in the program along with the parameters that best support an optimal design. The values included a wind speed of 10 miles per hour, nine softballs, and a lift coefficient of 1.2. The program was adapted so that the scoring potential was maximized and trade studies occurred using the optimization process.

4.5.2 Other Methods Used

Other than the optimization, several other programs were used to calculate various parameters of the design. Classical stability derivative analysis was used to size the horizontal and vertical components of

the V-tail. An analytical propulsion model was use to compare several combinations of motors and propellers to narrow down the wide range of options. The weight and center of gravity was also calculated using another program. Each structural analysis for each component was completed using several different spreadsheets so that as changes occurred, the updated structural data could be calculated. Initially sizes and component placements were laid out in a two-dimensional full-scale station line chart using yarn and construction paper to represent components and contour lines placed on pegboard. Then the layout was transferred to the CAD program for three-dimensional modeling.

4.6 Final Aircraft Configurations

At the end of the preliminary phase, the primary components of the configuration were all sized and placed according to the figures of merit. The wing was sized with a span of 9.5 feet and chord of 8.625 inches. The tail was sized so that the planform area of 1.3 square feet was kept within the quarter span of the wing. The fuselage length was minimized using the greatest suggested rake angles. The propulsion system was narrowed down to a few motor and propeller combinations to be tested in detail. The spars and ribs in the wing and tail were sized, along with the makeup of the fuselage monocoque design. The gear was designed so to soften landing shock while not causing the plane to bounce back into the air. A three dimensional model of the primary components was also completed.

4.6.1 Wing and Power Loading

Wing and power loading for the preliminary configuration was 3.5 lb/ft^2 and 47.4 lb/hp. The two loading values were a compromise to achieve the correct mission balance and achieve the best scoring potential. Higher wing loading allowed the plane to penetrate through the wind better, achieve higher cruise velocities, and makes the configuration less susceptible to gusting conditions because greater changes in pressure differential are required to disturb the plane. Drawbacks to high wing loading occurred at takeoff and climb. Higher wing loading was not helpful during the critical phases. To overcome the wing loading disadvantage, higher power loading was required to help the plane overcome takeoff and climb requirements. Therefore, a compromise was met between wing and power loading to result in the preliminary configuration.

5.0 Detail Design

After the primary components were sized in the preliminary phase, secondary components were sized and added. The aerodynamics group determined the dynamic stability as well as performance estimates and aircraft handling qualities. Hardware for component mating and control surface integration were selected. Manufacturing processes and tooling were developed to construct the aircraft and detail drawings for manufacturing the aircraft components were produced by the structures group. The propulsion group used the preliminary system combinations and experimental data to decide on the final propulsion configuration and number of batteries. The final configuration was then constructed and tested as a prototype to determine problems that had been previously overlooked.

5.1 Aerodynamic Performance Analysis

The optimization code was modified to provide data on the flight performance of the final configurations. The use of an airfoil analysis program allowed the characteristics of the airfoils of the wing and tail to be included in the analysis. The following sections contain results from the two computer codes, representing how the plane would perform.

5.1.1 Final Configuration Features

The final aircraft configuration is fairly conventional consisting of a low mounted wing with a large aspect ratio, approximately 13.2. The fuselage is contoured around the interior components creating a streamlined and drag efficient body. The tail is a V-tail configuration with the span of 1.75 feet, the quarter wingspan limit to avoid the RAC penalty if the tail was classified as a wing. The fins of the V-tail are approximately 40 degrees from the horizontal with a taper ratio of 0.5. The landing gear, which is 8 inches tall, provides a tail scrape angle of 25 degrees for safe pitching at takeoff. The wings can rock 13 degrees on takeoff before scraping the runway. All major components are designed to shear off under severe loading to dissipate the energy and reduce the damage sustained by the aircraft. A more complete description of the aircraft configuration features is listed in Table 8.

Fuselage	
Length	4.77 ft
Maximum width	12 in
Tail Properties	
Airfoil	S6063
Span	1.75 ft
Chord at root	14 in
Chord at tip	7 in
Projected horizontal area	1.53 ft^2
Projected vertical area	1.07 ft^2
Taper ratio	0.5
Incidence angle	0^o
Angle to horizontal	40^o
Ruddervator area	41.1 in^2

Wing	
Airfoil	SD 7062
Span	9.5 ft
Chord	8.625 in
Area	6.63 ft^2
Incidence angle	0^o
Aileron area (per wing)	57 in^2
Total aircraft	
Center of gravity location	25 in
Maximum weight	23.7 lbs
Number of softballs	24
Weight of softballs	9 lbs
Rated Aircraft Cost	8.765
Score (without Written Score)	1.75

Table 8 : Final Aircraft Configuration Features

5.1.2 Estimated Mission Performance

The estimated mission performance of the configuration was calculated from the performance code used in the optimization. Each stage of the mission was individually evaluated. The time consumed for each step was of particular interest. Table 9 shows the time spent in each portion of the mission profile. The time spent in cruise, both loaded and unloaded, was the total time at cruise velocity, including all turns and straight legs. Also included in Table 9 were the expected flight velocities and distance of each phase. The performance information was calculated assuming a wind speed of 10 mph. Accounting for

all phases of the mission as well as the RAC, the aircraft was predicted to score 1.75 times the report score.

Mission Components	Time Performed	Time Spent	Distance	Velocity
Takeoff unloaded	2	2.6 sec	37.81 ft	44.25 fps
Climb unloaded	2	10.2 sec	---	46.25 fps
Cruise unloaded (all 4 laps included)	1	89.2 sec	---	85.13 fps
Slow down unloaded	2	14.4 sec	---	---
Ground time for loading	1	45 sec	---	---
Takeoff loaded	1	8.6 sec	180 ft	56.10 fps
Climb loaded	1	24.1 sec	---	59.14 fps
Cruise loaded (both laps included)	1	75.8 sec	---	79.61 fps
Slow down loaded	1	9.11 sec	---	---
Ground time for unloading	1	45 sec	---	---
Total Time	---	5.85 min	---	---

Table 9: Time Spent in Mission Phases. Value predicted by performance program.

5.1.3 Takeoff and Climb

The takeoff distance for loaded conditions was 180 feet from the optimization code. Takeoff velocity was approximately 55 fps. The plane needed to accelerate after takeoff to a velocity of 60 fps to maintain the proper climb speed. For unloaded conditions, the takeoff distance decreased significantly to 40 feet, and takeoff velocity also decreased to a speed of 45 fps. When the plane accelerated to climb velocity at unloaded conditions, the plane moved at an airspeed of 47 fps. The time for takeoff and climb scenarios can be seen above in Table 9. The power loading at takeoff was determined to be 47.4 lb/hp, when the plane was fully loaded with softballs.

5.1.4 Flight Conditions

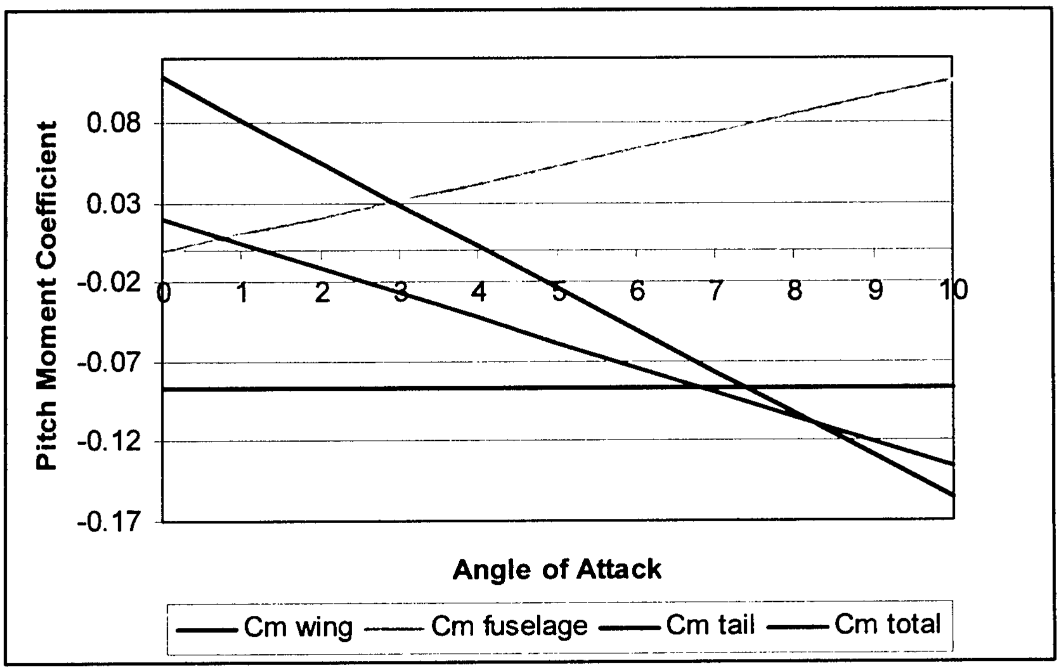
The aircraft was designed to perform best in a 10mph wind. At this wind speed, the empty cruise speed was projected to be 85.1 fps while the loaded speed was projected at 79.6 fps. When the plane carried the 9-pound softball payload, the wing loading is 3.5 lb/ft^2.

5.1.5 Handling Qualities

Once the final dimensions were known, the handling characteristics were calculated and trimmed to satisfy stability requirements. For example, the pitching moment contributions of each component were calculated and then summed to evaluate the pitching characteristics of the entire aircraft. Figure 14 shows the effect of each component on the total pitch moment. From the graph, the aircraft trimmed at 1.25 degrees angle of attack in cruise. The static margin was also calculated and found to be 13%. While the value produces a more stable plane, the preliminary estimates ranged from 5% to 10%.

Other handling qualities of interest were the control power derivatives, which are also listed in Figure 14. The control power was found with an airfoil program by determining the change in lift on the surface due to the deflection of the control surface. After correcting for the aspect ratio, the change in lift was converted into the appropriate moment and non-dimensionalized. The equations used in the static stability and control power analysis were obtained from Raymer (1999) and Nelson (1998).

Test flights of the prototype revealed even more handling quality issues than expected. One critical factor was the visibility of the aircraft. Without visual cues to help the pilot determine the orientation of the plane, maneuvers performed at the distant ends of the circuit become very difficult. This problem was compounded by designing the plane to be only slightly roll stable. The prototype testing also demonstrated the pitch and weathercock stability of the aircraft in takeoff under windy conditions.



Static Margin	$C_{L\alpha}$	$C_{m\alpha}$	$C_{n\beta}$	$C_{l\beta}$
13%	5.05	-0.895	0.085	-0.016
$C_{m\delta e}$	$C_{n\delta r}$	$C_{n\delta a}$	$C_{l\delta r}$	$C_{l\delta a}$
-1.16	-0.0480	-0.0142	0.0147	0.194

Figure 14: Coefficient of Moment of Aircraft Components Versus Angle of Attack and Stability Derivatives

5.2 Propulsive Performance Analysis

The final propulsion system was designed in the preliminary phase using analytical methods. Using a dynamometer, the propulsion group tested several of the top motors that remained at the end of the preliminary phase. The final motor and propeller combination was decided to be the AstroFlight 643S, an FAI 40 motor with a super box, and a 14.4 inch diameter propeller with a p/d ratio of 0.8. A gearbox was

not needed for this combination to produce the thrust and velocities required. Seventeen batteries were needed to produce the thrust. The batteries were decided to be 2400 milliamp hour, high rate, SR NiCad batteries. The final output from the above propulsion setup was found to be a takeoff thrust of 4.8 pounds, using approximately 760 watts from the batteries. Cruise thrust was found to be 1.9 pounds and only drew 490 watts from the batteries. The above values and setup were used in the construction of the prototype and refined the optimal results.

5.3 Structural Considerations

The focus of the detail design phase was the integration of the primary components with each other and the various systems comprising the aircraft as a whole. The details of the four major aircraft components, the fuselage, wing, tail, and gear, were designed to insure form, fit, and function of each component.

5.3.1 Fuselage Structural Details

The form of the fuselage was designed to accommodate the internal components and payload while exhibiting the lowest drag possible. Fuselage design was accomplished using CAD to shape the fuselage around internal components while minimizing wetted surface area. The contours of the fuselage were also designed to blend from one component to the next in a way that minimized interference drag between components. Functionally, the fuselage served as the junction between all aircraft systems, components, and payload. Propulsion system components were arranged in a fashion that would minimize wire length between components. The close packed configuration also allowed airflow to be routed from a chin scope in the forward bulkhead over the motor and around the battery packs. The payload deck was designed to be lightweight and removable. Lightening holes in the payload deck acted as cradles to secure the softballs during flight as well as permit precise loading and unloading. The payload deck was supported by the wing carry through section and both the forward and aft bulkheads. The payload hatch was designed to be removed in a simple quick motion allowing quick turn around times. The hatch release mechanism consisted of a four pins mounted on the fuselage, which slide into grooves cut into the payload hatch. The mechanism allows the hatch to be removed with a simple push, slide, and lift motion. Another factor in ground time was a fuse that could be easily and quickly removed. Finally, component weights and station locations were entered into a weight balance spreadsheet, Figure 16, to keep track of aircraft inertial properties and to place the aircraft center of gravity at the center of lift.

5.3.2 Wing Structural Details

The internal wing structure was comprised of four ribs, a full-length spar, and blue foam core. The foam core holds the airfoil shape while the spars and ribs provide load paths to the fuselage. Accommodations for the aileron servos were made in the two outboard spars, which can be accessed via removable plates on the lower surface of the wing skin. The plates were included in the wing lay-up during construction to ensure a snug fit and are secured by small screws. The hinge mechanism for the aileron was designed to minimize the gap between the aileron and wing surface. Graphite arrow shafts were embedded into the

wing core and into the aileron before the wing skins was applied and cured. After curing a second smaller graphite shaft was inserted though the tip of the wing into the larger shafts. The shafts in the aileron pivoted around the smaller concentric shaft creating a low friction lightweight hinge. The wing was also designed to breakaway in the incident of a crash. The breakaway design was to minimize damage to wing as well as the rest of the fuselage. The wing carry through section in the fuselage was designed to allow the wing to shear off the aircraft without damaging the structure. The breakaway was accomplished through the use of nylon connector bolts that are designed to fail under the high shear loads associated with a wing tip strike. The connecting wires for the servos embedded into the wing were also designed to shear without damaging the servos. Using CAD, design goals were incorporated into the wing without distorting the airfoil shape. Figure 15 illustrates the component configurations used to efficiently use available space and the contour the shape around the interior layout.

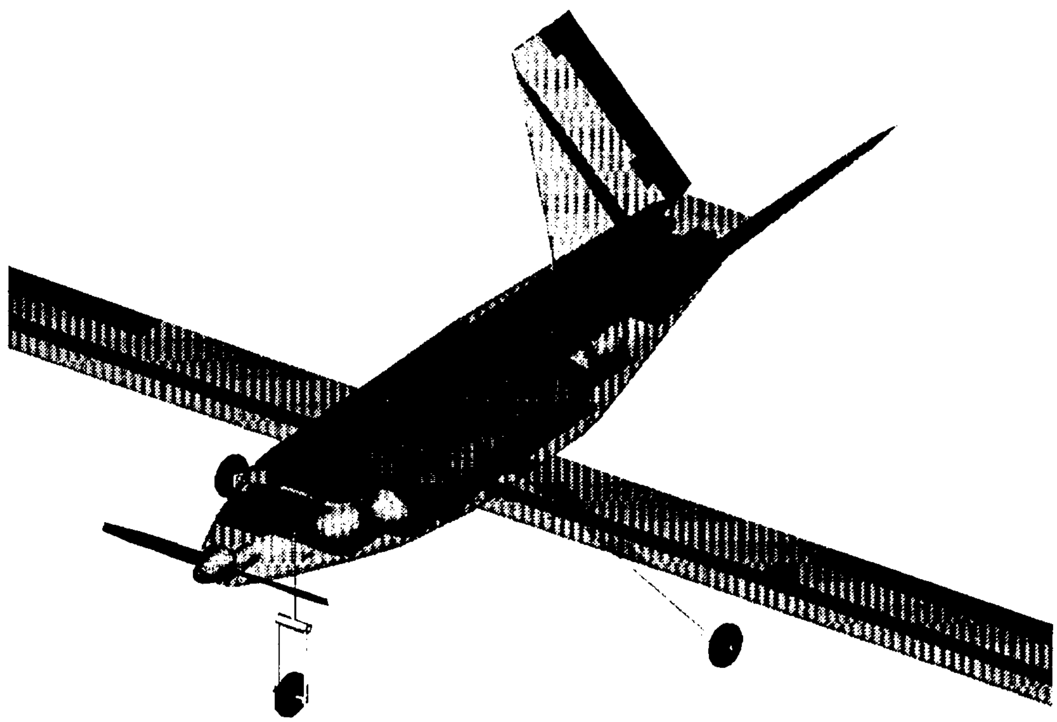


Figure 15: Transparence of Entire Plane

5.3.3 Tail Structural Details

The tail section was designed with similar goals to the wing. Once again graphite arrow shafts served as the hinge mechanism for the control surfaces. Like the wing a single carry through spar was incorporated to transfer loads to the fuselage. The tail was also built as a single module that could be easily removed and that could break away in the event of a crash with minimal damage to the aircraft structure. Servos

were mounted in the tail module with breakaway wires to allow them to remain with the tail instead of damaging the control surfaces during a crash. Nylon bolts were again incorporated as the shear mechanism to allow the tail to detach before it fails structurally. Control rods were placed in an area that will minimize drag and allow the servo to push instead of pull against the higher loads associated with pitching the nose up. All of the design goals for the tail were incorporated into a compact design that could be constructed, repaired, or even replaced with minimal modification to the rest of the aircraft.

5.3.4 Structural Ground Test Results

Before ground and flight-testing commenced a series of test were performed to test the interior and exterior structures of the aircraft. The wing was subjected to a 3.4 g load through the use of weights and a test stand. Deflections estimated were within 2% of the predicted values. The fuselage was also tested to a 3.4 g load. The fuselage deflection was within 5% of predicted values however the sides of the fuselage bowed out of tolerances under load. To correct for this stiffeners and doublers were placed in the high stress areas. In order to test the landing gear system an apparatus was developed to attach the gear to. Weight were added to the test rig and the gear was sent through a series of drop and impact tests. The bow gear provided adequate spring damping and deflection and stroke were very close to predicted values.

5.4 Detail Design Assumptions and Comparisons

During detail design analytical assumptions made in preliminary design were reinforced by static structural testing, propulsion dynamometer testing, and finally prototype flight-testing. The prototype aircraft, less payload and batteries, weighed in at 10.75 lbs during flight test number one. The models used to perform the structural analysis in preliminary design estimated the weight to be 12.39 lbs. The numbers helped validate the assumptions used to perform the structural analysis. Center of gravity estimates, shown in Figure 16, proved to be valid to within one half of an inch. In flight the aircraft proved to be stable in pitch and yaw but assumptions made concerning roll stability proved to be inadequate to produce adequate control. Two degrees of dihedral were added to correct for this. Takeoff thrust and climb performance were better than expected primarily due to the difference between early weight assumptions and actual structural weight.

5.5 Drawing Package

Detail drawings are provided in the last five pages of this document.

Aircraft Component	Station (inches)	Weight (pounds)	Moment (pound-in)
Batteries			
Brake battery	36	0.0882	3.17
Propulsion batteries	19	2.4119	45.83
Servo batteries	34	0.2006	6.821
Propulsion System			
Engine	2	0.8444	1.689
Propeller	-1	0.2050	-0.205
Watt-Meter	6	0.2491	1.495
Aircraft Structures			
Brake Controller	36	0.0441	1.587
Cargo Deck	24	0.4189	10.05
Fuselage (with hatch)	25	2.6081	65.20
Main Gear	27	0.8686	23.45
Nose Gear (with servo)	8	0.6151	4.921
Tail (with servos)	51	1.3999	71.57
Wing (with servos)	25	3.3951	84.88
Payload			
Payload of 24 softballs	11	9.764	107.4

Figure 16: Weight Balance Spreadsheet. Center of gravity located at wing quarter chord.

6.0 Manufacturing Plan

6.1 Manufacturing Process Investigated and Figures of Merit

In order to produce the aircraft, several different manufacturing techniques would have to be employed for the different components. Several different methods for obtaining the desired shapes and surface finishes were investigated for each component. The figures of merit used to evaluate different manufacturing techniques included the ability to produce the desired shape and finish, cost of tooling and manufacturing, build time, required materials, skill levels, and process repeatability.

6.2 Processes Selected for Component Manufacturing

6.2.1 Fuselage Manufacturing Process and Tooling

Careful consideration was given to the exterior shape and surface of the fuselage. The fuselage was shaped in a way that minimized wetted area and interference drag, resulting in a complex shape with compound curves and radii. Careful attention was also given to the surface finish on the outer mold line. Both male and female molds were considered to create the desired fuselage.

The principle difference between the methods was the dimensional tolerances and the outer surface finish. A female mold allows the dimensional tolerance to be kept on the outer mold line while providing a smooth exterior surface of the carbon fiber resin. A male mold would keep dimensional tolerance on the inner mold line. In order to produce a smooth exterior surface with a male mold, a separate mold conforming to the outer surface would have to be prepared and included in the lay-up. Both methods had

similar requirements in terms of tooling and manufacturing time. The female mold held an advantage in that the exterior finish and dimensional tolerances could be repeated easily on the final aircraft. Considering all figures of merit, the female mold was chosen.

Once the decision to use a female mold was made, the process of how to build the mold began. The first step was to create a male mold conforming to the outer mold line of the aircraft. Laminating resin was applied to the exterior of the male mold. The cured resin was sanded and polished to the desired finish. Several coats of release agents were applied to the surface and cured. A gel coat was than applied to the release material followed by several layers of fiberglass. The lay-up was vacuum bagged to the male mold and allowed to cure. The female mold, consisting of the fiberglass and the gel coat, was removed from the male mold and polished to the desired finish.

The final step in the fuselage construction was to lay-up in the female molds. The fuselage was split into top and bottom halves. Each section required a female mold. Hatches and access panels were included in the lay-up to ensure a tight fit against the fuselage. The lay-up consisted of carbon fiber and honeycomb core were vacuum bagged to the female molds in one assembly.

6.2.2 Wing and Tail Manufacturing Process and Tooling

The wing construction was a build up of foam core with a composite skin. To create the foam core, airfoil templates were created using computer software. The female sections of foam created from cutting the airfoil were finished using epoxy resin and used in the wing lay-up to give the resin in the carbon fiber prepreg a smooth surface to form against. The wing and spar was constructed in two sections and mated at the centerline of the fuselage. First, the foam airfoil for each section was cut and the spar material removed from the airfoil sections. The removed spar material was used as sandwich material for the carbon fiber spar skins. The front and aft sections of the airfoils were then re-bonded to the spar. Finally, the carbon fiber skins were applied and both sections vacuum bagged and allowed to cure separately. Bonding the spar sections and applying a wet lay-up technique to the center section of the wing mated the two sections.

The V-tail surfaces were constructed in similar fashion. The aft section of the fuselage and the V-tail constituted the tail assembly. Like the wing, the V-tails share a spar that carries through the aft section of the fuselage. The tail surfaces were foam core with carbon fiber skin and were constructed as one component; the tail assembly was then bonded to the aft fuselage section.

6.2.3 Landing Gear Manufacturing Process and Tooling

The composite lay-up for the bow gear was constructed on a male plug. The plug was cut from blue foam using templates made from CAD drawings. Layers of composite material were then laid up on top of the male plug. The nose gear was a stock Fults RF400 Dual Strut.

6.3 Analytic Methods Including Cost, scheduling and Skills matrix

6.3.1 Manufacturing Cost

With the preliminary design study complete, manufacturing and tooling costs were estimated. Projections showed sufficient carbon fiber prepreg was in stock to construct the prototype, leaving the major cost of manufacturing to be consumed by the propulsion equipment required for flight testing. Provisions for consumable materials such as breather, release, and epoxies were purchased for both the prototype and the final aircraft to cut down on lead-time. The total projected cost for the construction of the prototype aircraft and propulsion testing is $1175.

6.3.2 Skills Matrix

In order to assign tasks for the manufacturing process it was necessary to develop a matrix of the skills required for each task. Table 10 contains the skills matrix. In the matrix a component that requires a lot of skill in a certain area was rated a two, a component that required average skill in an area was score with a one, and a component that required no skill in an area received a zero. The columns of the skill matrix represent required skills in the manufacturing process; the rows represent the major assemblies and system of the aircraft. Members were assigned to components matching their expertise.

Primary Aircraft Assemblies and Systems	Foam Cutting	Composite Layup	Mold Preperation	Radio Equipment Installation	Electrical Work	CAD Modeling
Wing	2	2	1	2	1	2
Fuselage	2	2	2	2	2	2
Landing Gear	1	2	1	2	1	2
Tail	2	2	1	2	2	2
Propulsion System	0	1	0	2	2	1

Table 10: Skills Matrix for Oklahoma State Orange Team

6.3.3 Manufacturing Scheduling

The aircraft was constructed in four assemblies: the fuselage, wing, tail, and landing gear. The assemblies were constructed to allow components to be constructed simultaneously. Considerations such as material availability and coordination of oven times were of critical importance to maintaining a smooth manufacturing process. Figure 17 is the milestone chart developed for the manufacturing process. Figure 18 shows a photograph of the completed prototype after final assembly and completion of the prototype rollout milestone.

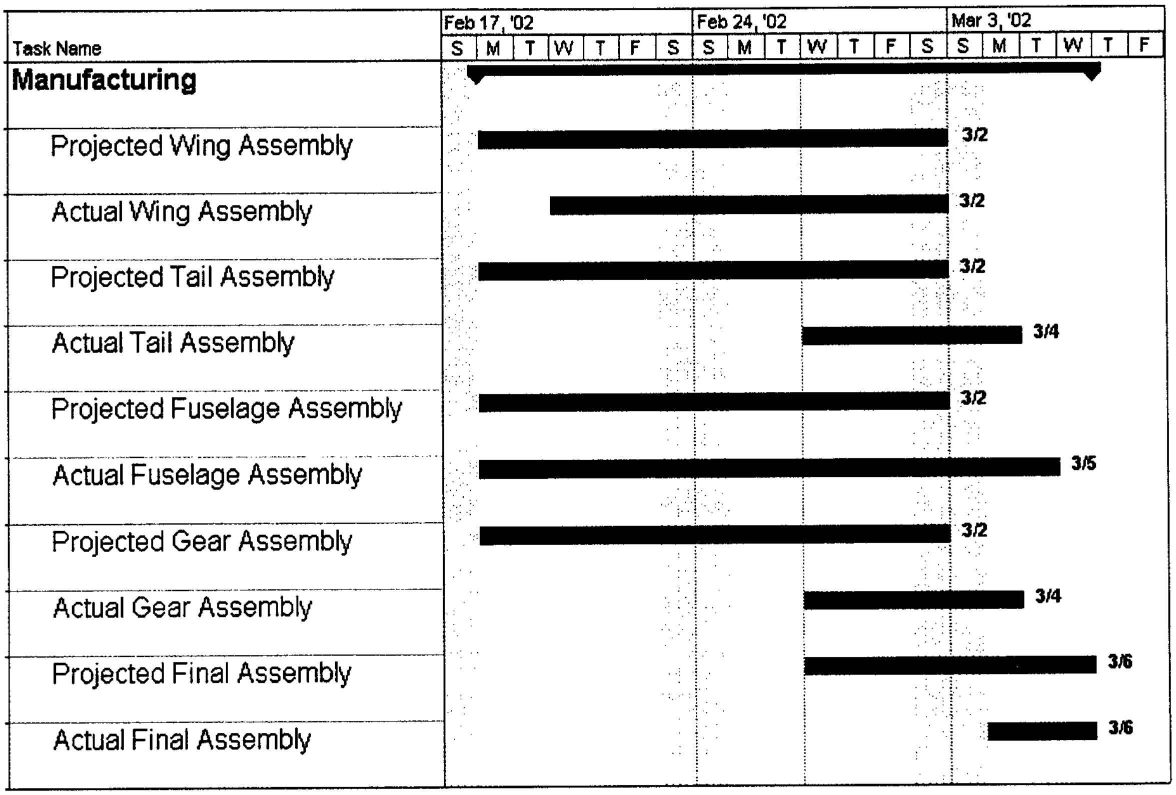
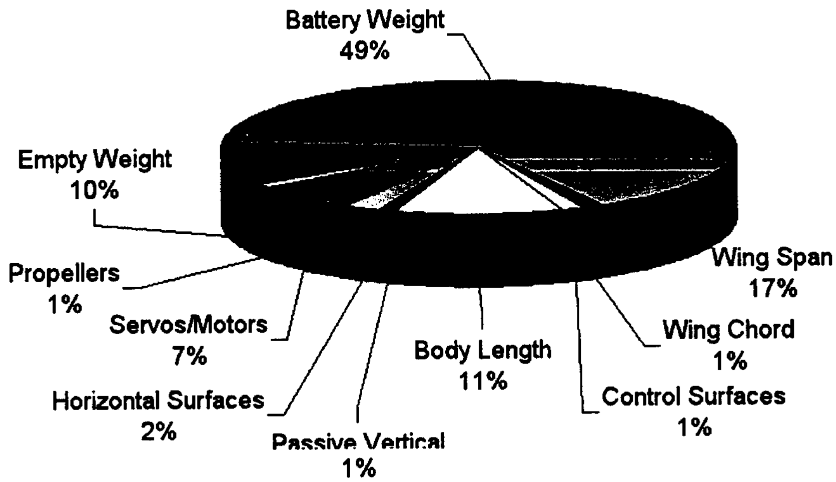


Figure 17: Manufacturing Schedule. Red represents the projected dates; green represents the actual dates.

Figure 18: Picture of the Orange Team Prototype

7.0 Rated Aircraft Cost

After parameters of the plane were determined, the dimensions were entered into the RAC program developed in the conceptual design phase. The RAC of the design was 8.925. The breakdown of each component's contribution to the RAC value can be seen in Figure 19.



Manufacturing Man Hours: Work Breakdown Structures					
Aircraft Component		Man Hours/Unit	Aircraft Parameter	Individual Hours	Individual Contributions
Wing WBS - sum for multiple wings					
	Wing Span	8 hr/ft	9.5 feet	76	1520
	Maximum exposed chord	8 hr/ft	0.71875 feet	5.75	115
	Control Surface	3 hr/surface	2 surfaces	6	120
Fuselage WBS					
	Maximum body length	10 hr/ft	4.75 feet	47.5	950
Empenage WBS					
	Vertical surface with no active control	5 hr/surface	1 surface	5	100
	Horizontal surface	10 hr/surface	1 surface	10	200
Flight Systems WBS					
	Servo/motor controller	5 hr/servo	6 servos	30	600
Propulsion Systems WBS					
	Engines	5 hr/engine	1 engine	5	100
	Propeller/fan	5 hr/propeller	1 propeller	5	100
		Total Manufacturing Man Hours			**3805**

Description	Value Definition	Multiplier	[illegible]	[illegible]	[illegible]
Manufacturers Empty Weight (MEW)	Actual airframe weight, lb., with all flight and propulsion batteries but without any payload.	$100	10.7 lb.	NA	**1070**
Rated Engine Power (REP)	"Total Battery Weight" defined as weight of propulsion batteries.	$1,500	1 Engine 2.7 lb. Batteries	NA	**4050**
Manufacturing Man Hours (MMH)	Sum of assembly hours defined by Work Breakdown Structure (WBS).	$20/hour	----	190.25	**3805**
		Rated Aircraft Cost			**8925**

Figure 19: Rated Aircraft Cost Worksheet and Pie Chart. Predicted Rated Aircraft Cost of 8.925

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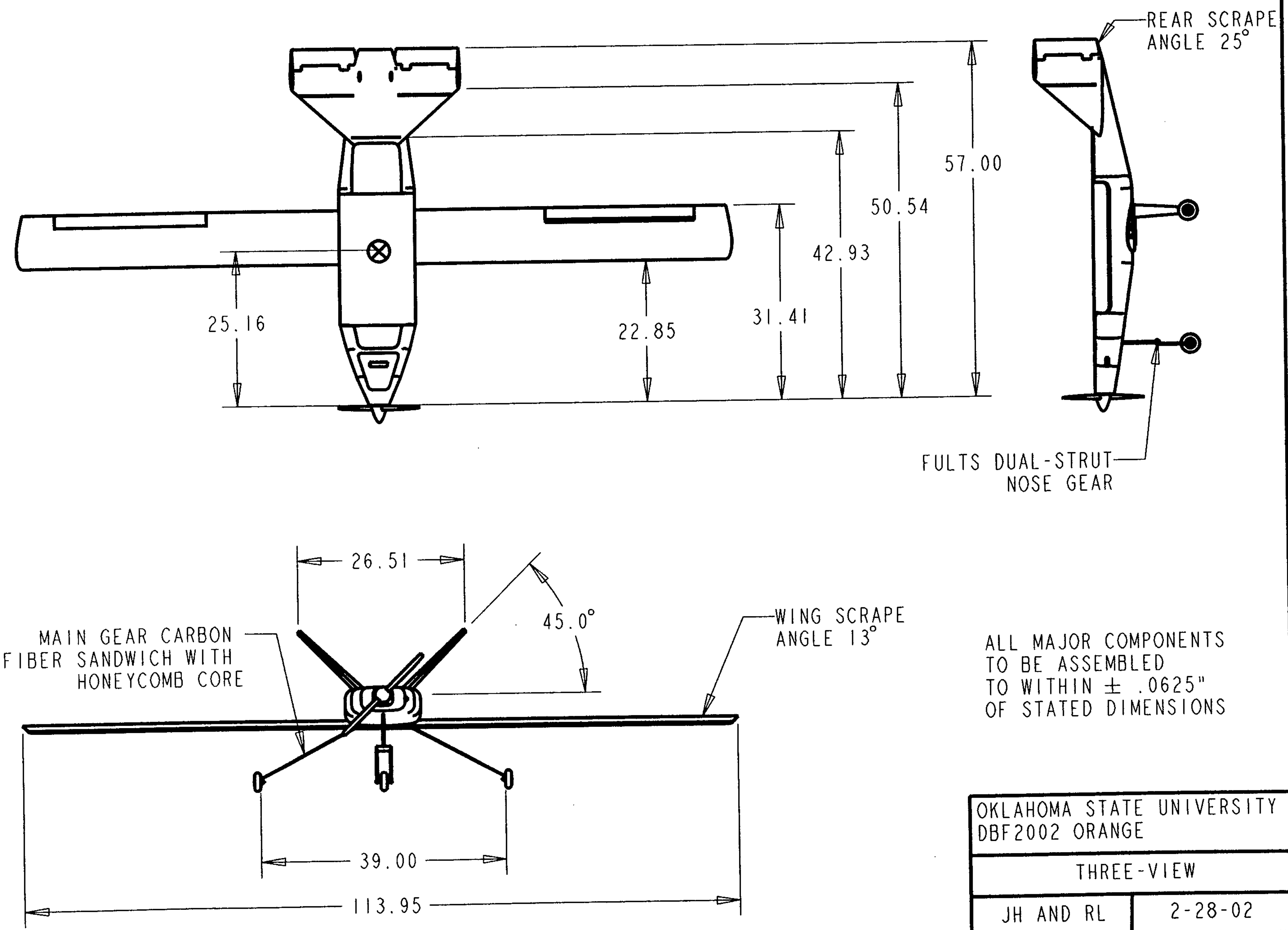
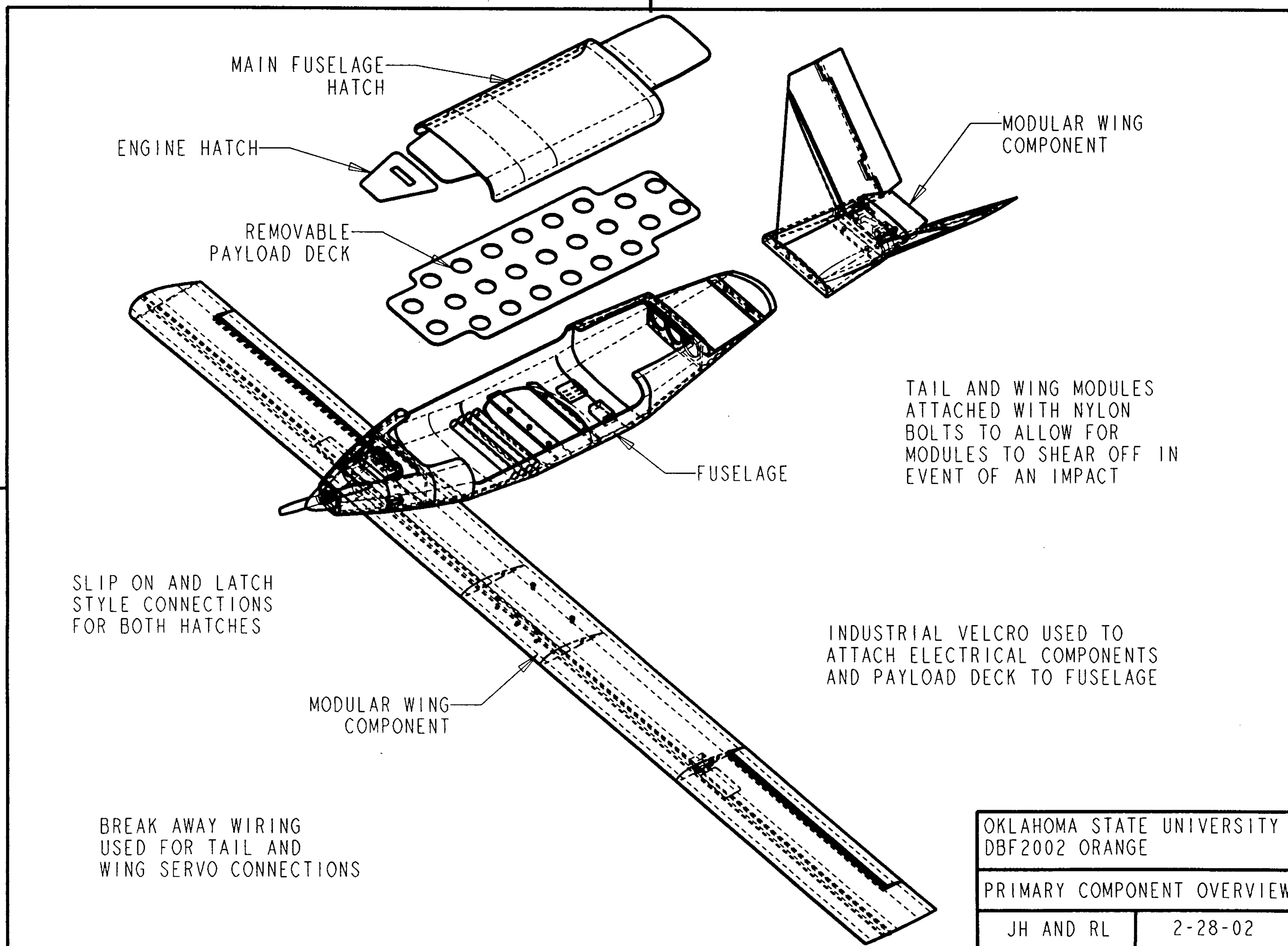
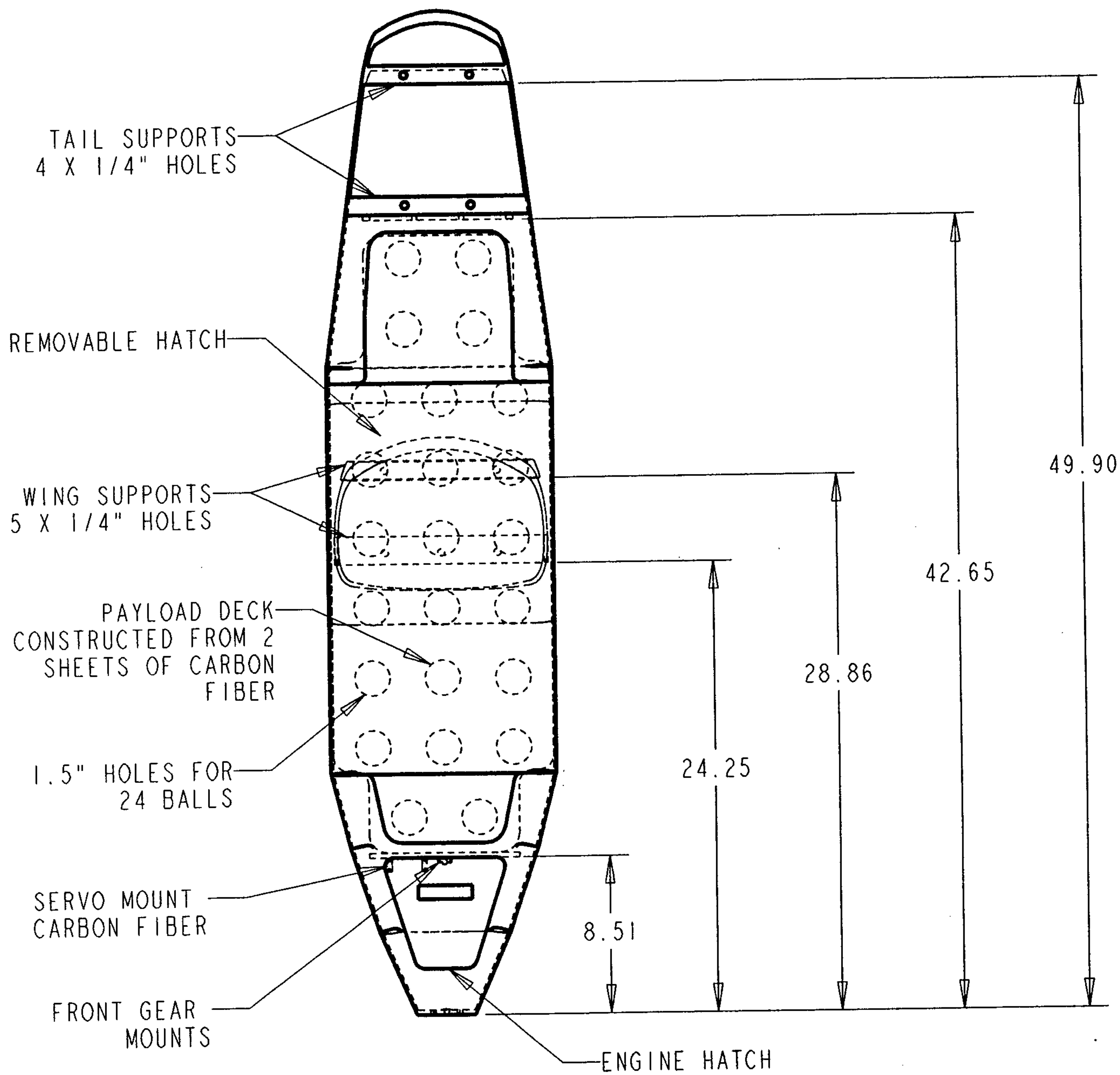


REAR SCRAPE ANGLE 25°
57.00
50.54
42.93
31.41
22.85
25.16
FULTS DUAL-STRUT NOSE GEAR
26.51
45.0°
MAIN GEAR CARBON FIBER SANDWICH WITH HONEYCOMB CORE
WING SCRAPE ANGLE 13°
39.00
113.95
ALL MAJOR COMPONENTS TO BE ASSEMBLED TO WITHIN ± .0625" OF STATED DIMENSIONS
OKLAHOMA STATE UNIVERSITY DBF2002 ORANGE
THREE-VIEW
JH AND RL
2-28-02


MAIN FUSELAGE HATCH
ENGINE HATCH
REMOVABLE PAYLOAD DECK
MODULAR WING COMPONENT
FUSELAGE
TAIL AND WING MODULES ATTACHED WITH NYLON BOLTS TO ALLOW FOR MODULES TO SHEAR OFF IN EVENT OF AN IMPACT
SLIP ON AND LATCH STYLE CONNECTIONS FOR BOTH HATCHES
INDUSTRIAL VELCRO USED TO ATTACH ELECTRICAL COMPONENTS AND PAYLOAD DECK TO FUSELAGE
MODULAR WING COMPONENT
BREAK AWAY WIRING USED FOR TAIL AND WING SERVO CONNECTIONS
OKLAHOMA STATE UNIVERSITY
DBF2002 ORANGE
PRIMARY COMPONENT OVERVIEW
JH AND RL
2-28-02

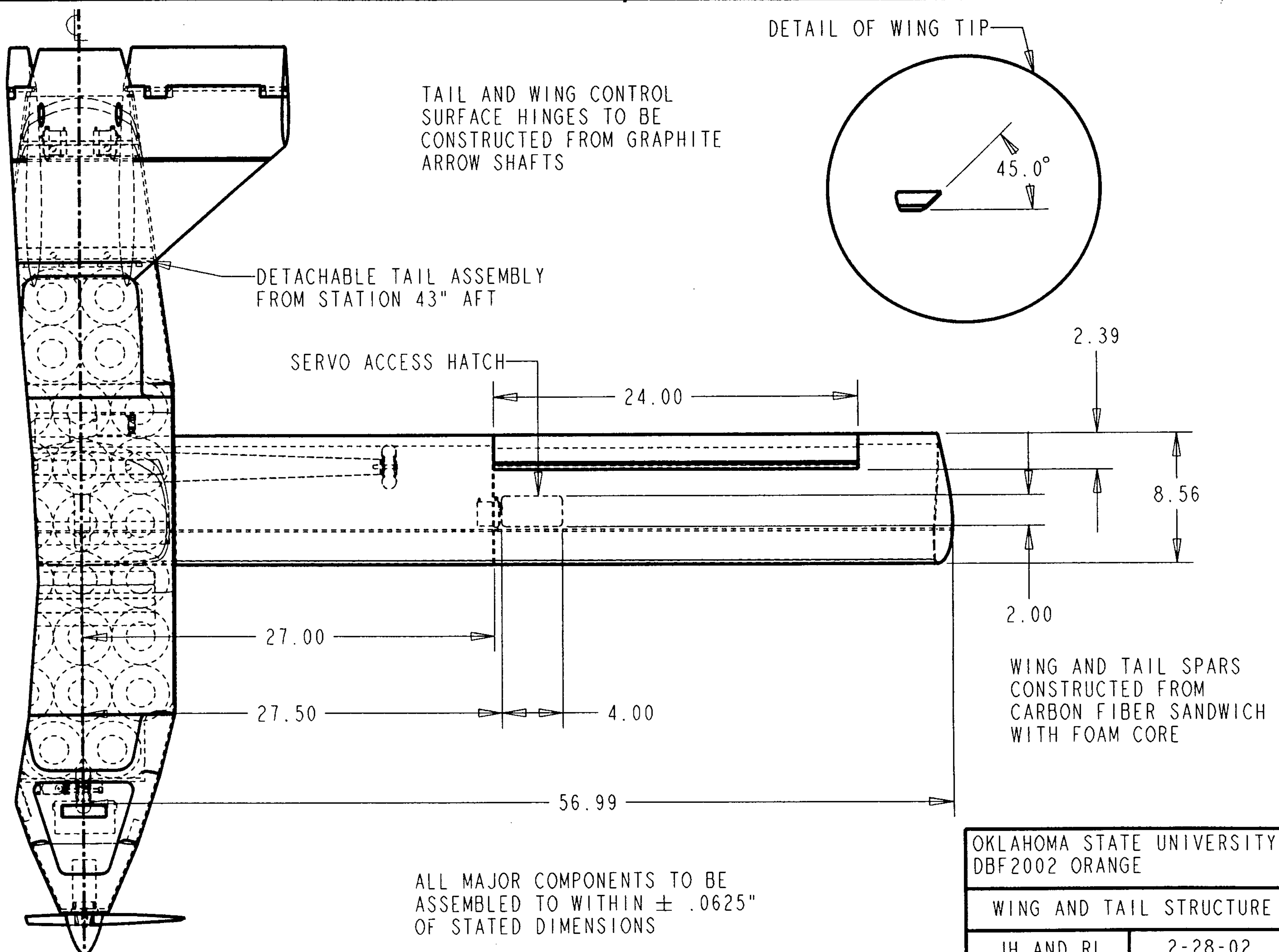


WING AND TAIL ATTACHED TO FUSELAGE WITH 2" X 1/4" 20 NYLON BOLTS

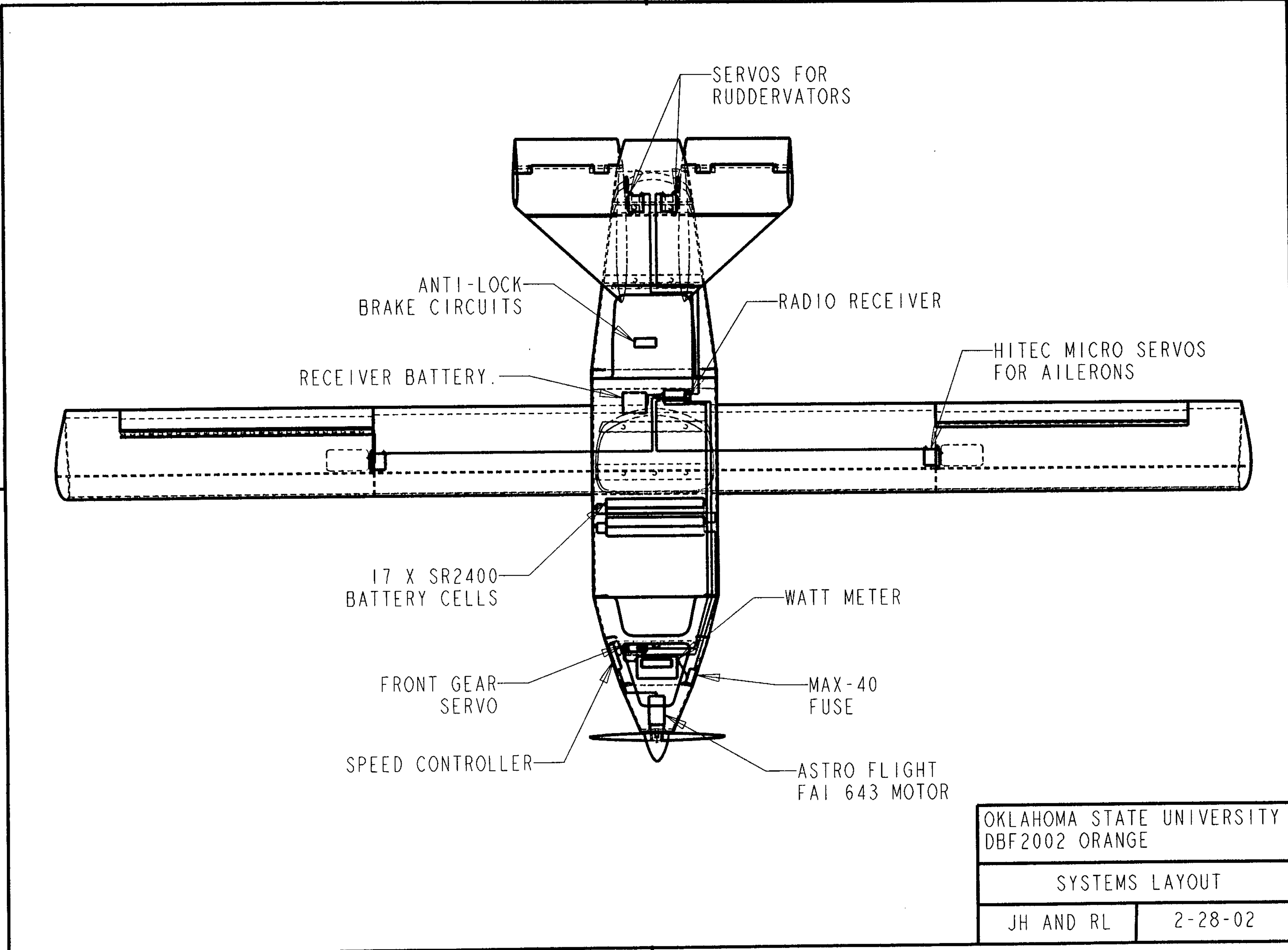
ACRYLIC WINDOW DISPLAY ON ENGINE HATCH FOR EASY VIEW OF WATT METER

ALL MAJOR COMPONENTS TO BE ASSEMBLED TO WITHIN ± .0625" OF STATED DIMENSIONS

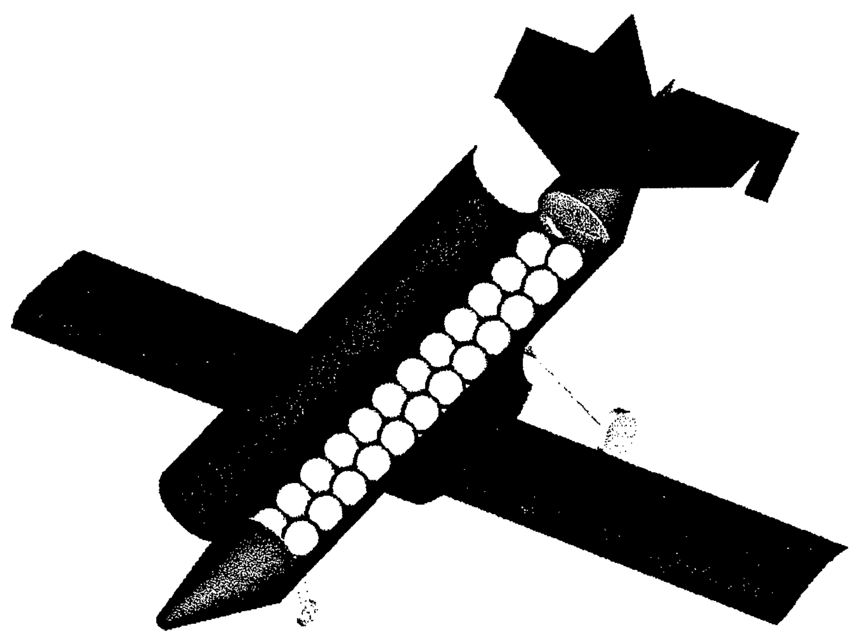
OKLAHOMA STATE UNIVERSITY DBF2002 ORANGE	
FUSELAGE DIMENSIONS	
JH AND RL	2-28-02



OKLAHOMA STATE UNIVERSITY DBF2002 ORANGE	
WING AND TAIL STRUCTURE	
JH AND RL	2-28-02


SERVOS FOR
RUDDERVATORS
ANTI-LOCK
BRAKE CIRCUITS
RADIO RECEIVER
HITEC MICRO SERVOS
FOR AILERONS
RECEIVER BATTERY.
17 X SR2400
BATTERY CELLS
WATT METER
FRONT GEAR
SERVO
MAX-40
FUSE
SPEED CONTROLLER
ASTRO FLIGHT
FAI 643 MOTOR
OKLAHOMA STATE UNIVERSITY
DBF2002 ORANGE
SYSTEMS LAYOUT
JH AND RL
2-28-02

2001/2002 CESSNA ONR / AIAA
STUDENT DESIGN / BUILD / FLY COMPETITION
PROPOSAL PHASE DESIGN REPORT



"TERP FLYER"

UNIVERSITY OF MARYLAND, COLLEGE PARK

TABLE OF CONTENTS

1. EXECUTIVE SUMMARY

The design of the "Terp Flyer" radio-controlled aircraft was begun with consideration by the design team of three concepts; these concepts were derived through creative 'brainstorming' by the design team, and through an investigation of the merits of previous designs. After choosing a concept to be developed, the team continued the design process by conducting trade-off studies and numerical analysis to evaluate configuration and design parameters. Finally, design drawings were completed, and a manufacturing plan was devised.

1.1 CONCEPTUAL DESIGN

1.1.1 Design Alternatives

Proposal and discussion at team design meetings produced three major design concepts; a flying wing concept, a pod-boom fuselage concept, and a conventional mono-wing concept.

The first concept investigated by the Terp Flyer Team was the flying wing. The team's goal was to have the wing/fuselage combination function as the main source of lift (as is typical of flying wing aircraft) with a tail extended aft of the wing/body for control. The top portion of the flying wing design would be faired so that the softballs could be loaded and removed with ease by removing the fairing.

The second concept the Terp Flyer team considered was that of a pod-boom fuselage design. The cargo would be carried in two separate fuselages, or pods, with two rows of six softballs each. A third central pod would contain the electronics of the aircraft—the motor, batteries, and communications equipment. The main wing would be positioned above the three pods, and would be raised slightly from the fuselages to limit interference drag associated with typical wing-fuselage joints. To account for the main wing lying above the three fuselages, two small airfoil shaped pylons would connect the booms and act as structural elements for the aircraft. Finally, the vertical tail would protrude from the center fuselage.

A third design concept considered was a conventional aircraft, with a single fuselage, one main wing and a conventional tail. The cargo would be carried within the fuselage. The aircraft would be constructed with a fairing or structural door capability on the upper surface or sides of the main fuselage.

1.1.2 Analytical Methods

The rules for the competition specify an equation for the calculation of rated aircraft cost (RAC), which provides a numerical standard by which to gauge the aircraft's monetary value. An aircraft

designed with a lower RAC will receive higher competition scores. A MATLAB program was written by the Terp Flyer team, which calculates the RAC, as well as the Manufacturer's Empty Weight (MEW), which is a parameter commonly used in the aircraft industry to quantify the efficiency of a design. In addition to these calculations, the team discussed and researched the merits of each design regarding ease of manufacturing, stability and handling during flight, and the speed with which the cargo could be loaded and unloaded during competition.

1.1.3 Results

After consideration of the above design parameters, the team concluded that the conventional aircraft was the most efficient and easily built design. The calculated RAC for this design was found to be 13.73, the lowest among the three concepts. The calculated value of MEW was 21.68 pounds, almost seven pounds lighter than the pod-boom concept. The flying wing concept was found to be the lightest, however the one pound decrease in weight was judged by the design team to be offset by complexity of construction, and difficulty of in-flight maneuvering. Ease of cargo handling was deemed to be best for the conventional aircraft concept.

1.2 PRELIMINARY DESIGN

During the preliminary design phase, analysis was conducted to more closely determine values for the specific design parameters of the chosen concept. Sizing trade, wing loading, and power required studies were carried out to compare the performance of the concept aircraft using varying design parameter values. This analysis determined the team's choice of a final aircraft configuration.

1.2.1 Design Alternatives

The design parameters determined during this phase of the process were fuselage size, gross weight, cargo configuration, wingspan and reference area, airfoil selection, and initial motor selection. The figures of merit used to analyze these qualities included wing loading, thrust-to-weight ratio, and power loading.

1.2.2 Analytical Methods

Materials samples were purchased and analyzed to determine their density in order to better estimate the aircraft's empty weight at the size indicated by payload considerations. A spread sheet program was then constructed to calculate and plot power required versus velocity curves to determine optimal wing loading at the assumed mission airspeed. The wing area and span were then chosen to meet this wing loading requirement. A high lift airfoil was chosen through research of airfoil specifications with an emphasis on obtaining a high maximum lift to drag ratio to reduce motor power requirements. After the airfoil was chosen, a planimeter was used to obtain a precise calculation of the wing's area and volume for use in the center of gravity

estimation chart. A motor was then chosen to meet this requirement with the assistance of the "Motocalc" program for R/C aircraft, available on the internet.

1.2.3 Results

The configuration arrived at through the above methods consisted of an eight foot wingspan with a reference area of eight square feet using a Selig 1223 airfoil, a full-width cross section fuselage length of four feet, a cargo configuration in the main fuselage two softballs across and twelve deep, an empty take-off weight of 16.5 pounds, and powered by an Astro Cobalt 60 motor. The wings and fuselage was to be constructed of pre-preg fiberglass composite over foam.

1.3 DETAIL DESIGN

In this phase of design, the team sought to refine their design through further analysis of the chosen parameters. The process included sizing of the horizontal and vertical tails, control surface sizing, takeoff and landing analysis, estimation of the aircraft's drag coefficient and center of gravity, landing gear selection, determination of battery power requirement, structural analysis of the wing and fuselage, and landing gear selection. These analyses determined the final configuration of the aircraft, which was then drawn in detail to facilitate construction.

1.3.1 Design Alternatives

The design team considered alternatives regarding control surface configuration, takeoff and landing distance and their relationship to the need for a braking system, and landing gear configuration. Much analysis was required to determine the final placement of the wing and components on the fuselage, for location of the center of gravity to ensure the aircraft's stability. Determination of the required battery power was crucial to the weights and balances chart as well as to the performance of the motor and servos. Two design concepts for the configuration of the fuselage structure were analyzed; a wooden spar reinforced cylinder of fiberglass composite with a wooden floor and bulkheads, and a molded fiberglass composite cylinder including a fiberglass encased foam base.

1.3.2 Analytical Methods

To analyze control surface sizing and configuration, a stability derivative spreadsheet was constructed and historical values were considered. Takeoff and landing distance were calculated using empirical equations with historical values for rolling ground resistance, rotation time, and takeoff segment velocities. "Tip-back angle" analysis was performed to determine landing gear placement and configuration. A weights and balances spreadsheet was written by the team to determine the location of the center of gravity and finalize component placement. Performance analysis was done through spreadsheet calculation of maximum climb velocity, maximum angular

velocity, load factor at maximum angular velocity, stall velocity, minimum turn radius and velocity at minimum turn radius. Capable Computing, Inc,'s Motocalc program was used to help determine battery power required at the design airspeed, and to select propeller size and pitch. Bending and buckling loads on the wing and two fuselage designs were calculated using finite element modeling. Modeling of the final aircraft design was completed using PTC's Pro-Engineer computer aided drafting software.

1.3.3 Results

The detail design phase resulted in a final aircraft configuration. Analysis confirmed the need for a braking system, in order to meet landing distance requirements. The optimal landing gear configuration was deemed to be a tricycle arrangement, with the main gear located to the rear of the airfoil. The tail design reflected the use of a flat plate configuration, as opposed to the NACA 0012 airfoil originally chosen, to facilitate ease and speed of construction. The lightest configuration for the fuselage (foam/fiberglass) was found to have more than adequate strength when subjected to mission airloads and also to the 2.5 g load factor requirement specified in the competition rules. Performance analysis suggested that the aircraft configuration should posses good handling qualities.

2. MANAGEMENT SUMMARY

The nine member Terp Flyer design team was divided into three groups; Aerodynamics, Structures and Configuration, and Propulsion and Electrical Systems, as depicted in Table 2.1.

Aerodynamics	Propulsion and Electrical Systems	Structures
Constance Lorsong	Jason Sexton	Christopher Curtis
Stephen Madison	Evandro Gurgel Do Amarel Valente	Michelle Dupee
Sean Roark	Woody Ruszala	Timothy Gaur
Project Manager: Stephen Madison Secretary: Michelle Dupee Finances and Procurement: Sean Roark		

Table 2.1

The team members met bi-weekly to discuss progress and participate in group design sessions, where problems and issues were discussed and solved by all group members. In addition, the three specialized design groups met as needed to conduct research and analysis. The entire

team met weekly with Faculty Advisor, Professor J. Gordon Leishman, to seek his input on design issues, keep him updated as to the project's progress, and request his assistance with administrative issues.

The Project Manager was responsible for coordinating the group's efforts and acting as a liaison with faculty members, as well as monitoring the team's chronological progress. The Project Secretary kept minutes at each meeting, prepared a timeline for the group's progress, and handled numerous administrative duties, such as booking hotel reservations and obtaining financial backing. The Finances and Procurement officer prepared a budget for the project, and coordinated the procurement of materials with Aerospace Engineering department staff. Other members of the group assisted with these duties as their expertise and knowledge would allow. Table 2.2 chart's the group's actual chronological progress, as compared to the original timeline.

Table 2.2

Terp Flyer Timeline		
Task:	Expected Date of Completion	Actual Date of Completion
Finish initial concept ideas	8-Nov-01	9-Nov-01
Reducing of number of initial designs	10-Nov-01	12-Nov-01
Initial Design Selection	13-Nov-01	19-Nov-01
Initial Sizing Complete	15-Nov-01	21-Nov-01
Start Pro E drawings	17-Nov-01	22-Nov-01
Finished with structural analyses	19-Nov-01	26-Nov-01
Complete Preliminary Report	4-Dec-01	3-Jan-02
Begin ordering parts	6-Dec-01	28-Dec-01
Pro E finished	15-Dec-01	5-Feb-02
Start construction	7-Jan-02	13-Feb-02
Wind tunnel testing	20-Feb-02	**Cancelled**
Construction complete	15-Mar-02	Not yet complete
Painting and surfacing	20-Mar-02	Not yet complete
Flight tests	21-Mar-02	Not yet complete

3. CONCEPTUAL DESIGN

The figures of merit (FOMs) considered by the design team in the phase were RAC, MEW, ease of construction, and feasibility of translating the theoretical design into reality. RAC was considered an important FOM because it gauges the financial efficiency of the design, as well as its chance for a favorable score in competition. The group considered MEW an important FOM because it could be used as a comparable measure of the aircraft's attainable speed and power needs. Ease of construction was considered important to the team's analysis since the time constraints involved in the construction phase of the competition are great, considering this team's relative lack of experience with composite construction. Feasibility was considered the by the team to be the most important FOM at this stage of design, since the choice of concept at this stage could determine whether or not the final aircraft configuration was capable of flight.

Assumptions made during this phase of design were related to area and volume calculations used in estimating MEW. The empirical equations used in the horizontal and vertical tail volume calculations involved assumption of 'typical values' for the volume coefficients based on general aviation aircraft. These values were considered by the design team to be the closest available approximations to those of R/C aircraft.

3.1 CONCEPT I

FLYING WING
Rated Aircraft Cost: 35.76
Manufacturer's Empty Weight: 20.6

The first concept investigated by the Terp Flyer Team was the flying wing. The team's goal was to have the wing function as the main source of lift, along with the integrated fuselage to achieve a span-loading effect, with a tail extended aft of the aircraft. The top portion of the flying wing would be faired so that the softballs could be loaded and removed. A flying wing would allow for a shorter overall length simply because the payload could be loaded as a 6 X 4 configuration.

Concerns were raised regarding this concept early in the investigative process, based on the overall structure and difficulty associated with flight controls for a flying wing. Structural concerns evolved from the complex shape of the vehicle. Flying wings must have tapered leading edges to ensure the center of gravity (c. g.) is behind the center of pressure to ensure stability. This is a typical guideline for most aircraft as it allows for a stable break at stall and will allow the aircraft to return to normal attitude. Also, adding the payload would add substantial weight to the aircraft. To ensure the payload would not alter the c. g., the payload would need to be added in a very even distribution. The need for such a precise c. g. location would introduce much uncertainty over the flight worthiness of the aircraft in its final inception.

Because the rules for the competition specify that the softballs may not be loaded in a staggered configuration, and the flying wing fuselage must be swept for aerodynamic reasons, it would be quite difficult to design the cargo compartment. Also, the tapered leading edge would make the simple flat-wrapped fuselage construction technique impractical. Consequently, the aircraft would be almost entirely a composite structure, increasing both cost and construction difficulty considerably.

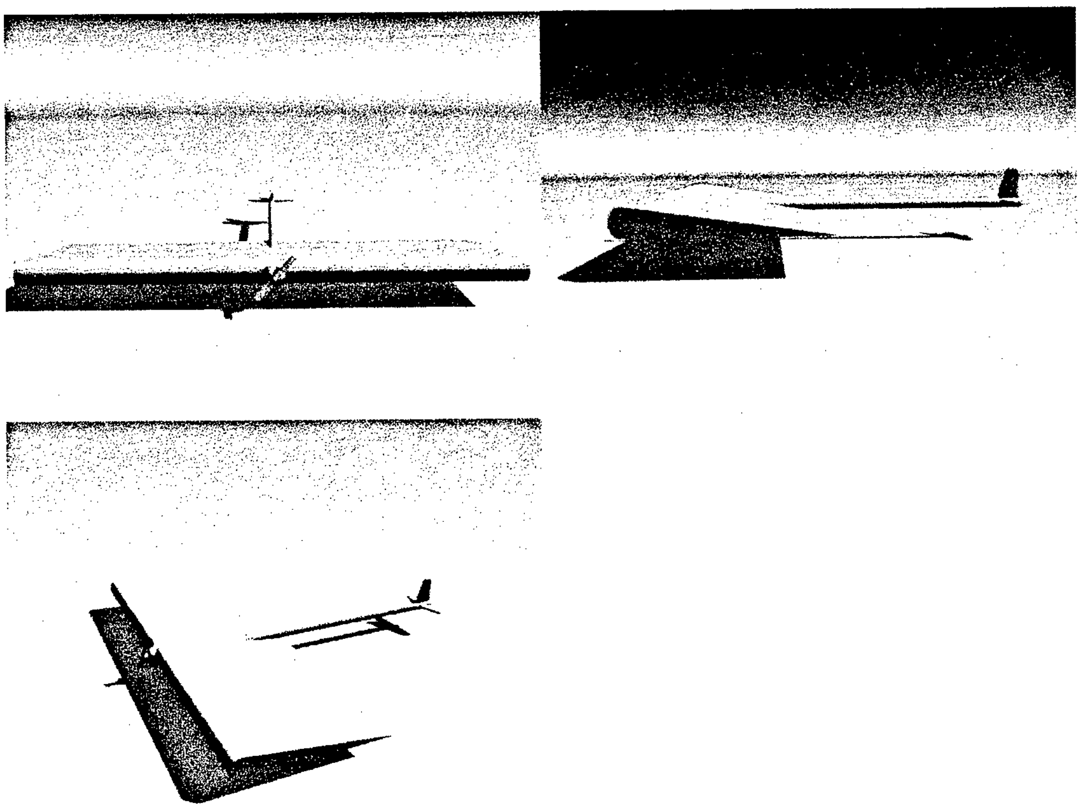


Figure 3.1

Aerodynamic concerns arose with the thickness of the airfoil, since the fuselage would need to have a minimum of a four-inch thickness (the approximate thickness of one softball). Because there is a need for multiple rows of softballs to allow for the maximum payload, the airfoil was calculated to need a nine-inch thickness to retain its aerodynamic cross section shape. At the low Reynold's number flight regime considered in this mission, sustaining lift at this airfoil thickness would be difficult. In addition, such a thick airfoil invoked concern about flow separation, even at these low speeds, as illustrated in figure 3.2. In addition, issues with drag include the large frontal cross-sectional area of the aircraft, since there are competition limitations on power plants to overcome such drag. Although second and third motors are allowed, the weighted cost for the

use of a second motor is prohibitive. Inevitably, the overall structure would lead to a rather large aircraft and would most likely necessitate a second motor, which would heavily penalize the design by increasing the RAC.

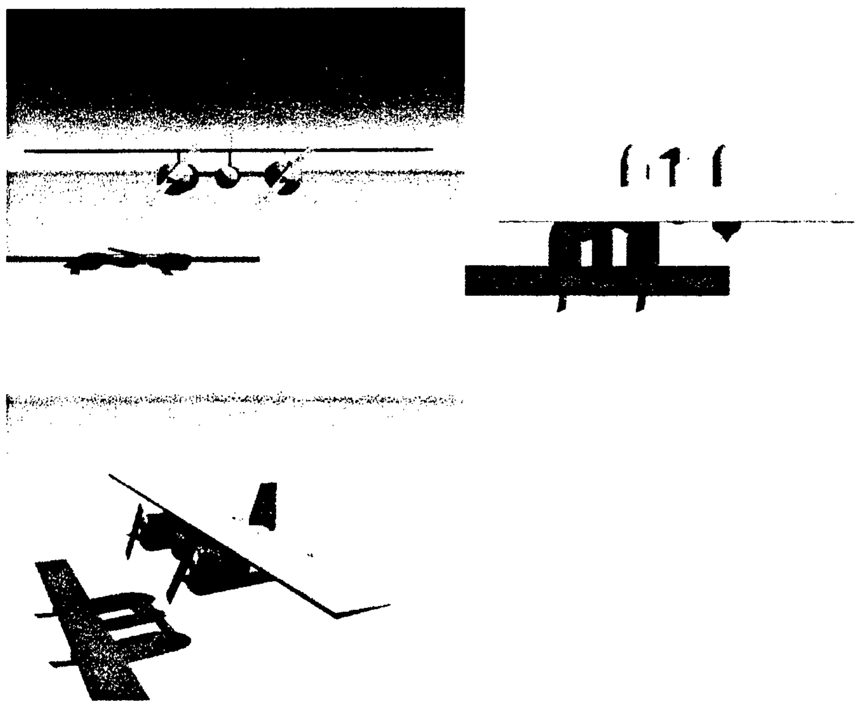
3.2 CONCEPT II

Pod-Boom Aircraft

Rated Aircraft Cost: 13.90 (single motor) / 16.05 (dual motor)

Manufacturers Empty Weight: 27.51 (single motor) / 28.13 (dual motor)

The second concept the Terp Flyer team considered was a Pod-Boom design. The softballs



would be carried in two separate fuselages, or pods, in two rows of six softballs per pod, for a total of twenty-four. In the center would be a central pod, enclosing the electronics of the aircraft—the motor, the batteries, and the communications equipment. The main wing would lie above the three fuselages, and would be raised slightly from the fuselages to limit interference drag associated with typical wing-fuselage joints. The main wing could be situated in between the fuselages and act as a structural element; however, this would vastly increase the size and complexity of the booms, nullifying their advantage of simplicity of manufacturing. To account for the main wing lying above the three fuselages, two small airfoil shapes would connect the booms and act as structural elements for the aircraft.

A tail lower than the main wing is highly desirable for aircraft stall performance, since the tail is out of the stalled wake of the fuselage, thus delaying the stall event for the tail . Based on this

concept, the tail structural element would also act as the elevator for the pod and boom design. This would eliminate the need to build another structure for a horizontal elevator. Finally, the vertical tail would protrude from the center fuselage.

The Pod and Boom design has numerous advantages for the objectives of this competition, but also has some serious drawbacks. One of the advantages of the pod configuration is that, with a duel fuselage, the overall length of the aircraft can be considerably less as compared to a standard fuselage configuration. Also, the separation distance of the two main booms would be ideal for storage of the landing gear within the booms. This would limit parasite drag that would result from a single fuselage configuration, where the landing gear protrudes into the airflow.

Another advantage for this conceptual design is the fact that both main fuselages, the center "control" pod, and the main wing, could all be easily manufactured using flat-wrapping. The RAC for this vehicle was calculated to be 13.90 for a single motor and 16.05 dual motor design. The analysis of this configuration was performed with two motors, in the event that one motor would not perform well given the uniqueness of the design.

On the downside, this type of aircraft is more complex in many ways than a conventional aircraft. The pod-boom concept is unconventional, and rather experimental. From a technical standpoint, the multiple boom design would have problems associated with loading and unloading the payload because of the location of the main wing. Therefore, a unique fairing design would need to be developed, which could take time and energy away from optimizing flying performance—a clear goal set by the group. Another concern is the fact that the weight of the aircraft is not evenly distributed about the roll axis of the aircraft. The design team was concerned that this distribution of weight would hinder the turning performance of the aircraft, and that correctly analyzing this issue would be very difficult.

3.3 CONCEPT III

CONVENTIONAL AIRCRAFT
Rated Aircraft Cost: 13.73
Manufacturers Empty Weight: 21.68

The third design concept considered during the conceptual phase of the Terp Flyer was a conventional aircraft with a single fuselage, a main wing, and a tail. The conventional aircraft would have a standard tail with horizontal and vertical control surfaces. The cargo would be carried within the fuselage. A conceptual view is provided in figure 3.4.

A fairing or structural door on the upper surface or sides of the main fuselage would allow for easy cargo handling. Because the softballs are required to be in a minimum of two adjacent

rows, the fuselage would have an elliptical cross section, so that the frontal area of the vehicle is decreased to reduce drag.

Since countless aircraft have been designed and built in this standard style, there is a wealth of information available for research and reference use. This concept has historically had great success in Design/Build/Fly competition, leading the design team to conclude that the concept has a great chance for success in flight. In addition, the aircraft is simple in design and would be easy to construct, fairly lightweight, and have good handling qualities.

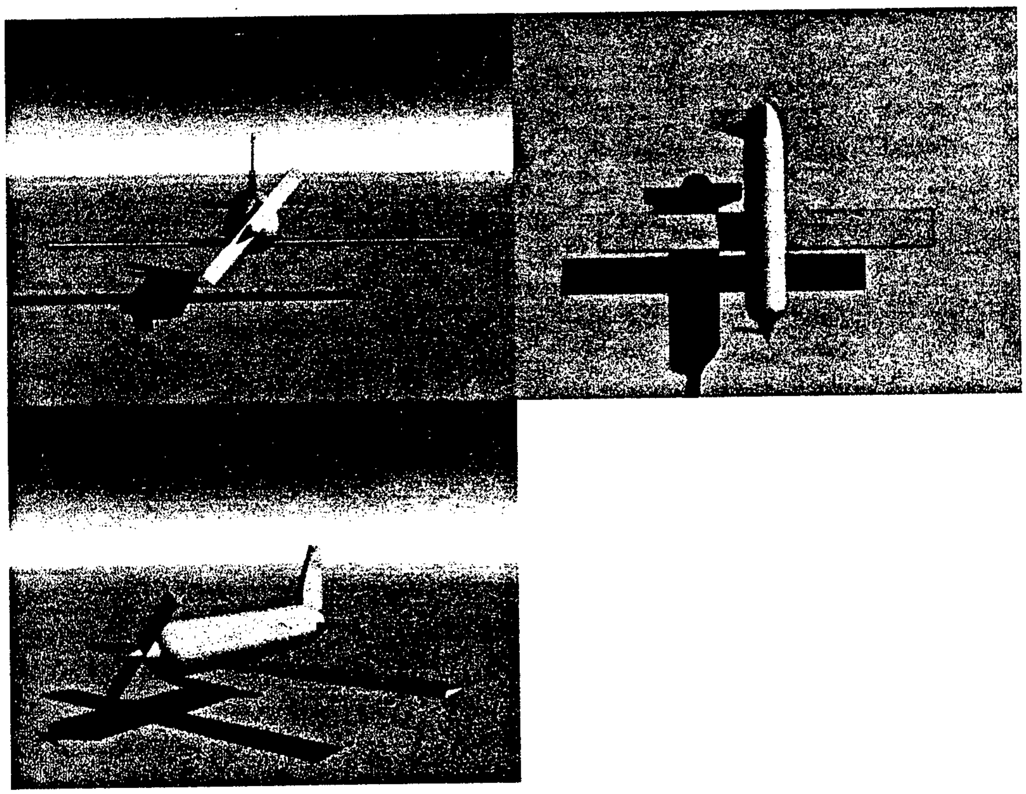


Figure 3.3

3.4 RESULTS

Having considered the pros and cons of each design and evaluated the FOMs for each (see table 3.1), the design team chose the conventional aircraft configuration for further development. The group felt that the simplicity of a conventional aircraft, as well as the historical success of such a configuration, provided the best probability for a favorable competition outcome.

Table 3.1

	Figures Of Merit:			
[illegible]	[illegible]	[illegible]	Ease of Construction	Concept Feasibility
Flying Wing	35.76	20.68	2	2
Pod-Boom	13.9	27.51	3	2
Dual Engine Pod-Boom	16.05	28.13	3	2
Conventional	13.73	21.68	4	5

Note: Ease of Construction and Concept Feasibility are rated from 1 to 5, with 5 being the most desirable

4. PRELIMINARY DESIGN

The team's goal for the preliminary design phase was to conduct sufficient analysis and research to arrive at a more complete and focused picture of the final configuration of the aircraft. The team was able to select an appropriate airfoil, motor, propeller size, wingspan, and fuselage length and structure. A more accurate takeoff weight was found with the use of a spreadsheet program that accounted for the weight and moment arm of each component. The parameters investigated include fuselage size, aspect ratio, power required, and wing loading. The figures of merit used to analyze these parameters include thrust to weight ratio, power loading, and lift to drag ratio (L/D).

4.1 ANALYSIS

4.1.1 Fuselage Sizing

The design team made the decision to attempt to carry the maximum cargo load, since historical research showed that all previous successful designs had done so. Because the cargo configuration was the driving factor behind the sizing of the fuselage, the team began this design phase by analyzing the feasibility of the cargo configuration mentioned in the conceptual design section (two balls wide, twelve deep.) This configuration was found to require an eight-inch wide fuselage, four feet in length. Through research, the structures team found that a fuselage with a circular cross section would be easier to construct than the elliptical one mentioned in the earlier concept description. The length of the fuselage was determined through calculation by use of an empirical equation based on empty weight, assuming single engine general aviation constants, as well as through analysis of the length needed to house cargo and components.

4.1.2 Gross Weight

The structures team wrote a spreadsheet program that would be useful for determination of center of gravity location, as well as for closely determining the empty weight of the fuselage. A wing reference area of ten square feet was assumed from historical data (and was later altered) for use in this calculation. The wing and tail volumes were found at first by drafting cross sections to scale, estimating the area, and subsequently integrating to determine the area under a plotted

curve of area versus chordwise location. The value of this integral determined wing and tail volume, when multiplied by the chosen span. After the aerodynamics group finalized the airfoil choice, this process was repeated using a planimeter, for increased accuracy.

Samples of foam, balsa wood, and plywood were purchased, weighed and measured to determine their density. The weight and thickness of composite skins was researched and estimated. Pre-impregnated fiberglass weave over foam was chosen as the material for wing and fuselage construction, in order to estimate the weight of these structures. Weights for servos, motor, payload and landing gear were estimated through research and use of historical values. These quantities were summed using the spreadsheet, and the empty weight was estimated as 16.5 pounds, considerably lighter than the conceptual phase calculation had suggested.

4.1.3 Airfoil Selection

The group decided that it was in the best interest of the mission to choose an airfoil that provided maximum lift. Through research, the aerodynamics group discovered a very high-lift airfoil that had been designed for R/C applications; its designation is Selig 1223. Through analysis of the drag polar, the group found that the airfoil had a maximum lift coefficient of 1.7, a much higher value than that of the other airfoils considered.

The aerodynamics group expressed concern that the highly cambered airfoil, which has a very thin trailing edge, might be difficult to construct. Through research, the group was able to determine that the airfoil had been used successfully in similar applications by teams with similar levels of experience. In light of this information, the decision was made to use the Selig 1223 airfoil for this application.

4.1.4 Wingspan and Reference Area

A spreadsheet was written by the team to calculate the optimal wingspan at the cruise velocity of the aircraft. It was determined from a mission profile analysis and optimization study that the cruise velocity of the aircraft would need to be thirty miles per hour in order to complete the contest mission in the required ten minute time period. Wing loading was held at a constant value, and expressed as a function of wingspan and chord, and subsequently used to generate power required versus velocity curves. An example is shown in figure 4.1. These curves were used to graphically determine the wing reference area needed at cruise velocity to produce the desired wing loading.

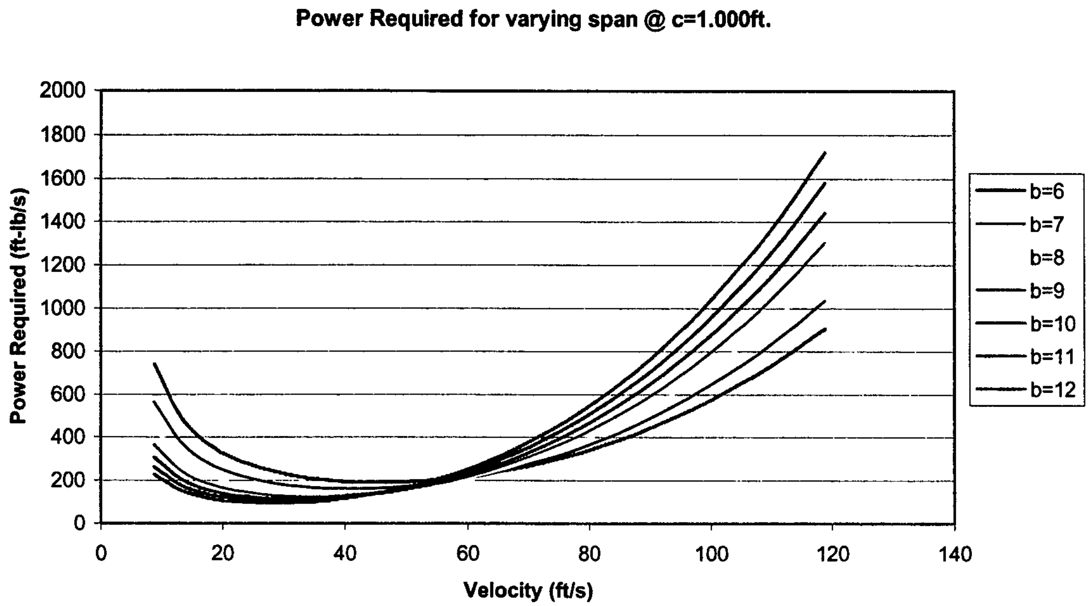
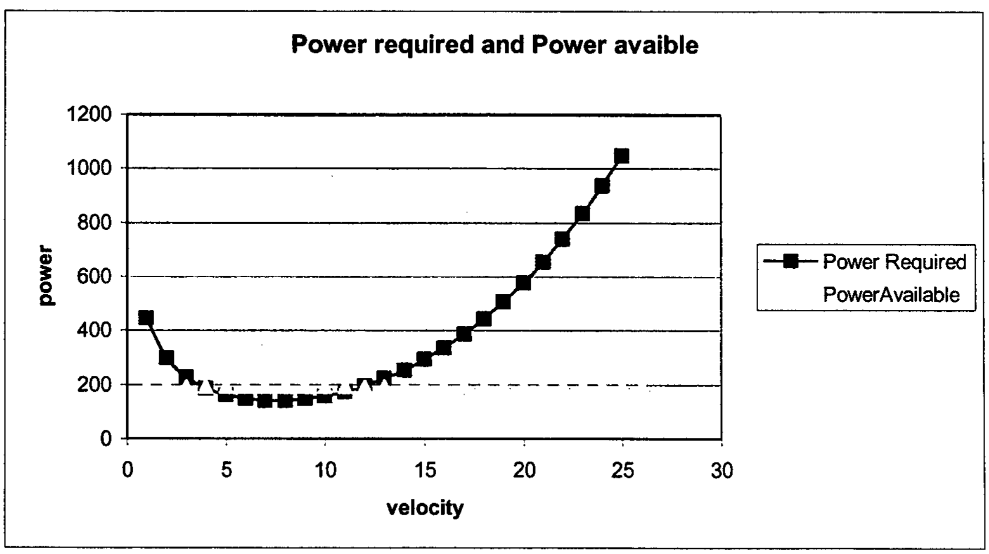


Figure 4.1

4.1.5 Wing and Power Loading



4.1.6 Motor Selection

Motor selection for the airplane was a difficult process based on our demanding requirements. The design requirements of the contest and the configuration of the aircraft design determined that only Astro Flight and Graupner brushed motors would be adequate to power the airplane. DC brushed motors were selected due to the reliability of this type of motor and the scale of the

aircraft. Astro Flight and Graupner Motor are respected names and are known for their development of DC motors. Since Astro Flight was founded by Bob Boucher and Roland Boucher in 1969, the company has set many world records and powered many first-in-flight electric airplane models. Graupner Motors also has a long history of producing high performance motors for remote control cars, airplanes and boats.

The team used the MotoCalc software program to aid in selection of the motor. MotoCalc predicts the performance of an electric remote controlled aircraft based on the motor, battery, propeller and gear ratio (all of which are input by the user). In addition, Volts, Amps, input and output Watts, efficiency, temperature, RPM, thrust, pitch speed, and run time are predicted for the input configuration. The program will also predict the handling characteristics and performance of an aircraft for all segments of a mission. These values were used as a check against those that were calculated theoretically by the team.

The team selected a motor and battery cell combination that will power the Terp Flyer according to the mission requirements. An Astro 60 motor in tractor configuration with a 16x10 propeller, powered by twenty-five battery cells, was chosen. This gave a maximum velocity of thirty-five mph (51.3 ft/sec) and a battery life of over fourteen minutes at full throttle carrying full payload. This configuration should allow the aircraft eleven minutes time to complete the mission.

4.2 RESULTS

The preliminary design phase resulted in the aircraft configuration outlined in table 4.1 below.

Table 4.1

Terp Flyer Preliminary Configuration	
Airfoil	Selig 1223
Wingspan	8 feet
Chord	1 foot
Fuselage length	4 feet
Fuselge width	8 inches
Empty weight	16.45 lbs.
Cruise velocity	30 mph

5. DETAIL DESIGN

Through the detail design process, the Terp Flyer team arrived at a final configuration, detailed three-dimensional drawings, and a manufacturing plan for its competition entry aircraft.

5.1 ANALYSIS

5.1.1 Handling Performance

Table 5.1

Parameters						
g, ft/s^2	T, lbf	C_{DO}	K	L/D	Z_{unload}	Z_{load}
32.2	7.8625	0.0742	0.049086	8.28519563	2.0974	2.231005

Velocity Min. turn Radius

V_{Rmin} **(ft/s)**

19.88702304

Min. Turn Radius

R_{MIN} **(ft)**

12.72330384

Max Climb Velocity

$V_{(R/C)max\ unload}$ $V_{(R/C)max\ load}$

24.4994865 27.503

Max. Angular velocity

ω_{max} **(rad/s)**

1.968489

Load Factor at max. ang. Velocity

$n_{\omega max}$

2.581435

The aircraft's minimum drag coefficient was found using the component build-up method; this value was used to calculate the handling qualities of the aircraft by constructing a spreadsheet to generate the aircraft's, maximum climb velocity, maximum angular velocity, load factor at maximum angular velocity, stall velocity, minimum turn radius and velocity at minimum turn radius. These results, presented below in table 5.1 and figure 5.1, were used to predict the mission performance of the aircraft.

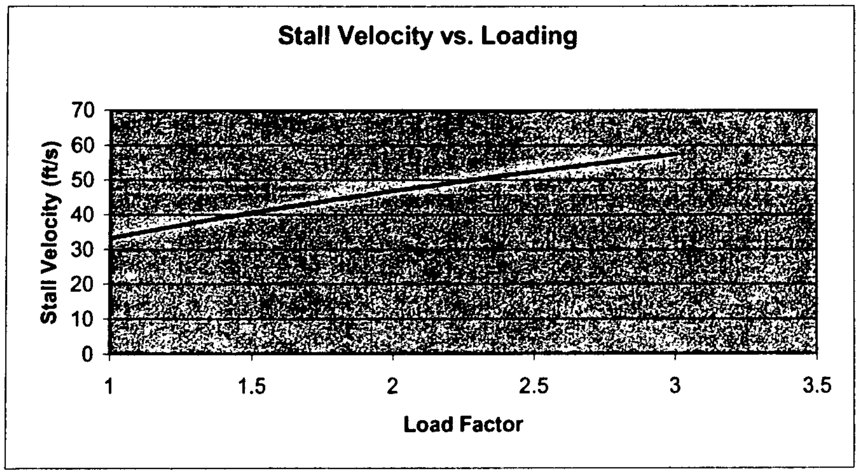


Figure 5.1

Takeoff and landing performance was analyzed using a spreadsheet program. Takeoff and landing distance were plotted versus wing loading, as shown in figures 5.2 and 5.3. Takeoff and landing speeds and distances were found to be adequate to complete the mission with a competitive time score.

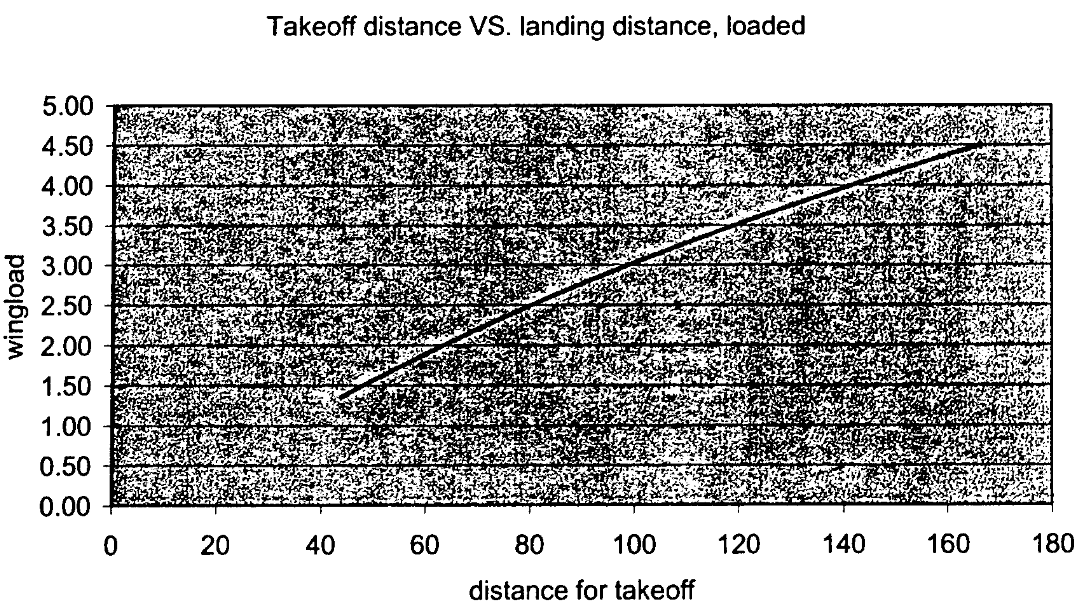


Figure 5.2

Landing distance VS. wing loading, unloaded

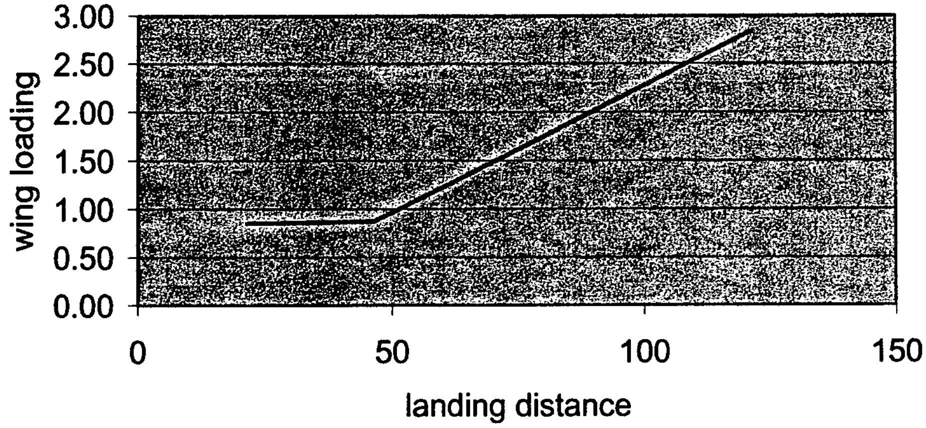


Figure 5.3

5.1.2 Estimated Mission Performance

Using a spreadsheet and performance data, the team was able to predict the time score of the final aircraft configuration, as shown in table 5.2.

layout of track	distance (feet)			ft/s	MPH	
straight aways x2	1000.0		Min velocity	42	28.84	
360 loop	961.8		Flying velocity	56	38.18	
turn	924.3					
total unloaded sortie	14464.4					
total loaded sortie	5848.6					
total distance	20313.0					
Unloaded						
Full track				**Oval**		
	distance (feet)	time (seconds)			distance (feet)	time (seconds)
take off	70.0	1.25			70.0	1.25
1st part upwind	430.0	7.68			430.0	7.68
turn	924.3	16.51			924.3	16.51
1st part downwind	500.0	8.93			500.0	8.93
360 loop	961.8	17.18			no 360 turn	
2nd part downwind	500.0	8.93			500.0	8.93
turn	924.3	16.51			924.3	16.51
2nd part upwind	400.0	7.14			400.0	7.14
landing	100.0	1.79			100.0	1.79
sub-total		**85.90**				**68.73**
Loaded						
Full track						
	distance	time				
take off	100.0	1.79				
1st part upwind	400.0	7.14				
turn	924.3	16.51				
1st part downwind	500.0	8.93				
360 loop	961.8	17.18				
2nd part downwind	500.0	8.93				
turn	924.3	16.51				
2nd part upwind	350.0	6.25				
landing	150.0	2.68				
sub-total		**85.90**				
total time		**481.05**				
		8.0175	minutes			

5.1.3 Component Selection and Systems Architecture

The systems architecture for the Terp Flyer is broken down into four major components; servos, speed control, power plant, and remote control.

1) Servos

   The primary factors taken to consideration for servos were weight, size and power. Drawing from team members' radio controlled modeling experience, servos were chosen that would provide more than adequate power and torque. Considering the physical size and loads on the control surfaces, the following setup was designed:

   - Ailerons

     Each aileron will be controlled by its own servo. This configuration was chosen based on the distance between the ailerons. A torque rod would be inadequate for the wingspan. Also, the computer radio gives the flexibility to program differential and flaperons.

   - Elevator/Brake

     The servos actuating the elevator will be mounted in the rear of the airplane. The elevator is divided in two halves such that each half is controlled by its own servo. Dual control rod elevator setup (one servo) will not be used in order to maintain open space in the fuselage for the cargo. The braking system will be activated by deflecting the elevator.

   - Rudder/Nose gear

     The rudder will be controlled by one high power servo that is located also in the rear of the fuselage. The nose gear will be also controlled by this servo via a flexible control pushrod, for control of the aircraft during taxiing.

2) Speed Control.

   A solid-state speed control was selected based on contest rules (maximum allowable current draw), motor and battery specifications.

3) Power Plant.

   Multiple iterations of Motocalc Software helped to configure the following setup:

   - Motor

     Tractor Configuration/Nose mounted Astro Cobalt 60.

   - Prop

     Standard 14 in x 6 pitch or 15 in x 6 pitch.

   - Battery

The pack will contain 25 cells of 1.2 Volts/cell & 3000mAmp capacity. The battery cells are to be configured in a lengthwise, end-to-end configuration. To secure the battery, a bed will be pre-cut along the length of the foam keel in the fuselage. This bed provides a desirable mating surface to which the pack will be secured.

4) Remote Control.

The pilot interface with the airplane will be configured such that competition rules are met. Adequate batteries for transmitter and onboard receiver will be included.

- Transmitter

  A programmable radio with advanced mixing controls.

- Receiver

  PCM receiver with fail-safe capabilities.

5.1.4 Structural Analysis

Because the design team did not have access to a CAD software program with Finite Element Modeling (FEM) capability, a simple FEM analysis spreadsheet was constructed to find the longitudinal stress due to pure bending. The two scenarios considered were: 3g loads placed on the wing tips (shown in figure 5.2), and 3g loads applied to the ends of the fuselage main body. The load bearing structure of the wing and fuselage will be a foam core keel wrapped with Cycom 919 "E" fiberglass. The insulating foam's main function will be to hold the shape of the fiberglass and therefore was not considered in the stress calculations. Data points for the S-1223 airfoil and the fuselage, were used to conduct 141 and 59 element models, respectively.

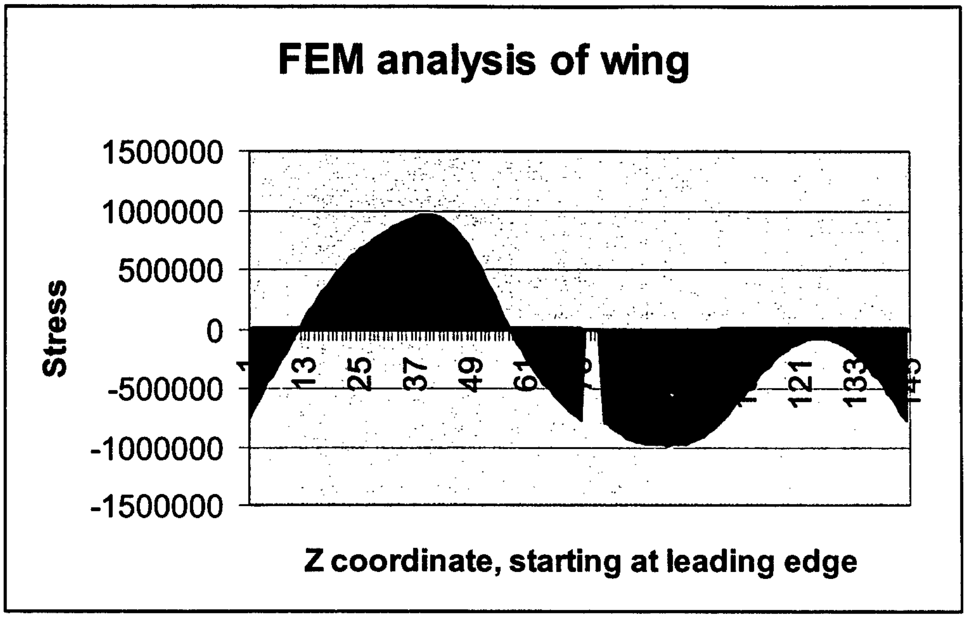


Figure 5.2

5.1.5 Weights and Balances

A weight table (table 5.5) was constructed to find the longitudinal and vertical locations of the c.g. A design requirement for this project is that the longitudinal c.g. will not move when payload is added and removed; therefore it is desirable to keep c.g. location at the midpoint of the cargo bay. The location of the batteries, the most massive internal component, will be adjustable to compensate for any change in design or calculation error. Calculation of the vertical location of the c.g. was needed so that the centerline of the propeller shaft could be properly placed to minimize the pitching moment due to the motor's thrust. The weight table was also used to help determine the pitching moments on the aircraft, and thus, the size of the tail feather surfaces and the incidence of the wing.

All distances in ft. All weights in lbs.	Surface Area	Thickness	Volume	Specific Gravity	Weight	Longitudinal Distance (datum point 1 ft. ahead propellor)	Wt. X Longitudinal Dist.		Vertical Distance From Line Running Through Prop. Shaft	Wt. X Vertical Distance	
Fiberglass Skin Data											
wing	16.80332856	0.001333333	0.022404438	141.68	3.174261	3.833	12.1669416		-0.708333333	-2.248434724	
h-tail	3.415188598	0.000666667	0.002276792	141.68	0.322576	6.333	2.042873473		0.083333333	0.026881329	
v-tail	1.3011	0.000666667	0.0008674	141.68	0.122893	6.333	0.778282838		0.520114943	0.063918606	
fuselage	8.377333333	0.001333333	0.011169778	141.68	1.582534	3.833	6.065853265		-0.25	-0.395633529	
f-cone	0.785	0.000666667	0.000523333	141.68	0.074146	1.333	0.09883644		-0.166666667	-0.012357644	
r-cone	0.785	0.001333333	0.001046667	141.68	0.148292	6.167	0.914515119		-0.166666667	-0.024715289	
							0			0	
							0			0	
Foam Data											
wing			0.5248	1.7	0.89216	3.833	3.41964928		-0.708333333	-0.631946667	
h-tail			0.165975213	1.7	0.282158	6.333	1.786905739		0.083333333	0.023513155	
v-tail			0.098116667	1.7	0.166798	6.333	1.056333845		0.520114943	0.086754306	
fuselage			0.253611111	1.7	0.431139		0			0	
							0			0	
Ribs											
f-bulkhead ring			0.012384259	0.0235	0.000291	1.833333333	0.000533555		-0.25	-7.27575E-05	
r-bulkhead ring			0.012384259	0.0235	0.000291	5.833333333	0.001697676		-0.25	-7.27575E-05	
							0			0	
							0			0	
Structural Supports											
wing supports					0.025	3.833	0.095825		-0.583333333	-0.014583333	
landing gear support			0.003962674	30	0.11888	1.833	0.217907422		-0.583333333	-0.069346788	
tail supports					0.01	6.333	0.06333		0.041666667	0.000416667	
motor mount			0.000651042	172.8	0.1125	1.5	0.16875		0		
										0	
										0	
Electronics											
receiver					0.09375	1.75	0.1640625		0	0	
batteries 1					4.6	4.67	21.482		-0.5	-2.3	
motor					1.5	1.667	2.5005		0	0	
dig. elec. speed control					0.0662	1.667	0.1103554		-0.166666667	-0.011033333	
										0	
Servos											
rudder					0.11	6.333	0.69663		0	0	
elevator					0.11	6.333	0.69663		0	0	
aileron					0.22	3.833	0.84326		-0.5833	-0.128326	
prop					0.01	1	0.01		0	0	
Landing Gear											
main gear+tires					1.475	3.833333333	5.654166667		-0.916666667	-1.352083333	
nose gear+tire					0.7375	1.833333333	1.352083333		-0.916666667	-0.676041667	
breaking system					0.0625	5.833333333	0.364583333		-0.583333333	-0.036458333	
							0			0	
Payload					10	3.833	38.33		-0.25	-2.5	
								Longitudinal C.G.			Vertical C.G.
SUM					[illegible]		[illegible]	3.621606275		[illegible]	[illegible]

5.1.6 Rated Aircraft Cost

Coefficient	Description	Value	Subtotal
A	Manufacturer's Empty Weight	$100	
B	Rated Engine Power Multiplier	$1,500	
C	Manufacturing Cost Multiplier	$20/hour	
MEW	Manufacturer's Empty Weight	Weight of airframe with batteries but not payload	15
REP	Rated Enginer Power	(1+.25(# of engines-1))*Battery Weight	5
MFHR	Manufacturing Man Hours	**MFHR = Σ WBS hours**	
		WBS 1.0 Wings	
		Wingspan = 8 hr/ft	64
		Max Chord = 8 hr/ft	8
		3 hours per control surface	6
		WBS 2.0 Fuselage	
		10 hr/ft maximum length	60
		WBS 3.0 Empenage	
		5 hr/Vertical Surface with no active control	0
		10 hr/Vertical Surface with active control	10
		10 hr/Horizontal Surface	10
		WBS 4.0 Flight Systems	
		5 hr/servo or motor controller	25
		WBS 5.0 Propulsion Systems	
		5 hr/engine	5
		5 hr/propeller or fan	5
		MFHR Total	193
RAC TOTAL	**12.86**		

5.2 RESULTS

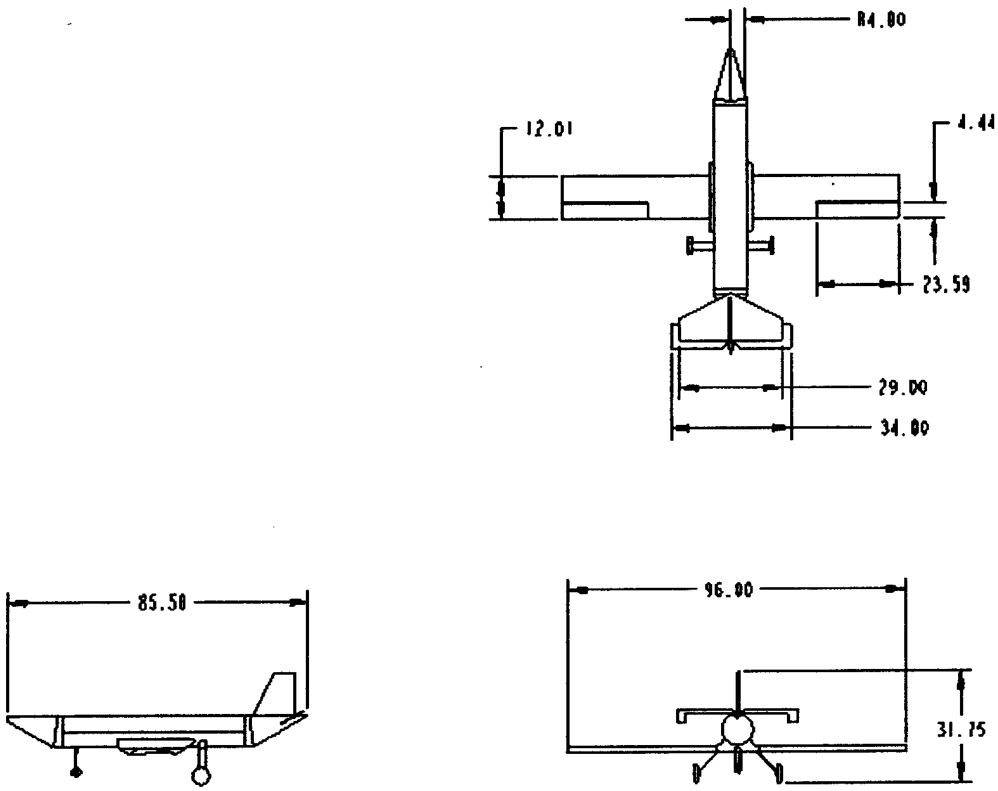
The detail design process produced a refined configuration for the aircraft that, according to the analytical process, would be able to complete the contest mission in an efficient manner. Analysis of design parameters showed that the aircraft's handling and performance qualities should be more than adequate to complete the mission. Structural analysis suggested that the wing and fuselage were more than five times stronger in bending than would be required to meet the mission standards. The RAC analysis done on the design shows it to be an efficient one. The specifications for the final design are listed in table 5.3 below.

Table 5.3

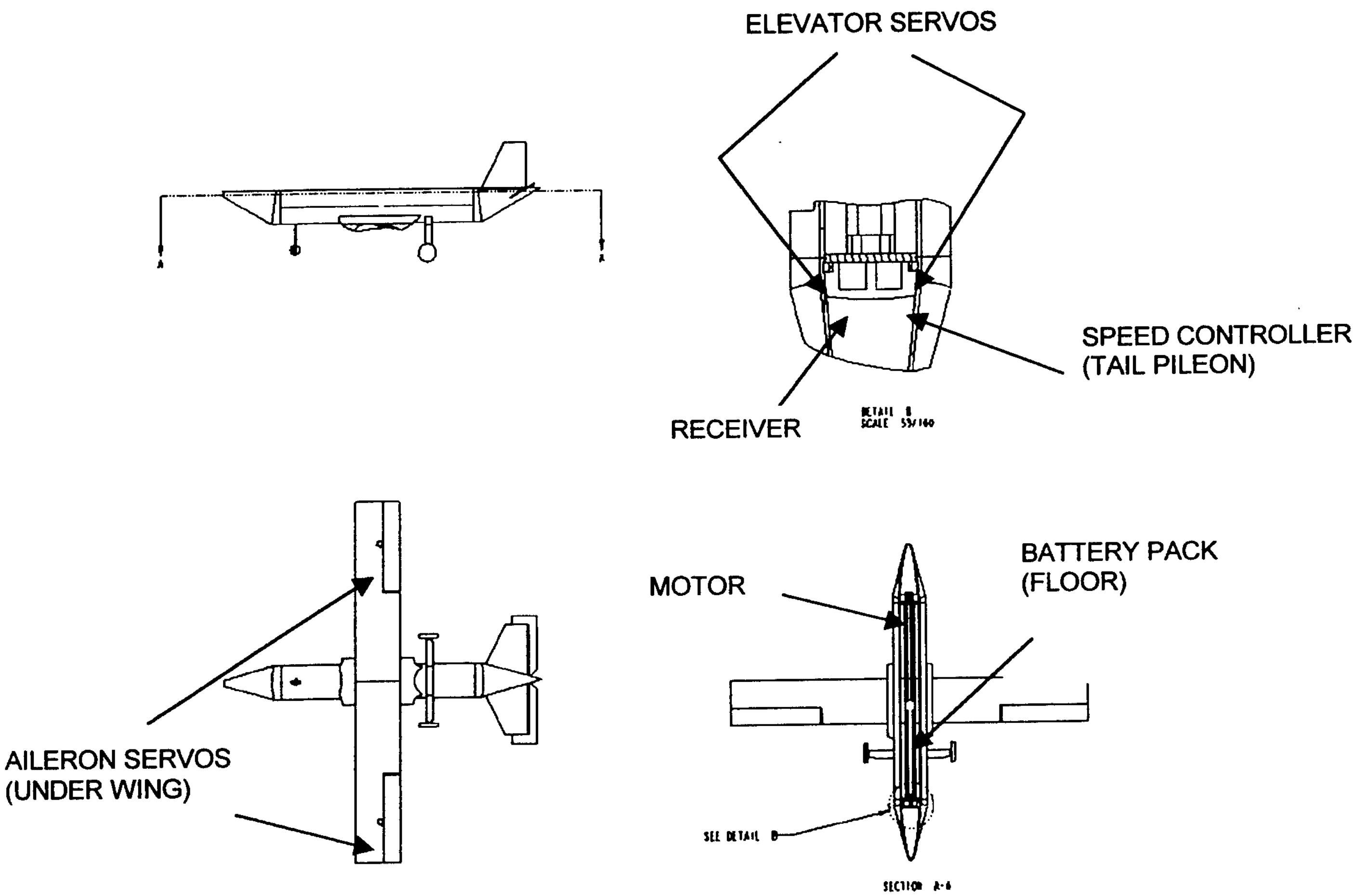
Wing Airfoil	Selig 1223
Tail Surface Airfoil	Flat plate
Wing Span	8 ft.
Horizontal Stabilizer Area	1.56 $ft.^2$
Vertical Tail Area	1.41 $ft.^2$
Wing Chord	1 ft.
Total Aileron Area	0.67 $ft.^2$
Elevator Area	0.56 $ft.^2$
Rudder Area	0.3 $ft.^2$
Wing Incidence	7 degrees
C. G. Location (from nose)	2.82 ft.
Empty Weight	16.5 lbs.
Motor	Astro Cobalt 60

5.3 DRAWINGS

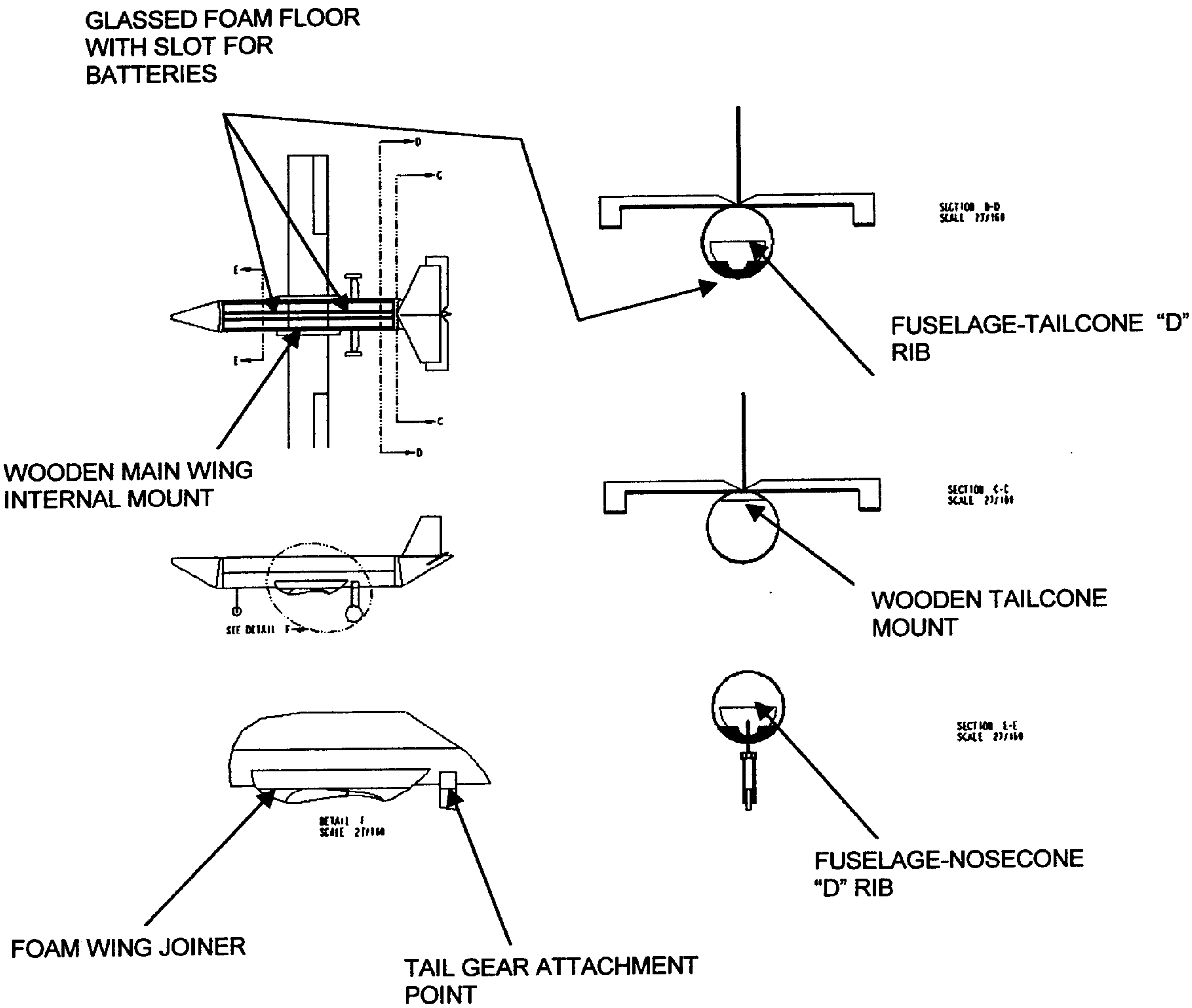
3 View:

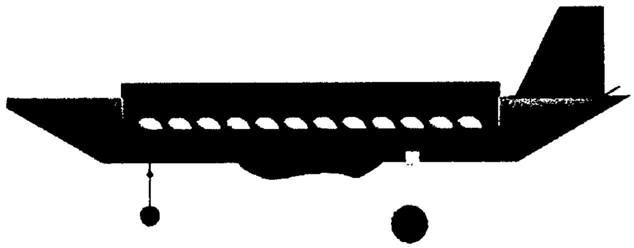


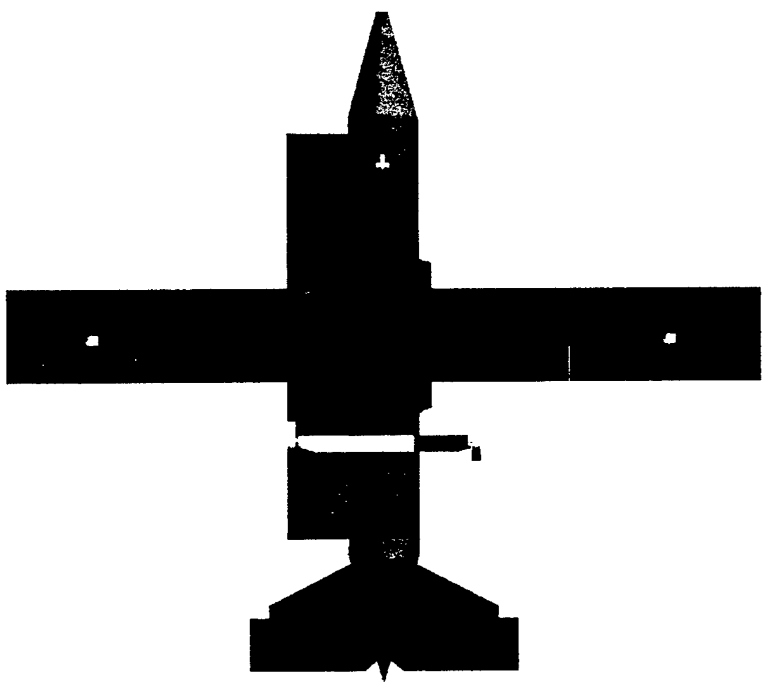
System:

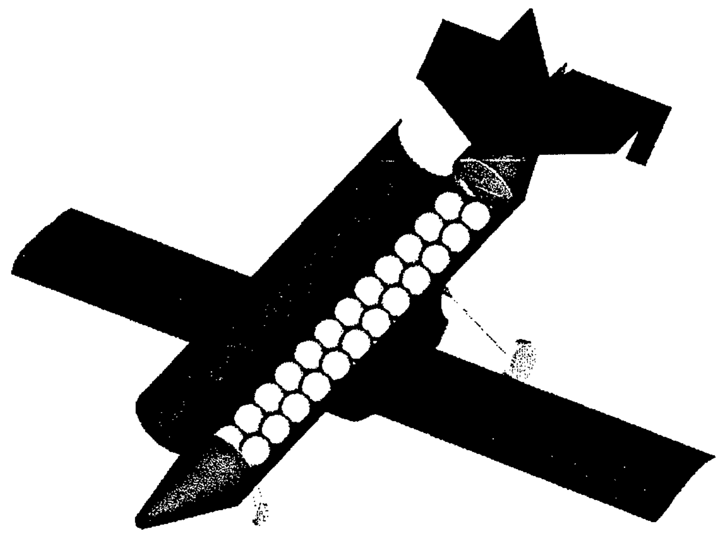

ELEVATOR SERVOS
SPEED CONTROLLER
(TAIL PILEON)
RECEIVER
BATTERY PACK
(FLOOR)
MOTOR
AILERON SERVOS
(UNDER WING)
SEE DETAIL B
SECTION A-A

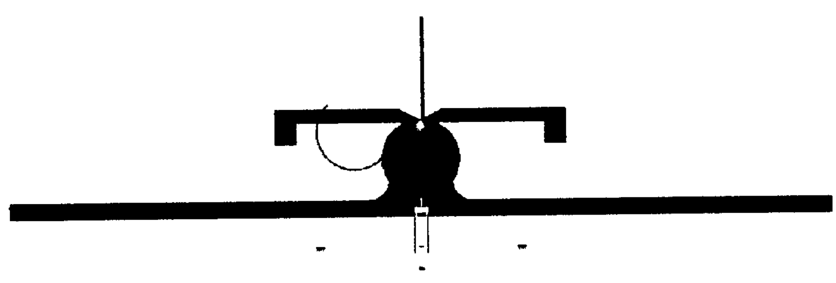
Structure:


GLASSED FOAM FLOOR WITH SLOT FOR BATTERIES
FUSELAGE-TAILCONE "D" RIB
WOODEN MAIN WING INTERNAL MOUNT
WOODEN TAILCONE MOUNT
FUSELAGE-NOSECONE "D" RIB
FOAM WING JOINER
TAIL GEAR ATTACHMENT POINT









6. MANUFACTURING PLAN

6.1 MANUFACTURING PROCESS INVESTIGATION

The design team analyzed three methods of construction: carbon fiber composite over foam, pre-impregnated fiberglass over foam with wooden longerons, and pre-impregnated fiberglass over foam with a foam keel. The latter configuration was chosen, because it was the found to be the least difficult to construct, the least costly, and had the safest production process of the three. A graphical depiction of the FOM rating of the three processes is shown in table 6.2.

Table 6.2

	FOM		
	Cost	Difficulty	Safety
PROCESS			
Carbon fiber over foam	4	5	5
Prepreg fiverglass/foam/wood	3	4	3
Prepreg fiverglass/foam	2	3	2

Note: figures of merit are rated 1 to 5, 1 being most desirable

Table 6.3

Skills Matrix

	Wood working skill	Foam hot wiring	Balsa wood framing	Fiberglass composite lay-up	Metal machining	Electronics wiring	R/C Aircraft Modeling
Number of team members posessing skill	1	0	2	0	2	3	2

6.2 PROCESS PLAN

The manufacturing process is divided in three sections. These sections are acquisition of materials, fabrication and assembly of components. The components of the airplane consist of electronics, wing, empennage and fuselage. The components fall into the manufacturing process as follows:

- Acquisition of Material

  All materials and electronics were purchased "off the shelf."

- Fabrication

  The wing is a foam core sheeted with fiberglass. The cores were cut with a hot wire. Two layers of pre-preg (CYCOM 919 Laminate "E" glass Fiber) were vacuum bagged onto the foam cores and cured in an autoclave. The empennage was manufactured similarly; however, balsa was used in place of foam. Lastly, our cylindrical fuselage was vacuum bagged inside a PVC pipe. For structural integrity, a foam keel runs the bottom length of the fuselage.

- Assembly

  The wing and landing gear were mounted with nylon screws. The empennage glued with Epoxy. Electronics were installed.

The team's manufacturing progress is outlined in table 6.4.

Table 6.4

Task:	expected date of completion	actual date of completion
Wings	Feb. 13-19	Feb. 21-26
Hot wiring	Feb. 13-15	Feb. 21-23
Fiberglassing	Feb. 18-19	Feb. 25-26
Assembly of wing joint	Feb. 14	Feb. 23
Fuselage	Feb. 18-25	Feb. 25-Mar. 4
Cut foam sections	Feb. 18-19	Feb. 25-27
Fiberglassing	Feb. 20-21	Feb. 28-Mar. 2
Construction of joining members	Feb. 25	Mar. 4
Wing and landing gear mount	Feb. 25	Mar. 4
Motormount	Feb. 25	Mar. 4
Tail section mount	Feb. 25	Mar. 4
Landing gear	Feb. 18	Feb. 15
Assembly of breaking system	Feb. 18	Feb. 15
Testing of landing gear	Feb. 18	Feb. 15
Control systems	Feb. 20-21	Feb. 28-Mar. 1
Servo installation	Feb. 20-21	Feb. 28-Mar. 1
Tail section	Feb. 13-18	Feb. 28-Mar. 5
Cutting of plywood	Feb. 13-15	Feb. 28-Mar. 2
Fiberglassing	Feb. 15	Mar. 4
Joining of vertical and horizontal tails	Feb. 18	Mar. 5

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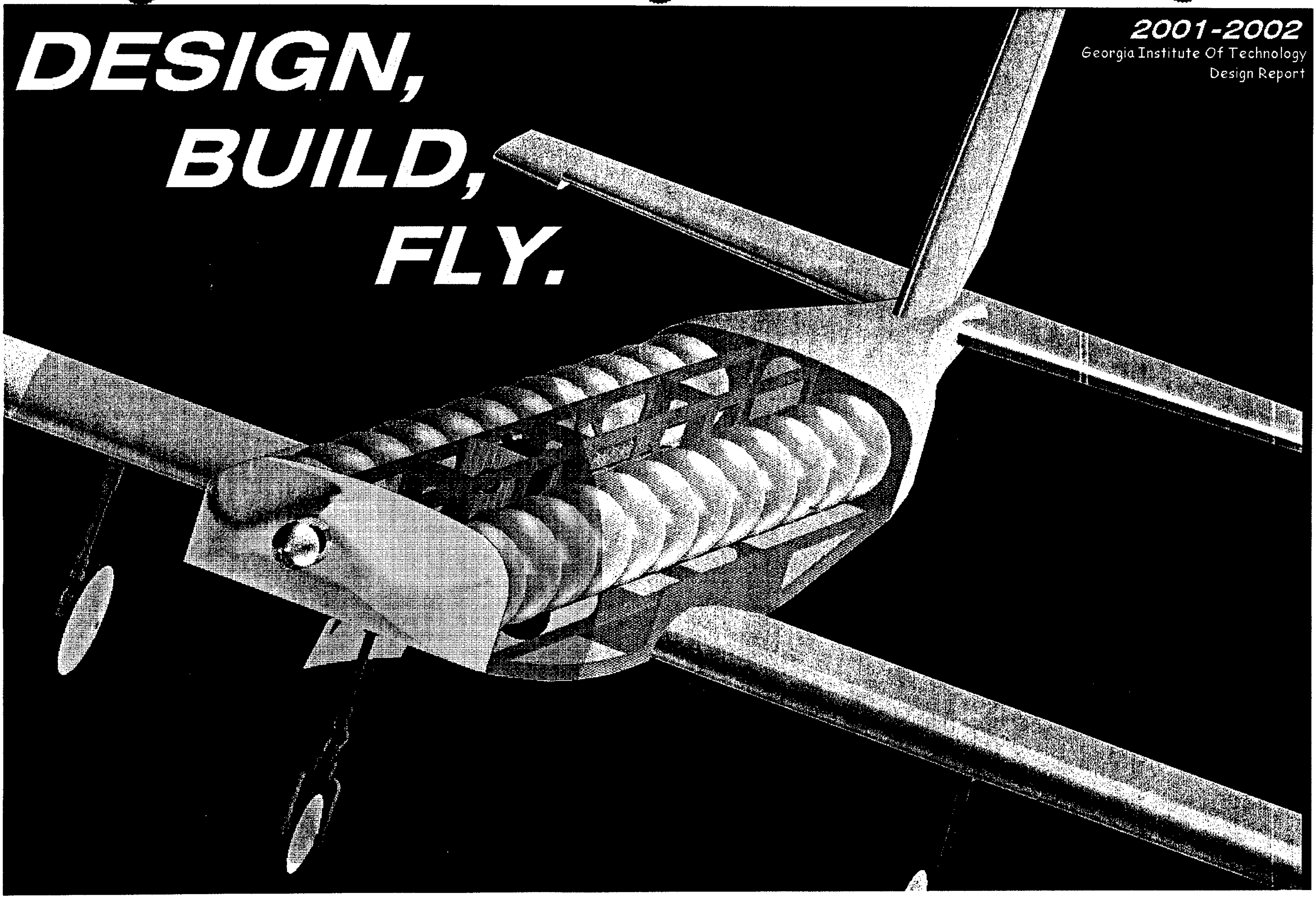
DESIGN,
BUILD,
FLY.
2001-2002
Georgia Institute Of Technology
Design Report

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1 Executive Summary

The American Institute of Aeronautics and Astronautics (AIAA) in coordination with Cessna Aircraft and the Office of Naval Research sponsor the annual Student Design, Build, Fly Competition to "...encourage innovation and maintain a fresh design challenge." This Academy of Model Aeronautics (AMA) sanctioned event involves drafting and building an unmanned remotely operated aircraft to fly a designated flight course while carrying heavy payload and within a specified time limit. The total score for a school's entry is based on three components: the design report's score, manufacturing cost model score, and total flight score. This report summarizes the development of Georgia Tech's 2001-2002 Design, Build, Fly contest entry.

When the rules were released in June of 2001, several graduate students working for the Aerospace Systems Design Laboratory (ASDL) at Georgia Tech consulted with professors and research engineers to formulate a plan of action for the coming contest season. The rules were analyzed in detail for major implications and the students developed a preliminary timeline for major developments in the project. This preliminary plan anticipated that an aircraft would be designed as early as possible, be prototyped early enough to test and make necessary changes, then another aircraft would be built shortly before the competition incorporating the lessons learned from the first. It was anticipated that manufacturing experience with the first airframe along with improved structure and load information would enable a reduction in weight and other improvements in overall performance. One of the research engineers at ASDL, Mr. Adam Broughton, a former student DBF member, was designated by the lab director, Dr. Dimitri Mavris, to lead the series of design competition classes which teach aircraft synthesis and design. These classes use the DBF competition as a practical application of design methods. For this reason the conceptual design of the project did not begin until the beginning of the fall semester, 2001.

A review board comprised of prior DBF team members, faculty, and the current DBF team members met to discuss prior year's experience with DBF entry development efforts. The review board determined that the primary contributor to previous DBF team's difficulties were uncertainty and error arising from a lack of applied design and fabrication experience. The board decided that validation of the design through building and testing were paramount in establishing the fit between the performance prediction models and achieved results. Based upon this decision, the 2001-2002 Georgia Tech DBF team emphasized testing and data gathering more than in previous years.

1.1 Progress Overview

The timeline in Figure 2-2 outlines the teams schedule performance as compared to plan. It can be seen from the figure that the team did a fair job of executing the intended plan to schedule. Conceptual design was completed in the first week in September, and the conceptual design review was done the following week. At this point the team down-selected to a single final configuration for the aircraft. This baseline was carried into preliminary design as information supporting a final configuration selection was still being refined. This preliminary design baseline was a low-wing monoplane with tricycle gear. The

canard configuration was a close second, but was dropped before the preliminary design phase and the team chose instead to employ the low-wing monoplane configuration. Other alternative configurations that were considered but not down-selected from the conceptual design phase were a pusher canard with tricycle gear, biplanes, flying wings, box-wings, and high-wing monoplanes.

Though initially in favor early in conceptual design, the canard configuration was eventually eliminated because of information gathered on pitching moments for canards. It was suspected that the canard would be less stable for most flight conditions if the center of gravity was set to rotate quickly for takeoff. Thus, the team progressed with the low-wing monoplane configuration.

The flight mission was analyzed in detail to determine estimations of power requirements, the impact of payload on score, and the penalties of slow flight or not completing laps. Coordination with the students working on propulsion systems research showed that not only was carrying a full payload feasible, but close to optimal. Preliminary structural analysis showed that the planform design of the wing was more limited by structural limits than by aerodynamic qualities. Using regression analysis against historical data, the team was able to determine a reasonable shape for the wing, the area of which was sized based on point performance requirements. The primary tool for this analysis was the thrust-to-weight vs. wing loading chart described in the preliminary design phase section of this report.

A principal element of the team's risk reduction and performance verification plan for their aircraft was early and continued design performance verification through fabrication and testing. A major milestone of this build and test approach was completed in the preliminary design phase upon construction and evaluation of the fuselage configuration concept. Fuselage characteristics that impact payload loading and unloading posed a high potential risk to aircraft competitiveness that required relatively low effort for verification. Originally, the aircraft was to be loaded from the front, and unloaded from the rear by tilting the aircraft and allowing the balls to roll down the length of the fuselage. Testing revealed that friction between the balls did not allow them to roll easily, and the plan for payload insertion/extraction was changed by the beginning of detailed design. The preliminary design was completed a week behind schedule, as was the preliminary review and report.

Detail design began the formal process of working out all the details of how to make the sized aircraft manufacturable. Final decisions about the motor and drive train were made, as well as final design specifications for construction of the wing, tail, and landing gear. From this frozen configuration, final estimations of thrust, weight, lift, and drag were used to predict the performance of the aircraft and the final score.

Building of the prototype was simultaneous to detailed design, as many parameters, such as finished component weights, were difficult to predict without parallel prototyping to gain relevant experience with both similar configurations and materials. The prototype was built to exact scale and fully functional. Aside from yielding great insight into the validity of the intended design, the prototype was intended to provide spare parts and/or a competition recovery plan in the event that the final production vehicle suffered damage or loss.

There were no universal design methods that guided detailed design because it was the most multidisciplinary of the three design phases. If conceptual design and preliminary design are considered "processes," then detailed design in a prototype development effort is so iterative and inter-linked as to be better described as a "phase." The general methodology followed during the DBF teams Detail Design Phase was a recursive approach of "design, build, test" and best characterized as "experimental engineering." This approach to engineering development was intended to address high levels of uncertainty, with the end-manufactured result being the gauge by which analytical performance prediction and manufacturing capabilities are assessed. It is necessarily recursive because of a lack of detailed, involved engineering, manufacturing, and operational experience applicable to the mission, as well as the design. Put into perspective, and taking into account the advantage of accumulated experience, the detail design effort shares risks of the same general nature encountered by coalition based industry teams working on high risk technology prototypes. Competing requires at a minimum: establishment of working relationships and lines of communication along with allocation of responsibilities and resources among the engineers; selection of a design configuration believed to be capable of meeting requirements; proper specification of the design through selection of buildable structural and mechanical concepts; selection of actual hardware and materials for manufacture; assembly and system integration; and eventually test and evaluation not only of the vehicle performance but development of an overall competition operation strategy. After all of this effort, if successful, the final purpose is performing the mission in a narrow window of opportunity, with no later chances for incorporation of lessons learned through design evolution.

In any product engineering effort, the initial engineering models and estimates, along with manufacturing related knowledge of producible characteristics are continually refined on the basis of ongoing experience with production, inspection, test, operation, and service of previous design. In a prototyping effort in which little team experience exists for the product being manufactured, a build and test early approach is necessary to verify predictions versus completed hardware characteristics so that the design may be properly closed. As an example, suppose the estimated drag (a function of several design variables) for the aircraft was 4 lbs at cruise, but actual realized results were 5 lbs of drag, and the aircraft was unable to complete all laps. In this case, one of the controllable variables would have to change, such as the design of the wing, the propeller selection, the amount of payload, etc. The actual path followed to arrive at acceptable results did not follow any path definable in advance—most significant problems were "unknown unknowns" and were only discovered by building and testing. Beam theory and materials research guided the structural design, motor theory and regression analysis guided the design of the drive train, and flight performance predictions were done using traditional fixed wing performance analysis, principles of engineering mechanics, and physics.

1.2 Other Tools

A dynamometer for electric motor testing was built at ASDL as part of a research contract with NASA Langley for electric propulsion systems for UAVs. Research engineers at Tech allowed students to use the dynamometer to test many of the Graupner and AstroFlight motors allowable by contest rules. Motor performance at a wide range of inputs was tested and recorded by DBF students. Some of the measurements taken were operating voltage, current, RPM, torque, and internal coil temperature. Using these measurements, the team was able to accurately predict the output performance of the motor given a set of inputs. The effects of motor timing on performance were also investigated, as well as the effects of motor cooling.

Once the data gathered from the dynamometer defined the capability of the motor, the team needed to determine the capability of propellers. An electric motor mounted to a strain gauge was used in a small wind tunnel to test the thrust performance of various propellers. Airspeeds from static up to 80mph were tested and recorded. This data was used to calibrate a Combined Blade Element Theory (CBEMT) model developed by two of the DBF students. The combined models for motor performance and propeller performance were used to figure propulsion performance for different scenarios. This was especially useful in estimating performance during preliminary design.

Onboard telemetry developed by the previous year's DBF team was available for installation on this year's prototype aircraft. The telemetry uses a traditional analog six-channel radio system with the transmitter circuitry mounted on board the aircraft and the receiver linked to a laptop on the ground. The telemetry records motor RPM, current, voltage, airspeed, motor (surface) temperature, and battery (surface) temperature and sends the signal to the laptop computer. A simple MS-DOS program records the data points and writes them to an MS-Excel file for post-processing. Six data points are recorded about six times per second, the team could define the performance of the aircraft at any point during a test flight. This data was used in takeoff, climb, cruise, turn, and glide performance analysis.

Another one of the major design tools was the use of computer aided design. IronCAD was the major CAD package used by the team. IronCAD was chosen because it is much simpler than most other CAD packages and can be installed as a drafting tool on a desktop PC in Windows. Every component of the aircraft, including retract units, struts, wheels, servos, and other components were drawn to scale in the program so that the team could organize the placement of components as to minimize the required internal area. Structural members were drawn to scale and printed for the students manufacturing of the airframe.

The use of CAD has been an invaluable tool in the development of DBF projects at Georgia Tech. In conceptual design, group discussions were often done in using NASA's Rapid Aircraft Modeler (RAM) projected onto a screen in front of the group. Team members could suggest configurations of wing, fuselage, and payload to discuss advantages and disadvantages of each design. A third, more sophisticated CAD package was used in structural analysis. The Finite Element Analyzer (FEA) of IDEAS

was used to visualize the load on the wing during flight (distributed load) and during the technical inspection (three-point loading). Bending and torsion loads were used in the spar design of the wing.

An important assumption made about the report format was that the limitation that all figures must be in ½ or full-page format does not apply to tables, which are lists of data. Throughout the report, all figures are scaled to ½ or full-page size, while tables are left in a readable format to make the presentation of pertinent data easier to interpret.

The results of this design as they apply to specific portions of the mission requirements are summarized in Table 1-1. Maximum velocity for no payload is not reported since the aircraft was designed in response to the most difficult (the fully loaded) case.

RFP Specifications	Constraint	Buzz Light		Section Discussed
Mission	6 laps with max payload in less than 10 min	5:32		5.8
Propulsion	Off-the-shelf electric motor	Astro Flight Cobalt 90		4.2.1
Payload	Minimum 10 Softballs - Maximum 24 Balls	24 Softballs ~ 9.6 lbs		4.3.2
Situation	Internal, at least two abreast, no stacking	Fuselage		5.1
Accesibility	Must load after first sortie and unload after second sortie	Removable fuselage cover		6.1
Takeoff	Under 200 ft.	190 ft.		5.8.1
General Performance Parameters		**24 Balls**	**No Balls**	
Payload Weight Fraction	-	0.405	N/A	4.5
Cruise Velocity	-	38 fps	50 fps	5.8
Maximum Velocity	-	70 fps	N/A	4.4
Max G Loading	-	4-g	6.5-g	5.8.2

Table 1-1: Compliance Matrix

2 Management Summary

In year's past, the design team has been divided into three groups: a structures group, a propulsion group, and an aero-design group. Although working well, it was recognized that better work was done when groups were small, and tasks could be delegated to individuals instead of groups. Early in the project, a diversity of ideas was important to screen competing concepts, but later in the design, toward detail design, jobs become more specific and group cohesion is less important.

The architecture of the group was organized under this idea of moving from general to specific. Early in the project all group members worked together on Conceptual Design to suggest concepts and discuss ideas. Two members were separated from the group to work on motor performance. The remaining students discussed Conceptual Design. Since many students had never previously participated in DBF, this was a good time for team members to become familiar with the challenges of the contest. The experienced members of the team were able to point out difficulties with specific configurations that less experienced members may not have understood.

After an aircraft and payload configuration baseline had been selected, the team began the mission analysis and vehicle sizing that would maximize the flight score. At this point, the team progressed into more detailed work, and team management decided to divide the large group to smaller groups. Just over half the group continued on to mission analysis and vehicle sizing, and the others were tasked with propulsion systems research. One student was assigned the task of building a database of UAVs, and recorded statistics on large electric UAVs. One additional student was assigned to join the two already gathering motor performance data with the dynamometer, and two were assigned to developing a model for predicting propeller performance.

Toward the end of Preliminary Design, the team was able to size the vehicle and make final decisions about the layout of components within the airframe. At this point, two more students were separated to begin work on building wing test sections and conducting research on aircraft structures. The material selection process was begun here, and the structural layout of the prototype vehicle was initiated.

As students were delegated to specific tasks, the remaining students continued discussing traditional aircraft design theory. These remaining students were the ones directly responsible for documentation and reporting during the course of the project.

Students were allowed to choose which group they would work in, and were encouraged to change groups at least once during the project. The advisor taught the students the fundamentals of the aircraft design process and guided the progress on the aircraft. Deadlines for major developments in the design were set by the advisor and enforced by the team leader. Later in the design, there were no more than three students in any one group at one time. Often times, research and development was done on the individual level, which made "groups" ill defined. By having one team leader and very small groups, students with more natural leadership ability were the drivers of major development. It was noted by one

student that many businesses are switching to this "team" working environment, rather than a clear hierarchy of leaders, sub-leaders, and workers.

At the end of each design phase, the students would work collaboratively to compile their results and present to local professors, research engineers, and graduate students for further advisement. In all, four presentations were made, and much of the information documented for the presentations was used to compile this report.

It was important to the senior members that the group was allowed to think creatively and contribute diverse ideas. It was also the group's responsibility to see that new, uncultured ideas did not introduce challenges that were outside of the capabilities of the team while encouraging creative solutions. For example, one student suggested that by moving the motor from the nose of the aircraft toward the wing, one could reduce the pitching moment required to rotate the aircraft, and the empennage could be made smaller, thus reducing weight and drag. Unfamiliar with rotor dynamics, many people opposed the idea with concerns that the use of a long driveshaft could create mechanical problems that might require great amounts of resources to solve. There were other concerns that the added weight of the shaft might offset any weight reductions gained from the smaller empennage. Alternatively, they did recognize the potential for shortening the overall aircraft and reducing drag. After consulting a local machinist, the team agreed with the idea and allowed the student to pursue the idea and employ it on the aircraft.

The areas of responsibility and timeline are shown on the next two pages.

Student	Rank	Design Theory	Aerodynamics/Fluids	Procurment and Building	Performance	Propulsion	Structures and Weight	Documentation and Reporting
Mark Birney	grad student	X	X		X			X
Andrew Nall	senior	X	X	X	X	X	X	X
Santiago Balestrini	senior	X		X	X	X		X
Jeff Brewer	senior	X	X					X
Josh Clark	senior	X	X	X	X		X	
Cesar Alzate	senior	X	X	X	X			
David Clark	junior			X		X	X	
Tim Cailloux	junior			X		X		X
Randy Reese	junior		X	X			X	
Johanna Kauffman	junior	X			X			X
Drew Smallwood	junior	X		X	X		X	
Colin Birney	junior	X			X			
Arron Weil	sophomore			X				
Dustin Thames	sophomore			X			X	
Vadim Kim	sophomore	X	X	X	X	X	X	
Graham Clark	sophomore		X	X		X	X	X
Mark Baglia	sophomore		X	X		X		
Arun Saini	freshman		X		X		X	
Damon Rousis	freshman	X		X				
Sarah Riley	freshman			X			X	
Michael Martin	freshman						X	
Justin Gomes	freshman		X	X	X	X		
April Moore	freshman	X			X		X	X

Figure 2-1: Team Architecture

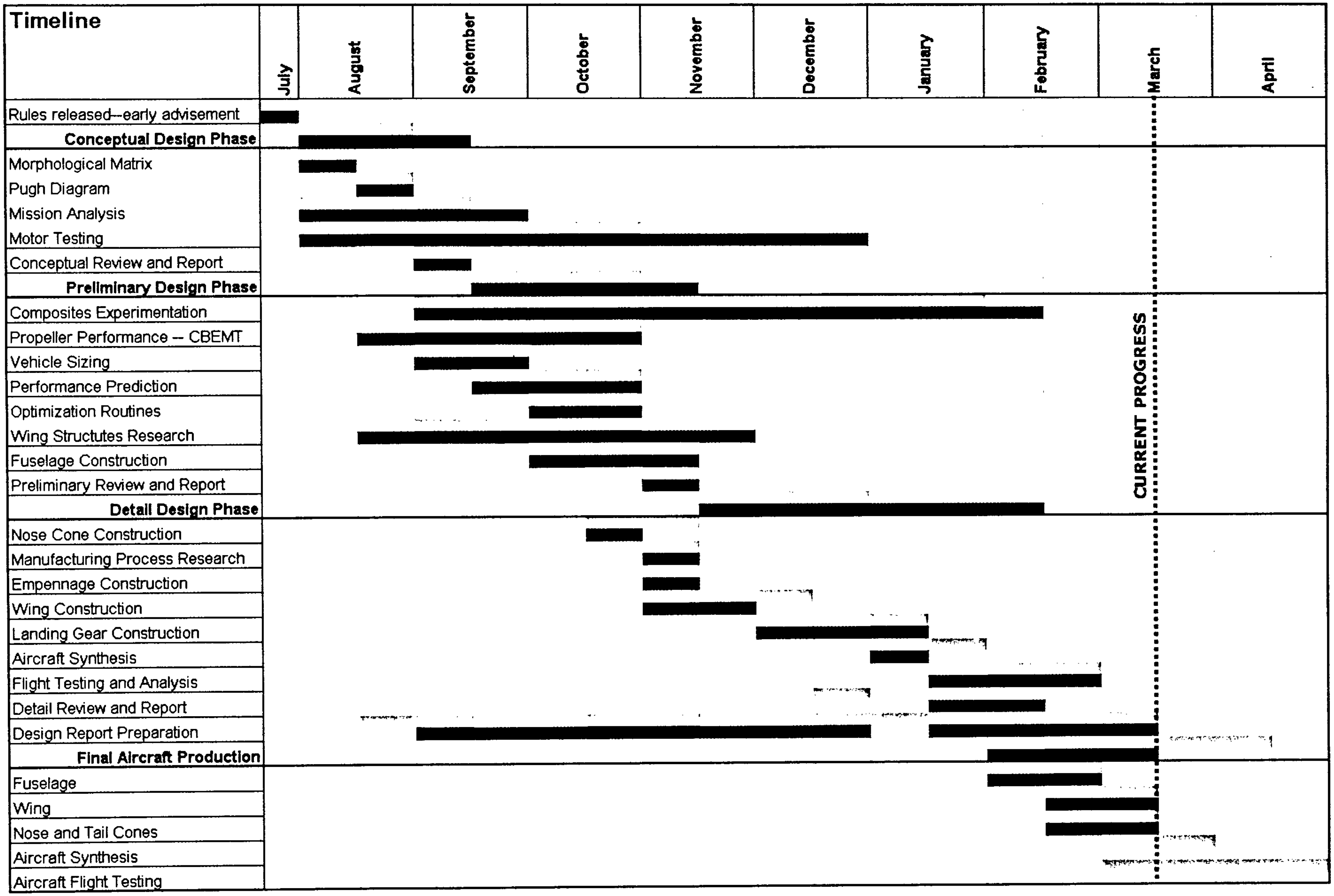


Figure 2-2: DBF Timeline

3 Conceptual Design

The team believed that conceptual design was synonymous with "configuration design" and that the purpose of conceptual design was to determine the overall configuration of the major components of the aircraft. There are currently no standard practices for conceptual design in the aerospace industry. The design team felt that conceptual design is an inexact art, involving more qualitative analysis than quantitative. With technological advances in manufacturing practices, the process itself has been given more attention in the past few years.

The approach to conceptual design that the Georgia Tech team chose, was to first clearly define the problem. This process involved studying the Request For Proposal (RFP), which in this case is the competition rules, for implications about the project and its optimal result. The rules were analyzed for hard constraints and requirements about the configuration of the aircraft. A morphological matrix was then created and analyzed against the constraints and requirements to produce eight configurations. Design parameters were then established as a basis of an overall design to ensure that the design is as efficient in all aspects of the design as possible. Relevant Figures of Merit pertaining to specific requirements were created as a means to compare configurations quantitatively. These constraints, figures of merit, and design parameters were used to form the selection of a configuration.

The combination of all applicable design alternatives in this range space was documented to form the master morphological matrix, which included all combinations of design alternatives most representative of reasonable possibilities. In all, this matrix consisted of some 17,000 combinations. This morphological matrix was further refined to a smaller, more concise set of alternatives used in a modified Pugh diagram. These alternatives were then scored relative to a corresponding figure of merit. Each Figure of Merit was assigned a weight, and these weights were used to score alternative configurations against each other.

The result of the conceptual design phase was intended to be one favored configuration. This configuration would be the basis for all additional work, and so was evaluated very carefully.

3.1 RFP Review

The RFP defined the scope and limitations of the team's contest entry. The first review of the RFP during conceptual design revealed several major points that would guide the design. One of the major points noted was that the maximum allowed payload was significantly less than the payload weight carried in steel the previous year (when battery pack weight limit was the same 5 lbs). In addition, two-thirds of the flights in the total mission are required to be flown unloaded. The payload weight issue, along with the limit on the number of laps, told the team that this season's contest entries would be faster and lighter than in the previous season. The emphasis on speed meant higher wing loading, lower static thrust for takeoff and more careful drag-reduction measures than in previous years.

In addition to flight speed issues, previous years have had a ten-minute time limit to complete scoring runs. It has been noted that in previous years, most teams would run out of battery capacity

before the ten-minute time limit was reached; not needing to rush the insertion/extraction of payload. In this year's competition, the flight time is scored to favor shorter mission times. The use of speed loaders is prohibited, and so the team spent a great deal of time discussing the configuration of the payload such that loading and unloading the payload was extremely fast.

Another major point noted from reviewing the RFP was that the wingspan limit was removed. It would be argued later that increasing aspect ratio (in order to minimize induced drag) and maintaining wing area and airspeed, would show diminishing performance because of very low Reynold's numbers. Empirical test data has been shown to indicate that a wing's performance is less predictable due to the formation (and movement) of laminar flow separation bubbles. Lift curves and drag polars are highly irregular for low Reynold's numbers. However, accurate information on the performance of airfoils at low Reynold's numbers (that may define these irregularities) was found unreliable itself. Without reliable information, and without the proper means in which to research the answer, the team had to deal with uncertainty as one of the leading design factors guiding decisions. It would be shown later that structural limitations (and weight implications) would have a greater effect on limiting the span than the tradeoff in performance due to low Reynold's numbers.

It was also noted that in the Rated Aircraft Cost (RAC) of the contest rules, there was relatively little penalty for using multiple motors as compared to previous years. Using multiple motors was therefore seen as a potential option for optimizing flight score. Performance data for the range of motors that were legal for the competition are very difficult to obtain. It was decided to purchase many of the motors and run experiments on a dynamometer in order to obtain the performance data of different motors under the same set of conditions. This performance data would later help the team to select the number of motors and predict the performance of the propulsion system.

In the RAC, the cost per unit length of chord is the same as the cost per unit of span. This meant that there was a penalty for any wing with sweep or taper, and it was assumed that the team would use a straight wing to maximize the scored wing area.

The RAC also defines a cost for the maximum length of the aircraft but not for the number of fuselages. The rules state "10 hr/ft body maximum length" which has been interpreted as a cost of ten hours per foot of the maximum length of the aircraft [1]. The team therefore assumed that a twin fuselage configuration that was shorter than a conventional single fuselage of greater length would yield a higher overall score. Similar arguments did not hold for the number of wings, as it was specified to "sum (the) values for multiple wings" [1].

3.2 Configuration Breakdown

The team thought of every major design decision that could be made without involving DBF specific criteria. The placement of the wing on the fuselage, the placement of the empennage, the type of empennage, and the orientation of the landing gear were some of the decisions included. This exercise was intended to identify all major combinations of aircraft configurations that would make up the design

space. This set of combinations was called the master morphological matrix and only existed in theory because of its nearly 17000 possible combinations.

This matrix was then refined by eliminating any configurations that clearly did not apply to this application, or were not allowed by contest rules. Sailplanes, jets, rotorcraft, and lighter-than-air vehicles were some of these configurations eliminated. The matrix was then further refined by eliminating any configurations requiring decisions that could not be made at a superficial level, or requiring detailed trade studies. Such decisions included the use of wheel pants, retracts, lifting bodies, circulation control, winglets, and faring options. It is important to note that some of the options eliminated in this step were not eliminated altogether, but only excluded from being considered at the conceptual level. Historical data from full-size aircraft and analysis of past DBF competitions were the basis of the selection of general configurations. The idea here was to eliminate useless configurations that would not be optimal for the type of mission required while not limiting options for that could maximize performance specific to each configuration.

Design Parameters were chosen in conjunction with Figures of Merit and analysis of the RFP to be used in the selection of the configurations. The design team interpreted Design Parameters as quantifiable terms used to drive the design and determine which configurations would be optimal for the required mission. As previously stated in the RFP review, speed is crucial to this year's contest. The flight score received is inversely proportional to the flight time during the competition, so the faster the plane, the lower the flight time and the higher the score. Similarly, the amount of payload carried is directly proportional to the score making it optimal to carry the maximum amount of softballs during the mission. Weight was also used as a parameter due to its impact on the Rated Aircraft cost. Both the aircraft's empty weight and the weight of the battery it carries affect its score and are therefore important when determining which configuration to be analyzed.

Assumptions were made for the design parameters so that exact numbers could be used when determining minimum requirements for each of the configurations. For example, each configuration drawn from the morphological matrix must, at a minimum, be able to complete the six laps, loaded and unloaded, in less than ten minutes. To attain this, the design team only considered those configurations that would be able to achieve a max speed of 55 mph. The complete list of assumptions made for the Design Parameters are shown in Table 3-1.

The eight configurations chosen to be analyzed in the Conceptual Design Phase were the high wing monoplane, mid-wing monoplane, low wing monoplane, biplane, flying wing, canard, box or joined wing, and delta wing as shown in Figure 3-1.

3.3 Configuration Analysis

Reviewing the RFP generated two lists:

1. Constraints and Requirements
2. Figures of Merit

Investigating the RFP for "design constraints" did not result in any specific conclusions about aircraft configuration. The rules define limits on takeoff performance, weight of the battery, configuration of the balls, etc, but there is no "constraint" on the physical dimensions of the aircraft in span, length, or otherwise. The only mention to configuration was that the vehicle could not be rotary wing or lighter-than-air. It was noted that the configuration of the payload was constrained between 10 and 24 softballs, and at least two abreast.

The team had defined "figures of merit" as those design considerations felt important to the development of the design, but not constraining to it. These were favorable or unfavorable characteristics about the aircraft that were worth considering, but not inherently constraining the design space.

An all-inclusive list of Figures of Merit was drafted, and later consolidated to facilitate scoring. Table 3-2 shows the first list of Figures of Merit.

This compilation of Figures of Merit was the basis for evaluating competing configurations. It was noted that many of these Figures of Merit could be combined into one category. For example, "Mechanical Simplicity" and "Minimum of intersections" would be argued similarly in evaluating configurations. Such smaller sets of categories would each be assigned a score based on how important the team felt it was to influencing the configuration of the aircraft. It was found that categorizing the Figures of Merit in such a way was much more logical and easier to evaluate competing concepts. These categories are shown in Table 3-3.

The scores assigned to each of these categories were figured by qualitative assessment of each category in-group discussion. A category such as "Stability and Control" was not weighted as very important. It was argued in the group that the use of computer controlled mixing, reducing control surface throws, shifting the center of gravity, and/or the use of piezo-electric elements for stability and control could offset many unfavorable flight characteristics. "Ground Handling" was also not scored very high either, but was scored higher than stability and control because it was felt that there was less freedom to correct unfavorable qualities on the ground than in the air.

"Rated Aircraft Cost" was originally part of the "Contest Score/Points Awarded" category, but was separated because of its heavy influence on aircraft configuration. "Ease of Analysis," one of the original Figures of Merit, was very important to the team because of the relative inexperience of team members was a significant risk factor for the team. "Aerodynamics" is directly related to "Ease of Analysis" but scores higher because of the team's dependence on aerodynamic theory and calculations.

"Reliability and Simplicity" were seen as the fundamentals of good engineering practices and scored a seven, but the "Function" category was weighted a nine. The argument here was that a mechanically complex configuration could be acceptable if its operation was highly robust. Difficulties with inserting or extracting the payload could have potential devastating impacts on score. The two other categories, "Contest Score/Points Awarded" and "Rated Aircraft Cost" were given the highest weight.

3.4 Configuration Selection

After reducing the morphological matrix down to eight aircraft configurations, the remaining set of options was the list of decision criteria that the team used to screen competing concepts and is shown on the left side of the modified Pugh Diagram in Figure 3-2. Each of these decisions was evaluated using the Figures of Merit described earlier. The value assigned to each decision was multiplied by the FOM weight and then totaled at the right in the figure. In each category, the decisions were ranked from lowest score to highest. The decision with the highest score was interpreted as the best option.

The placement of the fuselage to the wing was one of the most important decisions about the aircraft configuration. In the Pugh Diagram, the first three assume a single wing placed above, midway through, and below the payload.

It was argued here that if the center of gravity was located below the center of lift as in the high-wing configuration, then the vehicle would have some built-in inherent stability. However, placing the wing high meant that the main gear would most likely be located in fuselage (close together) or become very long if located in the wing. In the former case, the fuselage would have to be built very strong in order to support the gear. It was argued that a lighter construction would be to mount the gear in the wing, which is typically built very strong because of the three-point inspection and expected Wichita gusty weather.

Mid-wing constructions followed similar arguments of gear placement and heavy fuselage construction, but wing-mounted gear were seen as more feasible. One argument against the mid-wing configuration was that a wing strength member would likely have to penetrate the fuselage. This condition would make rolling the balls through the fuselage infeasible.

The low wing configuration was considered best for gear placement, but aerodynamically there was the potential for inherent instability if dihedral were not employed. The limiting case of required dihedral was not appealing with the team, but counter arguments indicated that the center of gravity could be placed close enough that inherent instability might not be a concern. It should also be noted that stability and control were not weighted that heavily as compared to other Figures of Merit.

Considering these first three configurations, it was noted that the thrust line could be assumed collinear with the fuselage. If the wing is placed above the line of thrust, the aircraft assumes a positive pitching moment. Similarly, if placed below the line of thrust, the aircraft assumes a negative pitching moment, and when placed in-line with the wing, no moment is produced. Clearly the high wing configuration was favored for this reason.

The biplane configuration was listed among the other configurations because of similar arguments. There was the advantage of favorable gear placement, no interference in the fuselage, and potential inherent stability. Arguments against the biplane included concerns about interference effects between wings, between the wings and the empennage, and added structural complexity. Ease of analysis was a large concern because of the interference of the two wings with each other.

The flying wing configuration was not favorable because of concerns with stability and control, and ease of analysis. Although this configuration scored well in the RAC, historical evidence showed that

there could potentially be serious problems with stability and control. It was also argued that a flying wing needs to have a swept wing in order to obtain an acceptable pitching moment. As described earlier, there is the same cost of 8 hr/ft in the RAC for unit length of chord as unit length of span. Therefore any wing with sweep or taper is penalized more heavily than a straight wing.

Canard configured aircraft were also considered. A strong argument in favor of the canard configuration was the potential for zero trim drag. Not sure at the time if this was true, the team decided to consider drag issues equivalent to conventionally configured monoplanes. Canard aircraft offer the advantage of undisturbed flow over the fuselage because of the necessity to mount the motor in the rear and employ a pusher propeller.

The impact of RAC on score was weighted very heavily in the Pugh Diagram. The RAC figures were some of the few that could be done quantitatively and are shown in Figure 3-3.

The idea of this cost model is to determine the effect the general configuration of the aircraft has on the overall cost model. The parameters that are consistent to each configuration were held constant while the other components were chosen to maximize the score for each configuration. In general, the only values that would change in the cost model were those pertaining to shape and size. For example, weight was kept constant during the analysis because there were no features for any of the configurations that would increase the weight of the aircraft. Similarly, the number of motors, the number of propellers, and the weight of the battery pack were also held constant.

Predictably, the flying wing produced the lowest cost model because of the lack of control surfaces and the lack of servos. A flying wing configuration also allows for a shorter fuselage while maintaining the same wing area as the rest of the configurations considered. Alternatively, the biplane scored so poorly due to the fact that it has more control surfaces and servos than the other configurations. The result that surprised the team was the low cost model of the delta wing. One would think that a tapered wing would score high in this cost model because a team is charged for the maximum chord and the maximum span, as described above. But like the flying wing, a delta wing aircraft will have a shorter and fatter fuselage as well as fewer control surfaces because the lifting wing also provides stabilization, eliminating the need for an empennage.

Assumptions and interpretations had to be made in order to complete this cost model. It was assumed that each configuration would have the most cost efficient empennage that would best compliment each configuration. For example, the three monoplane configurations were assumed to have a conventional tail with one vertical stabilizer and one horizontal stabilizer. Additionally, the elevator was considered as one control surface the spanned the length of the horizontal stabilizer. Conversely, the canard configuration was assumed to have two control surfaces on the forward mounted horizontal stabilizers. For simplicity, the design team also assumed that the effective wing area would remain the same for each configuration so that the shape of the wing would determine the cost score it would receive. This is feasible because the weight is assumed the same for each configuration thus making the wing loading required equal.

The results from the configuration rated aircraft cost model were then scored in the Pugh diagram on the same 10 point scale used for the other figures of merit. The flying wing and delta wing received the highest score, while the biplane and canard scored the lowest.

The analysis of the pugh diagram shows the low wing monoplane and the canard configurations to score the highest of the configurations. However, since the canard option had a higher RAC and greater uncertainty in control and the team had little experience with this configuration, the decision was made to continue with only the low-wing monoplane.

Design Parameter	Assumed Value
Speed	55 mph
Payload	24 softballs
Weight	
• Empty Weight	15 lbs.
• Battery	5 lbs.

Table 3-1: Design Parameters

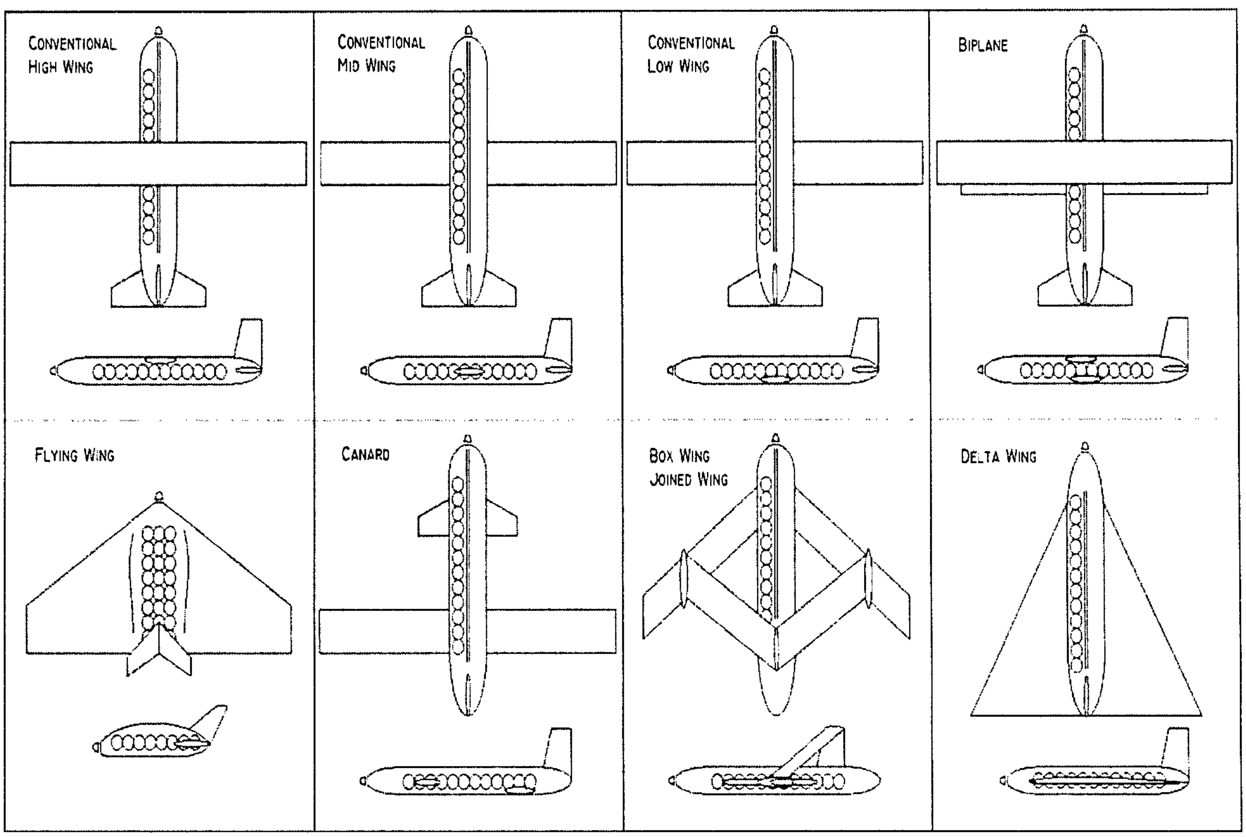


Figure 3-1: Alternative Design Concepts

Figure of Merit	Why important	Preferred Configuration
Flight speed	flight time is scored, high speed preferred	low drag configuration
Payload capability	points awarded per ball, 24 balls preferred	must carry softballs
Payload internal to fuselage	pods increase RAC, fewer points of interference of flow---lower drag	no pods
Payload configuration minimizes drag	lower frontal area an more streamlined body will have better fineness ratio and potential lower drag	only two balls abreast (minimum required)
Changing cargo must be fast	factor in scored time	any that favors easy access
Component changing should be fairly easy	speeds repairs/replacement of components, changing batteries between sorties is expedited	simplicity of design
Inherent stability	safety, less control inputs means lower drag, possible low trim drag	high wing, conventional tail aft
Minimum of intersections	lower interference drag, simplicity in construction, ease of analysis, robust design	single wing, no tail, no pods etc...
Low weight	lower power requirements, lower induced drag, higher speed, more efficient flight,	fewer components
Mechanical simplicity	speeds repair, ease of building, ease of maintenance, robustness of deisgn	any that favors fewer components
Building/ construction simplicity	fewer points for potential failure, ease of analysis ease of manufacturing, ease of maintenance	any favoring fewer structural members
Ease of analysis	validation of design, ability to analyze concepts, confidence in production	any with great historical prevalence
RAC	impact on score	any favoring lower RAC
Ground handling	safety, efficiency of takeoff ground roll/landing ground roll	any potential for favorable gear placement

Table 3-2: Figures of Merit

Category	Score
1. Reliability and Simplicity	7
- ease of manufacture	
- mechanical simplicity	
- robust design	
2. Contest Score/Points Awarded	10
- payload capability	
- minimum flight time with maximum balls	
- payload internal to the fuselage	
3. Ground Handling	5
- low rolling friction	
- rolling stability	
4. Function	9
- extraction/insertion of payload	
- extraction/insertion of battery	
5. Aerodynamics	8
- performance reliability	
- speed, low drag	
- minimum number of intersections	
6. Stability and Control	4
- high inherent stability	
- gust response	
7. Analysis	7
- complexity of analysis	
- available historical data	
8. Rated Aircraft Cost	10

Table 3-3: Figures of Merit Categorization

Wing/Fuselage Configurations	Reliability and Simplicity	Contest Score/Points Awarded	Ground Handling	Function	Aerodynamics	Stability and Control	Analysis	Rated Aircraft Cost	Total	Ranking
	7	10	5	9	8	4	7	10		
High wing monoplane	9	8	N/A	7	7	8	8	8	**430**	3
Mid-wing monoplane	9	8	N/A	7	7	7	8	8	**426**	4
Low wing monoplane	9	8	N/A	8	8	7	8	8	**443**	1
Biplane	7	7	N/A	6	9	8	7	7	**396**	7
Flying wing	6	9	N/A	7	9	6	6	9	**423**	6
Canard	8	8	N/A	8	9	7	7	8	**437**	2
Box wing/Joined wing	7	7	N/A	7	8	7	7	7	**393**	8
Delta wing	6	8	N/A	8	8	7	7	9	**425**	5
Number of wings										
Monoplane	8	9	N/A	8	8	7	7	9	**449**	1
Biplane	7	7	N/A	7	9	7	6	7	**394**	2
Multi-Lifting Surfaces	6	7	N/A	8	8	6	6	7	**384**	3
Empennage Type										
Conventional	8	9	N/A	N/A	7	8	9	9	**387**	1
T-Tail	7	9	N/A	N/A	8	7	8	9	**377**	3
Ring Tail	7	7	N/A	N/A	8	8	7	7	**334**	5
H-Tail	7	7	N/A	N/A	8	8	8	7	**341**	4
Canard	8	9	N/A	N/A	7	8	8	9	**380**	2
Empennage Placement										
Fore	7	8	N/A	N/A	8	9	7	8	**368**	2
Aft	8	8	N/A	N/A	7	8	9	8	**367**	1
Pods, Nacelles										
Used	6	9	N/A	7	7	7	7	9	**418**	2
Not used	8	8	N/A	8	9	8	8	8	**448**	1
Gear Placement										
Tricycle	7	N/A	9	8	8	8	8	N/A	**318**	1
Tail Dragger	8	N/A	7	8	8	7	8	N/A	**311**	2
Propulsion										
Tractor	9	8	N/A	N/A	8	N/A	9	8	**360**	1
Pusher	8	8	N/A	N/A	8	N/A	9	8	**343**	2
Ducted	7	8	N/A	N/A	7	N/A	7	8	**314**	3

Figure 3-2: Decision Matrix

Configuration Analysis
Rated Aircraft Cost

Description	High Wing Monoplane	Mid Wing Monoplane	Low Wing Monoplane	Biplane	Flying Wing	Canard	Box Wing / Joined Wing	Delta Wing
Manufacturer's Empty Weight (MEW)								
Empty Weight	15.000 lbs	15 000 lbs	15 000 lbs	15 000 lbs	15 000 lbs	15 000 lbs	15 000 lbs	15 000 lbs
Rated Engine Power (REP)								
Number of Motors	1 motor	1 motor	1 motor	1 motor	1 motor	1 motor	1 motor	1 motor
Battery Weight	5 000 lbs	5 000 lbs	5 000 lbs	5 000 lbs	5 000 lbs	5 000 lbs	5 000 lbs	5 000 lbs
Manufacuring Man Hours (MFHR)								
WBS Wing								
Total Span (Effective)	10 00 feet	10 00 feet	10 00 feet	10 00 feet	9 00 feet	10 00 feet	11 00 feet	10 00 feet
Max Exposed Wing Chord	1 00 foot	1.00 foot	1.00 foot	1.00 foot	1 foot	1 foot	1 foot	2 feet
Number of Control Surfaces	4 surfaces	4 surfaces	4 surfaces	6 surfaces	4 surfaces	5 surfaces	6 surfaces	5 surfaces
WBS Fuselage								
Body Maximum Length	6 00 feet	6 00 feet	6 00 feet	6 00 feet	4 00 feet	6 00 feet	6 00 feet	4 00 feet
WBS Empennage								
Number of vertical surfaces with no active control	0 surfaces	0 surfaces	0 surfaces	0 surfaces	0 surfaces	0 surfaces	0 surfaces	0 surfaces
Number of vertical surfaces with active control	1 surface	1 surface	1 surface	1 surface	0 surfaces	1 surface	1 surface	1 surface
Number of horizontal surfaces (span <25% of wing span)	1 surface	1 surface	1 surface	2 surfaces	0 surfaces	2 surfaces	0 surfaces	0 surfaces
WBS Flight Systems								
# Servos/Controllers	5 servos	5 servos	5 servos	5 servos	4 servos	5 servos	6 servos	5 servos
WBS Propulsion System								
Number of motors	1 motor	1 motor	1 motor	1 motor	1 motor	1 motor	1 motor	1 motor
Number of propellers or fans	1 propeller	1 propeller	1 propeller	1 propeller	1 propeller	1 propeller	1 propeller	1 propeller
Rated Aircraft Cost, $ (Thousands) =	**13.30**	**13.30**	**13.30**	**13.62**	**12.24**	**13.56**	**13.48**	**12.92**

Figure 3-3: Rated Aircraft Cost for Alternative Configurations

4 Preliminary Design

The preliminary design process was carried out by first considering the design parameters to be discussed. These are primarily related to the propulsion system, vehicle weight, and non-dimensional vehicle size. Once discussed, a number of assumptions can be made that effectively determines some of the parameters. From these simplifying assumptions, the remaining design parameters can be traded off and selected according to various metrics.

4.1 Design Parameters

The primary purpose of preliminary design is to investigate the trade-offs for a set of key design parameters. These trade-offs, if successful, indicate the best settings for the said parameters. However, this is often at the limit of the fidelity of the analysis tools used for this design phase, so caution must be used to arrive at the most robust solution due to uncertainty in the analysis and manufacturing of the vehicle.

The team identified several key parameters crucial to the ultimate description of the aircraft. These are mapped to one or more design or mission Figures of Merit (FOMs). These parameters and associated FOMs are listed in Table 4-2. The reasons for the association between the parameters and the FOMs will become readily apparent in the sections that follow.

4.2 Propulsion System

The propulsion system is required by the contest rules to be an over-the-counter brushed electric motor powered by nickel-cadmium batteries turning a propeller or ducted fan. Constraints are imposed regarding the weight of the battery pack and the maximum current draw for the motor. Together, these requirements represent an upper limit on the maximum power available from the propulsion system.

4.2.1 Motor Selection

The motor selection process for the vehicle was greatly simplified by the contest rules. Limiting teams to only two manufacturers identified the set of possible candidates. Traditional motor theory was used to predict motor performance, but the team relied more heavily on empirical test data using a dynamometer. One of the first decisions to be made was weather to use a single motor, or multiple motors. Then the team had to decide which motor would be optimal for the application.

Motor theory suggested that larger motors operating at high RPM would be more efficient in terms of transforming electrical power into physical power. This is partly because of Copper loss, which is back electromotive force (back EMF) caused by moving a coil through lines of flux. Another is iron loss, which is the loss due to the electrical resistance of the components (e.g., copper wire, carbon brushes, solder joints, etc.).

The higher efficiency is also related to the fact that more turns of wire of a larger diameter can be wound on the armature if the armature is larger. Smaller wires have more resistance per unit length,

requiring more turns in the coils to obtain the same effect. Reference [11] states that the force acting on a conductor is given by

$$Force = \frac{B \times I \times l}{10} dynes \tag{4.1}$$

which is quite literally proportional to the torque divided by the diameter of the armature. Generally speaking for a given "size" motor, armatures with fewer wounds correspond to low voltages, and armatures with higher wounds correspond to higher voltages.

The Graupner line of motors offered several different armature types from which to choose, but the AstroFlight line of motors did not. AstroFlight does however offer a much larger motor than Graupner, the Cobalt 90. This was the motor chosen by the design team.

Given a set of input conditions, such as voltage and load, the AstroFlight Cobalt 90 has the capability to put out more power than its next smaller option, the Cobalt 60. This was appealing to the team when considering that the aircraft would require over 2 Kilowatts to complete the mission at acceptable speeds.

4.2.2 Battery Selection

The contest rules specify that the battery pack or packs for the propulsion system may not weigh more than five pounds. This represents a limit both on voltage and on energy storage capacity. The number of batteries in series determines the voltage available to the system, while the number of batteries in parallel determines the energy capacity of the system.

The current input to an electric motor dictates the torque available to the propulsion system, while the voltage is directly proportional to the shaft speed. Therefore, applications requiring more torque (current) and less speed (voltage) favor a parallel battery setup because this effectively doubles the capacity of the batteries while keeping the voltage the same. Applications requiring more speed favor series connections because of the higher voltages allowed. As outlined above, the system chosen for this aircraft was a large, direct-direct motor. This implies a smaller diameter propeller that requires more speed than torque. Therefore, the favored battery setup is a fully series connection.

The question remains as to how many, and what type, of batteries to use. Almost all nickel-cadmium batteries on the market today give the same voltage, so the discussion turns to a matter of capacity. In general, larger capacity implies a higher weight. For the same weight, it is possible to get more smaller capacity cells, and thus voltage, than larger capacity cells. This, of course, comes at a cost in total run-time for the same current draw. Therefore, selection of cells is a compromise between voltage and vehicle run-time.

The main assumptions used for battery selection involved the current draw. The best design will achieve a maximum speed with a maximum payload and will be able to accomplish the mission with little in reserve energy capacity. This implies that maximum power be drawn from the system, where power

input to the system is given by the product of the input voltage and current. Therefore, smaller capacity batteries will yield higher power for a fixed current draw, but will have a drastic decrease in run time. The team selected four candidate batteries of differing capacities to determine the maximum power available and the subsequent static run time. This static run time was found by assuming a current draw of 40 Amps (the maximum allowed by the competition rules) and assuming that the best configuration is that with the maximum allowable battery weight of five pounds. This weight was modified by adding the stipulation (from experience) that one ounce of shrink wrap, solder, wire, etc. must be added for every ten cells in a battery pack. The voltage was estimated to be 1.2 Volts for each cell. The results of this study are given below in Table 4-3. Note that the voltage, power, and run-time are for the ideal case of zero losses and are thus for comparison purposes only.

This table indicates what is expected: increasing capacity gives larger run-time at the expense of power. Note that as real effects are included, the voltage drop for the higher cell packs will be more noticeable due to the added resistance of more cells and connections. More importantly, however, is the static run-time. At this point it is safe to assume that the entire mission will take more than four minutes, so the last configuration appears as the only feasible option. It is important to note that the current draw should decrease when the vehicle is in flight, so the run-time numbers are greater when considering flight time. However, the large voltage drop seen in battery packs when the capacity is near exhaustion may offset this extra flight time. As such, the team chose the last setup, that of 26 Sanyo 3000CR nickel-cadmium cells in series. This yields a battery pack weight of approximately 79.9 ounces (76.9 ounces of batteries plus 3.0 ounces of shrink wrap, wiring, and solder), a mere 0.1 ounce shy of the contest requirement. The nominal battery pack voltage is 31.2 Volts, with a theoretical maximum power of 1,248 Watts at the contest maximum 40 Amps current draw.

4.2.3 Propeller Selection

This final aspect of the propulsion system is extraordinarily important, as the propeller is the only portion of the system providing a positive momentum flux to the flow to produce thrust. Model aircraft propellers are chiefly described by two parameters: pitch and diameter. Propeller pitch can be thought of as the screw angle, that is, a propeller with an eight-inch pitch will advance eight inches through a solid medium for every revolution. Pitch is typically measured at 75% of the propeller radius, and is related to the induced velocity of the propulsion system. Propeller diameter is simply the diameter of the rotating propeller disc and is related to the mass flux of the system. Together, these parameters specify a combination of required torque (current) and shaft speed (voltage) for a given thrust. This thrust and thrust variation with speed (often called the thrust lapse) can be uniquely determined for a given propeller geometry, motor, and battery setup.

In general, increasing a propeller's diameter will increase the static thrust of the system due to the larger disc area and corresponding larger mass flux. However, this typically comes at a cost in thrust lapse rate as larger discs have lower induced velocities and thus pitch speeds. Increasing propeller pitch sacrifices low speed performance and efficiency (indeed, the blades can stall at low freestream velocities

if the pitch is too extreme), but allows for better high-speed performance because of the larger induced velocity through the propeller disc. These sensitivities to propeller parameters were quantified with the motor and battery setup listed above with a program called Motocalc [10]. Though previously unreliable, the latest version of Motocalc has completely updated thrust and power models that are far more accurate than before.

In this case, the best propeller will be that with the most favorable thrust lapse (highest pitch) and largest disc (highest diameter) that will not exceed the 40 Amp current draw limit. Note that this assumes that the peak current draw will occur for static conditions. In this case, pitch was considered to be more important than diameter due to the requirement for high speed to win. This will result in less efficient takeoff, landing, and taxiing, but will allow for faster response in the air along with higher velocities. Motocalc was used to compare the effect of varying pitch on current drawn for a sweep of diameters. These effects are shown in Figure 4-1. The ranges of diameter and pitch shown in this figure were dictated by the practical limits of the motor system (direct drive, etc.). Also, the trends below are for a generic propeller. The actual efficiency of the propeller is partially determined by its shape, which generally differs for different brands.

This figure shows that the best propeller for the aircraft is one with a 16-inch diameter and 14 inches of pitch. This amount of pitch is extreme in the face of current radio controlled model airplane trends, but is necessary to maintain a competitive speed in the air.

4.2.4 Propulsion System Analysis

The definition of all the parameters discussed in this subsection allows the designer to determine the characteristics of the propulsion system useful for the design and performance analysis of the aircraft. Motocalc calculations revealed the static thrust of the configuration to be 121.3 ounces at 726.5 Watts. This is accomplished at a current draw of 37.3 Amps and voltage of 28.0 Volts after losses, yielding a motor input of 1045.2 Watts. This yields a motor efficiency of 69.5% and an overall efficiency of 62.4%. The in-flight response of the system can be described by the thrust lapse. This is shown with current in Figure 4-2, and is a direct output from Motocalc.

The variation of current with airspeed is important to make sure that the current draw does not exceed 40 Amps in any flight condition. The previous assumption that maximum current draw occurs at static conditions is shown to be false from this diagram, although it is very close to the actual maximum value so it can be a close approximation. The estimated peak value of the current draw is 38.0 Amps between 25 and 35 miles per hour, still safely below the 40 Amp maximum.

The thrust lapse is equally important to understand for performance estimation. It plays a role in determining the wing loading and thrust-to-weight ratio of the aircraft as discussed later, and also is pivotal in the estimation of aircraft performance. Therefore, a polynomial regression was used on the thrust lapse, and the resulting formula for thrust lapse was

$$T = 121.56 - 0.5154V - 0.00102V^2 \tag{4.2}$$

where T is the thrust in ounces and V is the flight velocity in feet per second.

4.3 Sizing Investigation

The payload capacity specified in the competition rules was required to fall in a certain range, so studies were necessary to determine the optimal payload and the vehicle weight to carry such a payload. These studies are outlined below.

4.3.1 Preliminary Weight Estimation

Current practices in preliminary weight estimation often use regressed empirical data at some level of abstraction. This level is usually some gross predictor variable (such as the ratio of empty weight to gross weight) or broken down to component weight estimates. In either case, the selection of the database and of the predictor variable or variables is of paramount importance. The database is often limited to a certain vehicle configuration and mission profile.

There is little in the way of component weight data for radio-controlled aircraft. Therefore, a component-based weight estimate can only be based on experience or testing. The team constructed multiple test articles, keeping close scrutiny on the materials and fasteners used, in an attempt to create a component-based weight model for model aircraft. Unfortunately, not enough data points were possible by the publication of this report to create a useful regression. Therefore, the team had to rely on gross vehicle-based weight estimates.

The database chosen for the estimated weight regression was from the pool of previous design/build/fly entries. Much data was available on entries from the 1996-1997 through the 1998-1999 contest years from the contest website [1]. Information was also available from other sources, such as from former team members of other universities who were willing to furnish such information. Finally, reliable data was available from the Georgia Tech DBF team archives.

The problem with this database is that the vehicles were designed according to different mission profiles and payload requirements for each year. Therefore, the selection criteria for fuselage size, propulsion system size, etc. changed drastically across each contest year. However, these effects can be considered to offset each other for a zeroth-order analysis such as a weight regression with a single predictor variable.

Payload weight was chosen as a candidate predictor variable for vehicle gross weight. This is sensible because all DBF aircraft, regardless of contest year, are designed to carry some form of payload. Thus, this should be a strong predictor of vehicle weight. Note that this assumption was the main reason for the motivation to include only DBF aircraft in the database as opposed to other radio controlled aircraft that do not have a payload requirement. Of course, there was a reasonable scatter in the data points due to the multiple different construction techniques for each aircraft, as well as the different requirements for each contest year. The final regression model was fit to several transformations of the predictor variable to determine the best function. These transformations were

linear, polynomial (2^{nd}, 3^{rd}, and 4^{th} order), exponential, and logarithmic. The R^2 values of these regressions, which indicate how much of the variation of the model is due to the predictor variable, were used to choose the final regression function (higher R^2 indicates better fitness). The best function was a logarithmic transformation of payload as shown in Figure 4-3.

This model indicates that approximately 85% of the variation in the vehicle takeoff weight can be explained by the payload weight, with the rest of the variation due to other predictor variables or sampling error. Note that this relation is only good within the range of the payload weights investigated (6.6 to 25.7 pounds). Indeed, this relation gives a negative takeoff weight for zero payload, obviously a nonphysical result. The softballs specified for the competition weight approximately 6.4 ounces each, so this weight regression will not be applicable for payloads less than 17 balls (6.8 pounds).

The weight estimation was further modified by a factor to reflect the level of improvement available in gross weight expected due to better manufacturing and weight control techniques. Often, teams that show up to the competition have little experience in manufacturing radio-controlled aircraft, so the takeoff weight data points obtained above are slightly higher than that of a team with manufacturing experience, both from performance and previous competitions and due to individual team member contributions. Adam Broughton, the advisor for this project, has extensive experience in manufacturing radio-controlled aircraft, and believed that strict weight-control techniques could reduce the weight predicted from the above regression by approximately ten percent. As such, the final weight estimation function used was

$$W_{TO} = 0.9\lfloor 34.88\ln\left(W_{payload}\right) - 52.564\rfloor \tag{4.3}$$

where W_{TO} is the takeoff weight in pounds and $W_{payload}$ is the payload weight in pounds.

As a final point, this analysis could be of greater fidelity, either in terms of modeling (predictor variables) or error (number of data points) if the competition administration could once again publish aircraft statistics on the contest website. This was discontinued after the 1998-1999 competition, and the team requests that the administration considers publication of aircraft statistics at the completion of future competitions.

4.3.2 Payload Capability Investigation

The total score of the vehicle in the competition, given as the sum of the five best single flight scores times the written report score divided by the rated aircraft cost. The single flight score is given as the sum of the number of laps completed (up to a maximum of six) and the number of balls carried divided by the total mission time. As seen here, the payload directly effects the total score via the expression for number of balls carried in the single flight score. However, the payload has indirect effects on the mission time and rated aircraft cost, and these effects must be quantified to make an intelligent assessment about the optimum payload capability for maximum score.

Qualitatively, payload capacity will indirectly effect the rated aircraft cost through increased fuselage size, wing area, and empty weight for a given wing loading. In this sense, the sensitivity of rated aircraft cost to payload capacity should be relatively high and will have an adverse effect on the score. Payload will have a relatively shallow effect on total mission time, although it too will be adverse. This is because larger payload at a given wing loading implies a larger vehicle (larger parasite drag) and requires more lift (larger induced drag), so for the fixed propulsion system outlined above, the maximum attainable velocity for a smaller vehicle will be higher. This, of course, translates into shorter mission times.

The quantitative capture of these effects required multiple assumptions. First, the propulsion configuration was frozen as that given in the previous subsection, fixing thrust and thrust lapse. The drag coefficient was assumed to be constant for any given payload, implying a fixed configuration. The wing loading fixed as well. The actual values chosen for the drag coefficients and wing loading at this point were arbitrary; they were used simply to compare different payload capacities and do not represent the actual values selected. The vehicles were all assumed to be of the same configuration, fixing most of the variables in the rated aircraft cost, and all were assumed to have the necessary capacity to complete all six laps. The empty weights of each vehicle were calculated via the methods of the previous subsection with the exception of those of 16 balls of capacity or less. The weights for these vehicles were chosen from expert opinion for comparison purposes only.

The scores were calculated with these assumptions and normalized to the vehicle with the ten-softball capacity. It is important to note that a mistake was made when choosing the payload capacity of the vehicle. The formula the team used to calculate the single flight score included the product of the number of laps completed with the number of balls carried, whereas the actual value was related to the sum of these parameters. This erroneous calculation resulted in the team's choice of 24 balls as the optimum payload capacity. Later on, after initial flight tests and upon revamping figures in the rough draft of the design report, the mistake was discovered and the results recalculated. This resulted in a much shallower trend, with the maximum normalized score located at approximately 20 balls. However, the difference in normalized score between 20 and 24 balls was relatively minor. The normalized flight score trends with payload capacity are given in Figure 4-4 for both the mistaken formulation and the actual calculation via the contest rules. Unfortunately, design had progressed to a point where changing the payload capacity of the aircraft and all associated design parameters would be impossible, so the small decrease in optimum score will have to be taken in stride.

With the payload selected, the estimated weight was found by inserting the payload weight into Equation (2.3). A payload of 24 softballs weighs approximately 9.6 pounds, which yields an estimated takeoff gross weight of 23.7 pounds. The vehicle operating empty weight, found simply by subtracting the payload weight from the gross weight, then comes to 14.1 pounds, for a payload fraction of 0.405.

4.4 Constraint Analysis

Consideration of the point performance constraints imposed on an aircraft is important for selecting the geometry of the vehicle. These constraints are typically mapped on a plot of thrust-to-weight ratio and wing loading. The most general form of the relation between these two powerful parameters can be found from a consideration of the kinetic and potential energies of aircraft maneuvers. The thrust-to-weight ratio and wing loading can be equated via an extensive derivation given in Reference [5]. The result of this derivation is

$$\frac{T_{SL}}{W_{TO}}=\frac{\beta}{a}\left\{\frac{qS}{\beta W_{TO}}\left[K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right)^2+K_2\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right)+C_{D_0}+\frac{R}{qS}\right]+\frac{1}{V}\frac{d}{dt}\left(h+\frac{V^2}{2g_0}\right)\right\} \tag{4.4}$$

where T_{SL}/W_{TO} is the thrust-to-weight ratio, W_{TO}/S is the wing loading, β is the mission segment weight fraction, a is the thrust lapse, K_1 is the induced drag factor, K_2 is the drag offset due to camber, n is the load factor, C_{D0} is the zero-lift drag coefficient, q is the dynamic pressure, R is the additional drag due to flaps or landing gear, h is the altitude, and g_0 is the acceleration at the Earth's surface due to gravity. In this form, this so-called "master equation" can be used to map point-performance constraints on a design plot as a visualization tool for investigation of the design space. Its only requirements are a generic mission model to determine the weight fractions, a thrust lapse model, a parabolic drag polar, and a flight condition (altitude, Mach number, load factor, etc.).

This constraint analysis is simplified with a few assumptions. First, the aircraft is not burning fuel, so its mission segment weight fraction β will always be equal to 1.0. Furthermore, the drag due to camber, K_2, is typically a very small negative number, so it is often ignored in preliminary analyses. Therefore, the only information needed to be estimated at this point is the drag polar and normalized thrust lapse. Recall that the actual thrust lapse was estimated in Equation (2.2), so a normalization of this equation to the system static thrust is sufficient.

The only inputs required for the drag polar is the zero-lift drag coefficient and induced drag factor. The zero-lift drag was assumed to be approximately 0.03 for this preliminary analysis due to past experience. This value is high compared to full-scale aircraft and reflects the effects of lower Reynolds numbers, and thus much higher flow separation, on the aircraft. This effect is often seen to more than offset the lower skin friction drag coefficients seen at these smaller scales. The induced drag factor was estimated based on a moderate Oswald span efficiency factor of 0.75 as will most likely be the case with a rectangular wing and a high aspect ratio. This final estimate, from the familiar induced drag equation, put K_1 at approximately 0.047.

The only true constraint imposed by the contest requirements comes in the form of a takeoff constraint. The aircraft must be able to take off from a paved runway in a ground roll of less than 200 feet. The limiting condition will be that of zero headwind (here the assumption is that the contest directors will always let the aircraft take off into the wind if it is present) and maximum payload, thus maximum weight. Furthermore, high temperatures and altitudes will have an adverse effect on the takeoff performance. Reference [6] states that the highest recorded temperature in the region of Wichita, Kansas is 96 °F. This, combined with the standard pressure altitude of approximately 2,000 feet, represent the worst-case conditions for takeoff. For completeness, takeoff at both standard sea level conditions and Wichita hot-day conditions were considered in the constraint analysis.

The only other constraint lay within the selection of the propulsion system. This, combined with the weight estimation from the previous section, gives an estimate of the upper limit on the sea level static thrust-to-weight ratio at maximum gross weight. At 121.3 ounces of predicted thrust and 23.7 pounds at the predicted maximum takeoff gross weight, the upper limit on thrust-to-weight ratio is 0.32.

Finally, estimates of vehicle top speed are important at the conceptual level, although no constraint is explicitly defined. Therefore, contours of constant maximum speed were placed on the constraint plot to aid in the selection of the design point. The final constraint plot is given in Figure 4-5.

The design space visualized in the above figure pushes the thrust-to-weight ratio and wing loading to higher values for high speeds. Consideration of the max thrust-to-weight ratio available and takeoff performance indicate that the upper right corner of the design space correspond to the maximum speed vehicle that can meet all of the constraints. However, the design point was placed slightly inside of this point because of uncertainty in the thrust, weight, and drag calculations. The design point selected results in an estimated top speed of 70 miles per hour. This is a more efficient operating point for the motor selected when one refers again to Figure 4-2, as the current drawn is lower at this flight condition.

4.5 Preliminary Design Results

The analyses outlined in the previous subsections quantified the necessary values for the preliminary design parameters discussed at the beginning of this section. The results are summarized below in Table 4-1.

Parameter	*Symbol*	*Value*	*Units*
Wing loading	W_{TO}/S	2.6	lbf/ft^2
Thrust-to-weight ratio	T_{SL}/W_{TO}	0.30	--
Payload weight	$W_{payload}$	9.6	lbf
Operating empty weight	W_{empty}	14.1	lbf
Number of cells	n_{cells}	26	--
Brake horsepower	BHP	0.97	HP
Propeller geometry -- pitch	p	16	in
-- diameter	d	14	in

Table 4-1: Preliminary Design Results

Parameter	*Symbol*	*Units*	*Figure(s) of Merit*
Wing loading	W_{TO}/S	lbf/ft^2	performance
Thrust-to-weight ratio	T_{SL}/W_{TO}	--	performance
Payload weight	$W_{payload}$	lbf	Score, RAC*
Operating empty weight	W_{empty}	lbf	RAC
Number of cells	n_{cells}	--	Score, RAC
Brake horsepower	BHP	HP	performance
Propeller geometry -- pitch	p	in	current/voltage required
-- diameter	d	in	performance
*payload weight affects RAC due to increase in fuselage size, weight for larger payloads			

Table 4-2: Preliminary Design Parameters and Associated Figures of Merit

Battery	*Capacity (mAh)*	*Weight (oz)*	*Total Cells*	*Voltage (V)*	*Power (W)*	*Run-Time (min)*
Sanyo 1700SCR	1700	1.89	40	48.0	1920	2.55
Sanyo 2000SCR	2000	2.05	37	44.4	1776	3.00
Sanyo 2500CR	2500	2.84	27	32.4	1296	3.75
Sanyo 3000CR	3000	2.96	26	31.2	1248	4.50

Table 4-3: Static Run Time for Four Different Battery Species

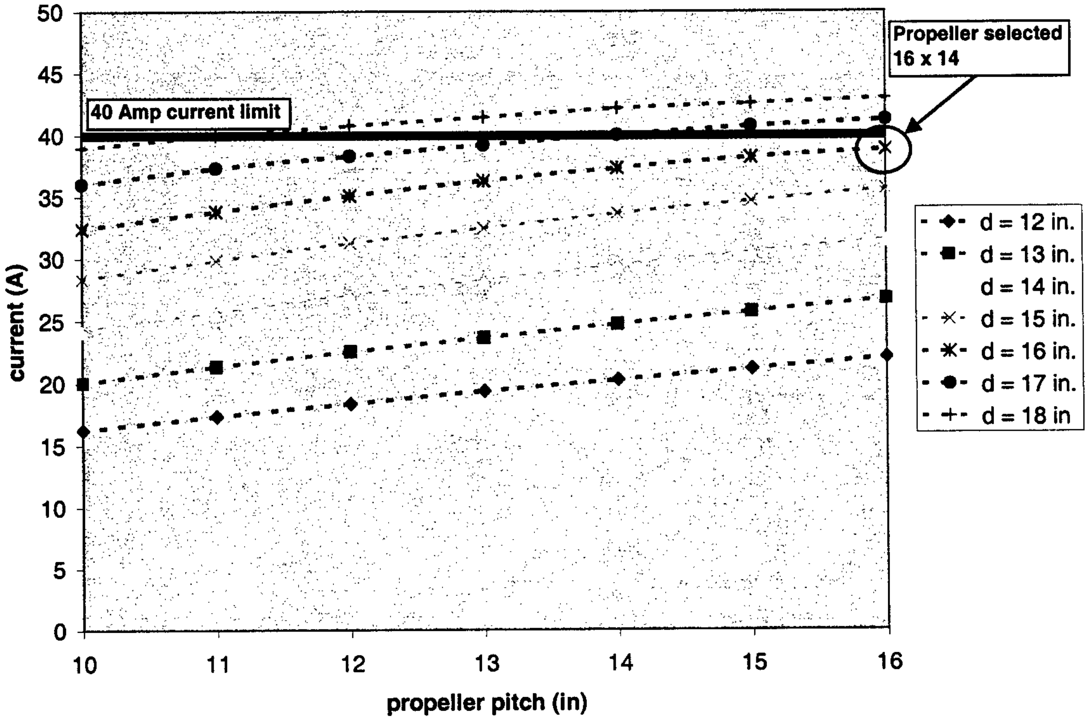


Figure 4-1: Propeller Pitch and Diameter Effects on Static Current Drawn

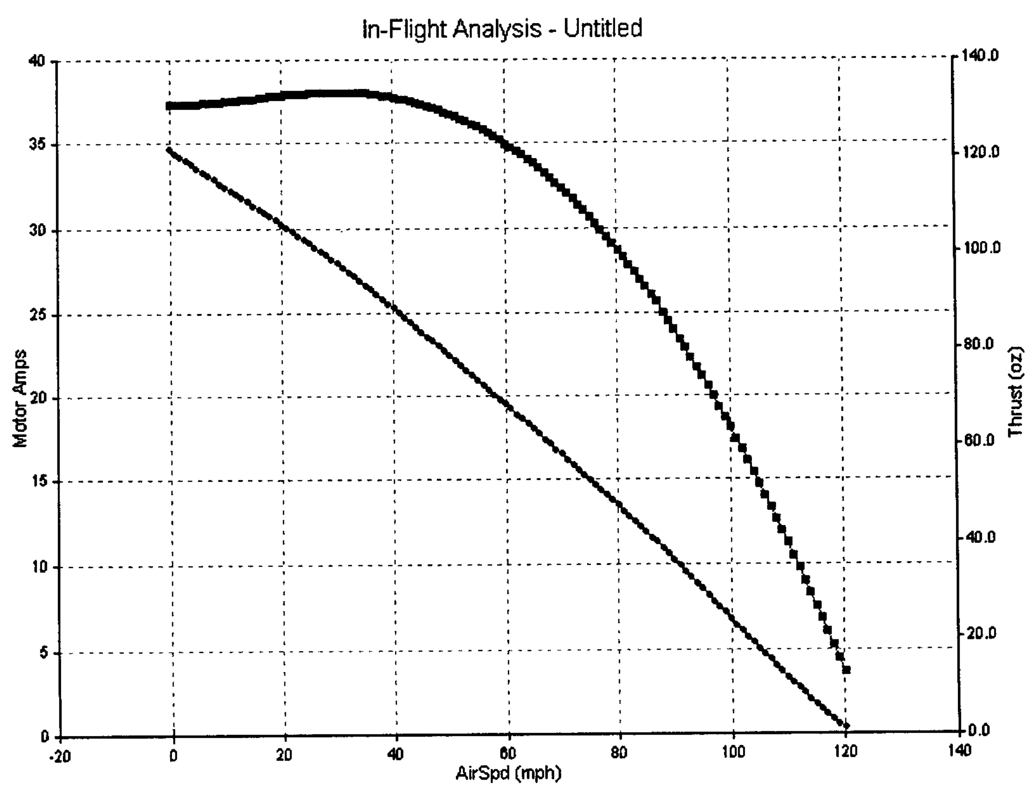


Figure 4-2: Thrust and Current Variation with Airspeed

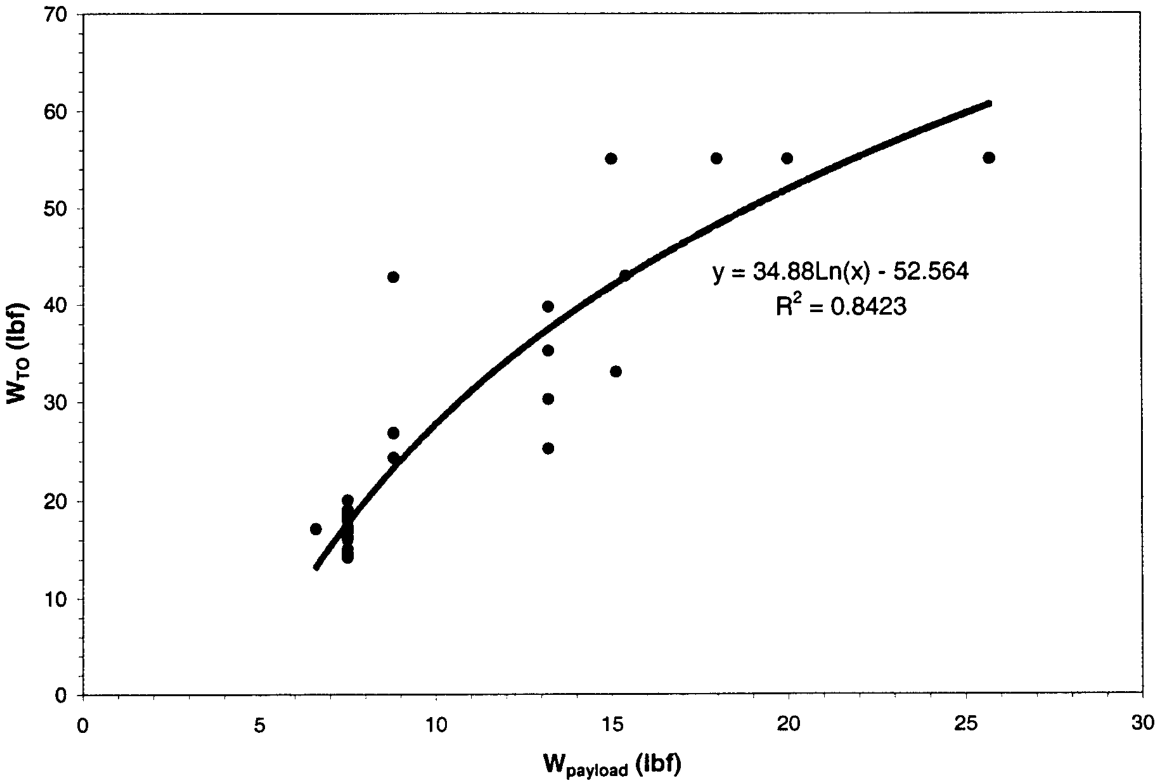


Figure 4-3: Data Points and Regression for Weight Estimation

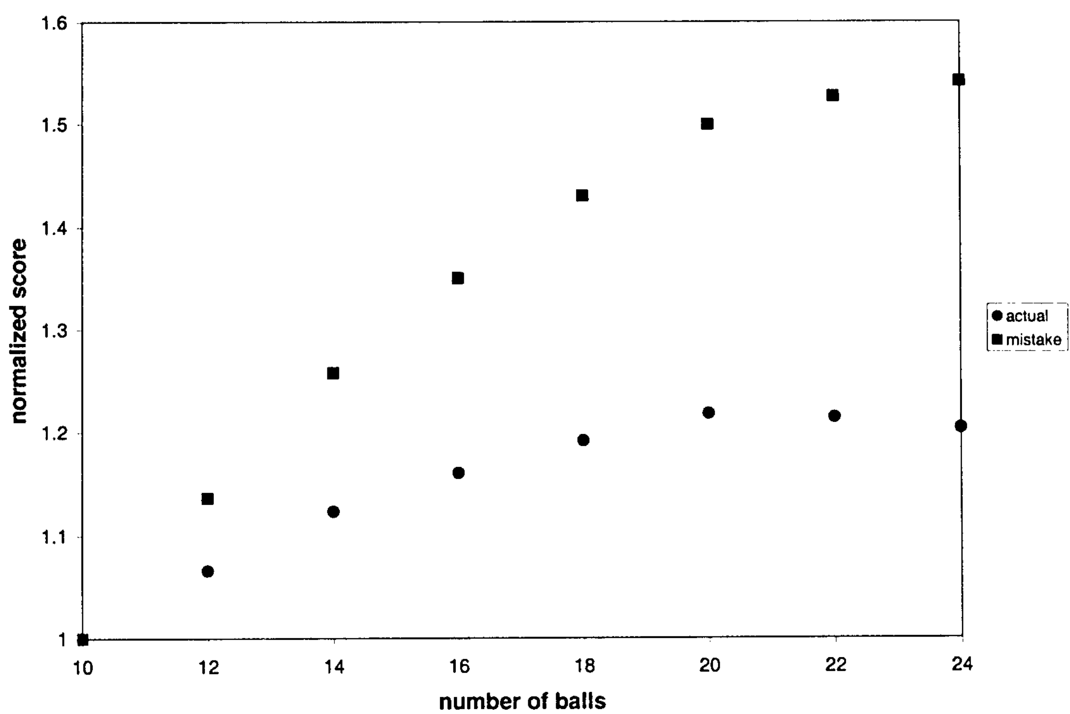


Figure 4-4: Variation of Normalized Score with Payload Capacity

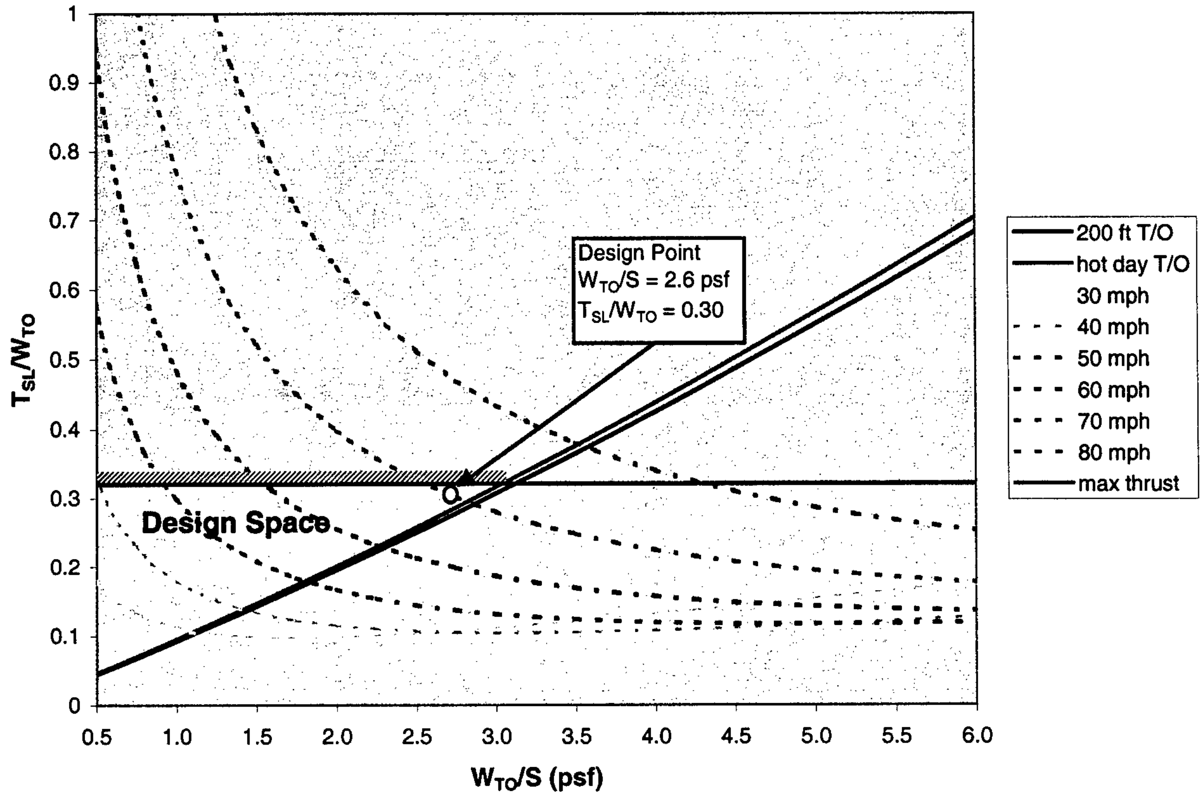


Figure 4-5: Constraint Plot and Design Point Selection

5 Detail Design

The purpose of detail design is to build upon the preliminary design point in order to provide the information at a component level needed to build the aircraft. The major components of the aircraft which require detail design include the wing and tail, which are sized with respect to the design point, the fuselage, whose detail design is influenced mostly by structural and constructional considerations, the propulsion system, and the landing gear.

5.1 Cargo Configuration

At this stage the baseline cargo configuration of 2x12 softballs was revisited. Since the weight of the softballs at the maximum of 24 balls is about 9.6 lbs., which is less than the cargo achieved in previous years, it was assumed that the final design would be able to carry this maximum cargo. However, since softballs are light but of high volume, the challenge became minimizing the frontal and wetted area generated by this cargo. Since scoring assessments call for a faster vehicle, frontal and wetted area should be minimized to reduce drag, but with greater emphasis placed upon reducing the frontal area of the vehicle.

The baseline configuration assumed that the cargo was arranged in two columns of 12 balls each. However, at this stage, a spreadsheet analysis was performed to determine what the optimal cargo configuration would be for the design. Figure 5-1 shows the results of this analysis. The chart shows the difference in wetted and frontal area from the minimum area value, with frontal area given a weighting to appear more important in the analysis. For example, the frontal area is minimized for a single column configuration, while the wetted area is minimized for a more "squarish" configuration of 4x6 or 6x4 softballs. The frontal area is minimized for a configuration of 1x24 balls. This configuration, however, is neither practical nor allowed by the rules. The team decided to keep the 2x12 configuration since it minimized the frontal area within the limits of the rules. This means that there will be a penalty in terms of the wetted area, but the vehicle will have a better ability to penetrate the air for higher speed flight.

5.2 Wings

Beginning with the wing area and aspect ratio specified during preliminary sizing gives the main wing characteristics seen in Table 5-1. The taper, which refers to the ratio of the tip chord to the root chord was left at a value of one to avoid the potential for tip stalling. Tip stall is a common occurrence with small aircraft that have taper, and was witnessed last year in Georgia Tech's DBF design. The wing sweep was set to zero for similar reasons. Wing sweep is primarily useful for structural considerations or for situations where the flow on the wings may become supersonic in some regions. Since this aircraft operates at extremely low speeds, and there is no taper which might warrant wing sweep for ease of construction, the sweep is zero.

The dihedral, or angle between the wing and the horizontal plane will be zero for the prototype and increase to as much as 5° on the final aircraft if the prototype needs more stability in roll. Dihedral generally makes an aircraft more stable, but was not used on the prototype for ease of construction.

Similarly the twist on the prototype will be zero for ease of construction, but will be introduced at a later date if tip stalling becomes a problem. Twist can be difficult to manufacture, but does provide the benefit of making the lift distribution more closely resemble an ellipse [3]. Twist, along with dihedral and the potential for wing tips will be reserved for improving the aircraft once initial flight testing has been conducted to better understand the design.

The selection of an airfoil for the prototype begins with the calculation of the required 2-dimensional coefficient of lift required at the different flight conditions. This particular design has two flight regions of primary concern—cruise loaded, and cruise unloaded. Taking the weight buildup, simple aerodynamic calculations and the approximation that the 3-D C_L is 90% of the 2-D c_l gives the results shown in Figure 5-2.

5.2.1 Airfoil Selection

Once the required lift coefficients were determined the search for a suitable airfoil began. An airfoil was needed that had good low-speed characteristics and matched the required lift characteristics. Airfoils for this research came from Reference [6], and were analyzed using Java Foil [7]. After reviewing many airfoils, a Selig 8052 was selected. This airfoil is seen in Figure 5-2.

The results from the analysis are summarized in Figure 5-2, taken from Java Foil. These results indicate that the wing can achieve the required lift for fully loaded cruise with less than 4° angle of attack. This low angle of attack will allow the aircraft to move through the air without exposing too much of the fuselage to the airstream, and helping to limit the drag because of this. The C_{Lmax} of this particular airfoil for the conditions in question is 1.06, also given by Java Foil.

5.2.2 Control and Lifting Surfaces

The team decided to investigate the use of flaperons instead of conventional flaps and ailerons so that servos could be limited and the complexity of the wing manufacturing kept low. The transmitter's computer allows for combining two channels into one signal, thus "mixing" two functions that can be operated by one servo. Computer mixing eliminates the need for highly complex mechanical systems that would mechanically mix control movements. The freedom to mix any series of control inputs with a computer transmitter makes computer mixing very common on UAVs, where control inputs have to be electronic, but are still rare on full-sized aircraft. Common applications are V-tails, flaperons, elevons, and yaw-to-elevator mixing for p-factor correction. The team chose to use computer mixing to employ flaperons for flapping response and roll control. There are several advantages to mixing the two functions into one control surface, as opposed to the traditional independent flap and aileron functions. As mentioned, this simplifies an otherwise mechanically complex system. The team also wanted the advantages of flapping response across the entire span for greater lift during takeoff. This would minimize

the required wing area for the takeoff constraint in the preliminary wing sizing. The aircraft cost model was also of great importance to the team, and the team was able to eliminate a servo by using computer mixed flaperons. Each flapping surface is 4.5 inches wide, by over 5 ft in length. Regardless of the material chosen to construct the wing, its physical dimensions are very poor for torsional rigidity. If one servo had been chosen for roll control (no flaps),then the most mechanically simple configuration would be to place the servo in the center of the span and apply load to the root of the control surface. With sufficient load, the control surface could twist along its entire length. Such twisting could be extreme with lightweight materials. A better alternative would be to support the surface in the middle of the control surface, requiring one servo per control surface. This configuration would cut moment arms by half. Using the CBEMT program developed by the team, performance predictions for a high-pitch, low diameter propeller show low thrust at low speed, but relatively higher thrust at higher speeds as compared to low pitch, high diameter propellers. This tradeoff meant that the team could expect low thrust for takeoff. Graphing the estimations at the time of wing design showed insufficient lift for the airspeed expected at 200ft ground roll. It was then concluded that flaps would be required to raise the C_{Lmax} to give sufficient lift for takeoff.

5.3 Empennage

The baseline tail configuration is the conventional vertical surface with a low-set horizontal surface. At this stage of the design, other options for the tail were considered to see if any might provide some benefit to the design. An H-tail was rejected off-hand because it would require complex construction and controls for the two vertical surfaces. A V-tail was also rejected because the controls system would require mixing which was beyond the scope of the team's experience. A T-tail would allow for a shorter vertical tail because of the end-plate effect that it has [3], but it would also require additional strengthening in the vertical surface. This position would also complicate construction in that the control wires would have to be run through the vertical surface. Lastly, a T-tail may have difficulty in stall recovery if it is blanketed by the flow from the wing. After considering these options, the team decided to continue with a conventional tail configuration.

The sizing of the tail is much less performance driven than the wing. The tail must be of sufficient size to stabilize the aircraft, and its control surfaces large enough to provide yaw and pitch for control of the whole aircraft. Especially important is the size of the elevator which must rotate the aircraft on takeoff. The design of the tail apart from detail controls analyses involves the sizing of the tail based upon "historical" volume coefficients and shape trends. For the purposes of the prototype, tail sizes were selected based upon Raymer's trends for general aviation aircraft [3]. The intent with this first sizing was that the tail be approximately what the aircraft requires, with corrections made to the final aircraft. On the prototype, several fittings were available to vary the moment arm length, and tails are small enough that new ones can be constructed quickly.

The estimated tail sizes from [3] are shown in Figure 5-3 for both the vertical and horizontal tails. The airfoil selected for the empennage was a NACA 0009 because it is symmetric and very thin, reducing its profile drag.

The final sizing of the tail was done using graphs from Figure 5-3. These show the calculated sizes needed for the tail for different lengths from the wing center of pressure. Since the aircraft is already long and slender due to the 2x12 cargo configuration, it was desirable to keep the moment arm as short as possible. Further, there are cost penalties associated with the fuselage length, whereas there is a constant score for control surfaces of different sizes. Keeping the moment arm to only 16 inches gives a horizontal tail surface area of 1.35 $ft.^2$, and a vertical tail surface area of .74 $ft.^2$. Standard taper and tail sweep were also selected based upon Raymer [3], and the complete dimensions of the empennage are summarized in Table 5-2.

5.4 Landing Gear

During conceptual and preliminary design phases, it was assumed that fixed landing gear would be used, as is most common with aircraft of this type. However, various types of retractable landing gear are available, and so the team investigated the possibility of using this type of landing gear.

Basing the investigation on pneumatically retracted gear, the team discovered that fixed and retractable landing gear weigh roughly the same, with only a little extra weight needed for the structure and discharge tank of the retractable gear. Using retractable gear requires a strengthening of the wing because of the necessary wheel-wells. However, it was determined that the area needed to store the wing would not damage the integrity of the wing.

The benefits of using retractable gear are very large in terms of parasite drag. Based upon experience on the team, the landing gear can contribute up to 45% of the total parasite drag at cruise speeds. Because of this large impact, the team decided to proceed with retractable landing gear.

5.5 Fuselage Structure

The internal structure of the aircraft was decided upon by structural comparison of configuration alternatives. In Conceptual Design, the strength of the fuselage was considered as important criteria in deciding upon the most favorable aircraft configuration. The payload insertion/extraction was originally thought to be done by tilting the aircraft and allowing the balls to roll out. This meant that the cargo area could not be interrupted by the wing's structural member. As seen in Figure 5-4, a high wing aircraft with payload side-by-side would have a bulge in the top of the aircraft in which to house the radio gear and battery pack. In the low-wing configuration, there is a similar case with equal amount of space. By spacing the balls just slightly (2 inches in this case), the calculated moment of inertia, I, as shown is much higher for little penalty in frontal area. Even without adding in the strength of the lower area, the moment of inertia is nearly twice that as the two previous alternatives. Because most of the load on a solid is carried on the surface of the structure, this analysis was sufficient argument to warrant the building of the side frames in this manner.

This configuration also allowed much greater freedom in the placement of components, such as the battery and the motor. These are both located between the side frames at the same elevation as the payload. The first two alternatives in the figure would not allow such a configuration. If either of the first two had been chosen, the motor would have to be mounted in the nose cone. Also, the battery would probably have to have some unusually flat layout, or divided into two smaller packs (fore and aft of the wing).

5.6 Component and Architecture Selection

As shown in Figure 5-14, the aircraft features one servo per control surface, and uses wing mixing to trigger the pneumatic retract valve. The team chose JR 8231 digital servos for their superior accuracy and holding power. The elevator servo is mounted only 6 inches from the elevator to minimize the length of the control linkage. The elevator servo horn is attached on the top of the elevator such that it is in tension during "up" elevator. It can be noted in the figure that the elevator employs outboard "diggers" to offload some of the force on the servo. These counterbalances also reduce the amount of twist in the soft balsa elevator when under load. The rudder servo is centrally mounted at the rear of the cargo bay. A tube-in-sleeve linkage ties this servo to the rudder, and pull/pull cables link the servo to the nose gear in the front of the aircraft. Flaperon servos are mounted in the center (left to right) of the control surface. They were mounted in the center for the same twisting argument at the elevator.

The motor is set back 18 inches from the propeller to reduce the pitching moment to rotate the aircraft. By doing so, the empennage area and tail moment can be greatly reduced. A shorter aircraft was preferred because of the cost of fuselage length in the RAC. There were concerns that the use of such a long shaft would create vibration problems and potentially lead to catastrophic drive failure. The planned construction of the shaft was rather simple, and the team simply built one and tested it with a 20x8Mater Airscrew propeller at 5,000 RPM. Results were surprisingly smooth.

Although not shown in this figure, Figure 5-13 shows the hub detail employed on the aircraft. The shaft is supported at the propeller by a ball bearing, but torque is delivered to the propeller by a thin clutch bearing. This allows the propeller to "windmill" freely when not under power. Potential drag reductions were never calculated or measured, but the team knew that if there were a force wanting to turn the propeller, then resistance to this force would be energy lost.

The motor is mounted to the airframe by a milled 6061 T6 Aluminum motor mount, which has ribs for added cooling. Aluminum has excellent heat transfer properties and is much less dense than many other metals. The drive shaft and coupling were also turned from 6061 T6 Aluminum.

The propulsion battery pack is located between the two side frames in a large bay over the wing. The length of the bay between fuselage formers was chosen to allow movement of the battery pack to offset any errors in calculating the center of gravity of the airframe. Actual placement of the battery pack was fully forward, which worked out nicely because leads to the motor were kept very short. The battery pack can be accessed through the same hatch as the cargo, eliminating the need for extra access panels or hatches.

The retractable gear's air tank is centrally located in the last bay of the cargo area. When fully charged, this tank will cycle the three gear about 5 times. Only three cycles are required for a total mission. The actuation of the gear is done through mechanical mixing of the flaperon servos. When aileron inputs are given to the aircraft, they are mixed out by use of a swivel/pivot post. When the control surfaces operate in the same direction, they swing the pivot post to push a linkage to the gear actuator valve. This mechanical mixing eliminates the need for a retract servo, which is scored in the RAC. The team is currently investigating a similar set-up for mixing rudder inputs to aileron inputs.

5.7 Aircraft Stability and Control

During the design process, care was taken to make sure the center of gravity of the entire aircraft is in the correct location for the stability purposes. A spreadsheet was developed to keep track of all the component weights along as their location. The battery packs have some freedom to move to allow for transfer of the center of gravity of the aircraft. The center of gravity point on the aircraft was chosen to be the midpoint of the fuselage, which is also the quarter chord point of the wing, since the payload and fuselage are balanced about this point. A summary of the aircraft component weights is shown in Figure 5-5 and the aircraft weight balance is seen in Figure 5-6.

In terms of static stability, roll is the most controllable since the wings can be modified in a straightforward manner to increase or decrease roll stability (dihedral). However, yaw and pitch are stability are less clear and so an analysis was performed on each of these characteristics. Stability in yaw was estimated using a C_n vs. β chart (Figure 5-7). The positive slope of this chart shows that for any perturbation in yaw angle, the aircraft is stable and will not become uncontrollable. A similar chart was created for pitch, as seen in Figure 5-8, where the negative slope of the general trend indicates stability.

5.7.1 Handling Qualities

An initial static stability analysis was performed to determine the stability of the aircraft during steady, level flight. From this analysis, the aircraft was determined to be stable in both the longitudinal and lateral directions. Based on the aircraft configuration resembling historical aircraft, no analysis was done of the longitudinal dynamic stability of the long and short period modes or the lateral dynamic stability of the Dutch roll and spiral modes of the aircraft. In the unlikely case that the frequency or period of any of these modes is deemed unacceptable, the empennage design will be augmented to correct the problem.

5.8 Performance

Once this initial design point was set, a determination could be made of what the optimal cruising speed would be based upon these initial assumptions. It was estimated that the aircraft must be able to cruise at approximately 36 fps (25 mph) in order to complete the entire mission in under 10 minutes. As shown in Figure 5-9, the minimum speed for total drag is 50 fps. This corresponds to maximum L/D and is the best cruising speed for the aircraft. Based upon this cruising speed, the cruising segments of the

the flight portions of the competition should be completed in 5 minutes and 32 seconds. This means that the team will have 4 and a half minutes to complete the two loading and unloading processes

5.8.1 Takeoff Performance

Takeoff performance is shown in Figure 5-10. To takeoff within the required 200 ft., at least 2 lbs. of thrust will be needed. For increased average thrust, the takeoff performance increases dramatically.

5.8.2 G-Loading

Using the deflection formula for loaded beams, the wing spar was modeled in a spreadsheet. For level flight (the 1-g case) the deflection of the spar will be 1.05 inches. The loading case for 4-g's is shown in Figure 5-11. For a balsa and composite spar, the maximum deflection allowable is 2 inches, and so the 4-g loading is the limiting case for the fully loaded aircraft. This means that the aircraft will be able to survive the required three-point loading test, which estimates a 2-g loading case.

5.9 Rated Aircraft Cost

The RAC for the final aircraft is shown in Figure 5-12. These numbers reflect a battery pack of 5 lbs and a total empty weight (including the battery) of 15 lbs. The other major cost contributors are the two empennage surfaces and the five controller servos.

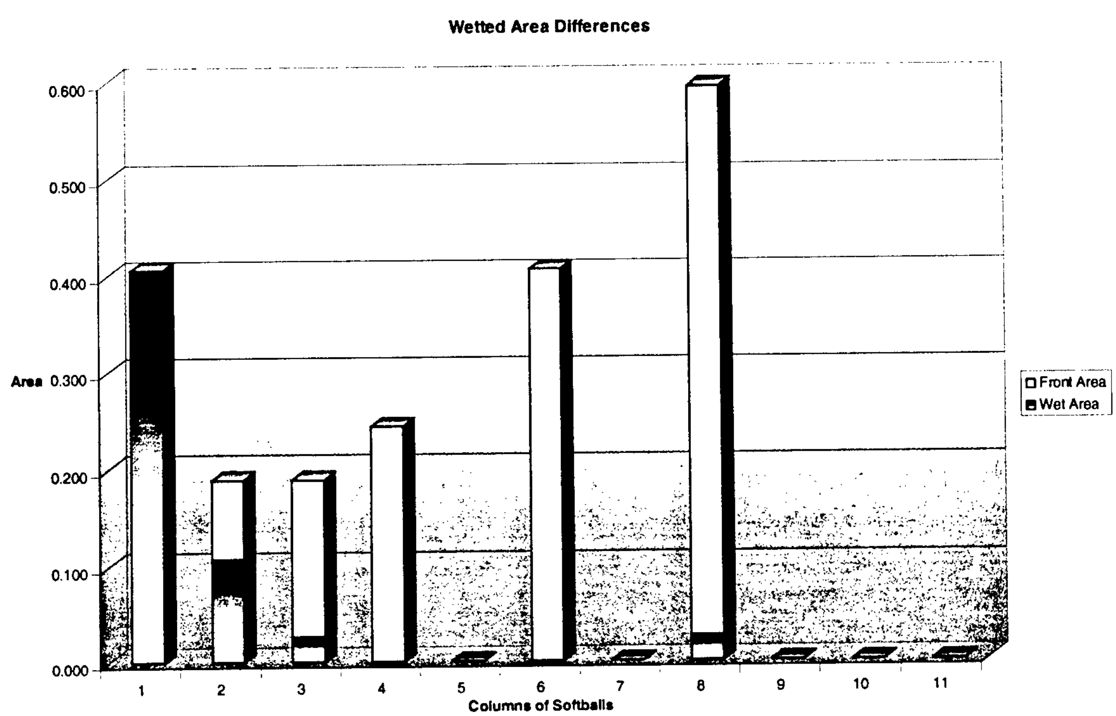


Figure 5-1: Area Analysis for Varying Cargo Configurations

	ft.	in.
Wing Area (S)	10.07	120.83
Wingspan (b)	10.03	120.42
c0 (root chord)	1.00	12.04
ct (tip chord)	1.00	12.04

	Loaded	Empty
2-D Cl	0.5478823	0.33916524

Table 5-1: Wing Dimensions and Required Characteristics

Horizontal Tail		Vertical Tail	
Λ	0 °	**Λ**	35 °
Arm	3.5 ft.	**Arm**	3.5 ft.
S	1.35 ft.^2	**S**	0.74 ft.^2
AR	6	**AR**	2
b	2.84 ft.	**b**	1.22 ft.
		taper	0.6
		c0	1.521751 ft.
		ct	0.91305 ft.

Table 5-2: Tail Size

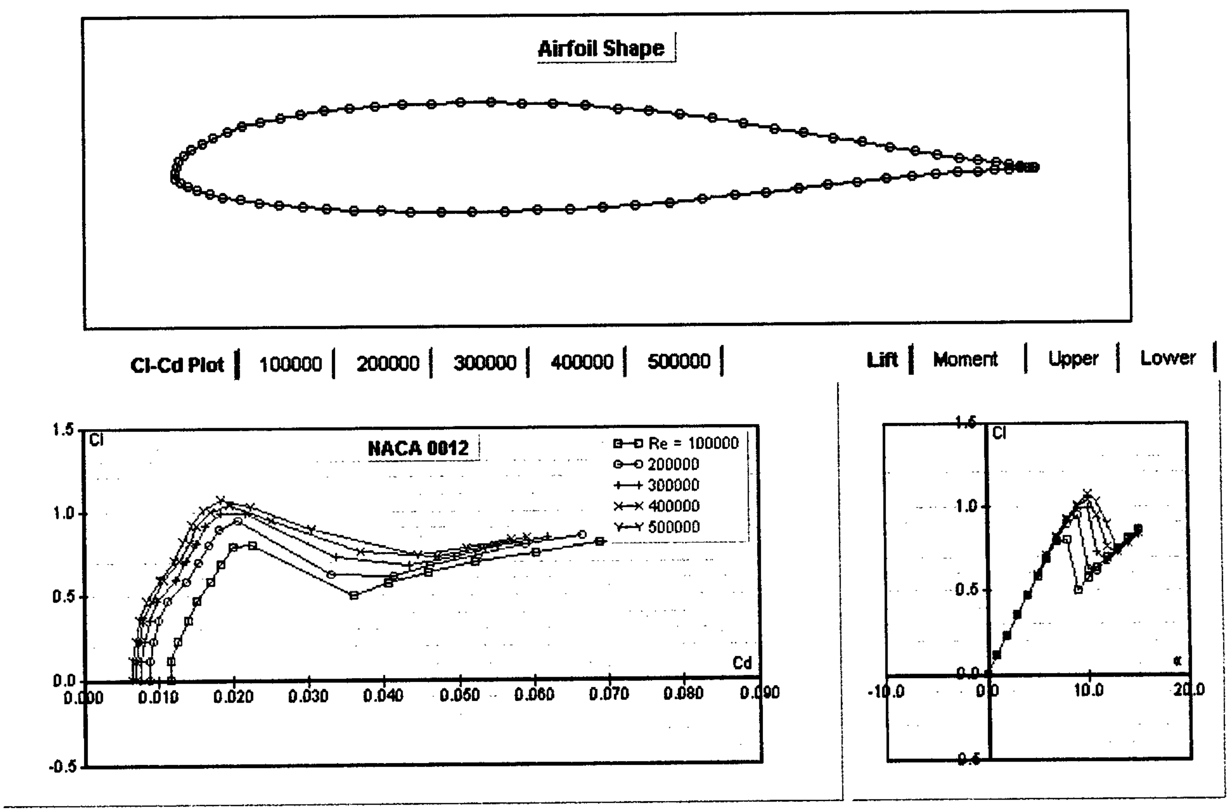
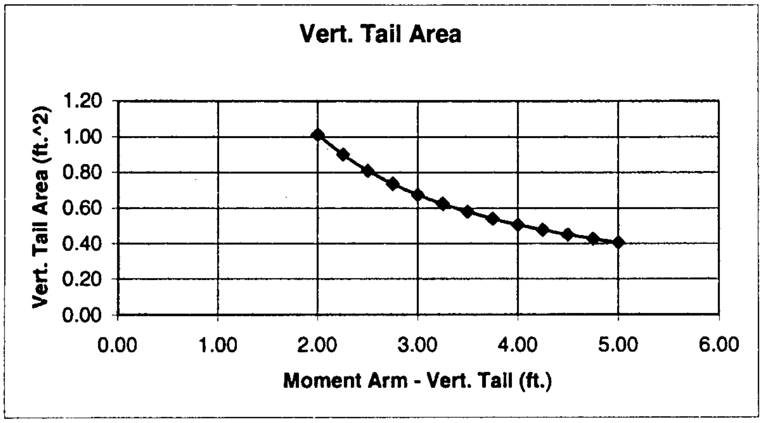


Figure 5-2: Java Foil Analysis for Selig 8052



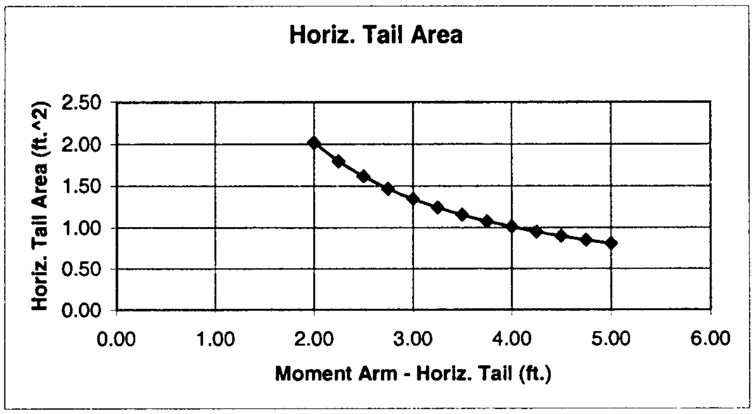
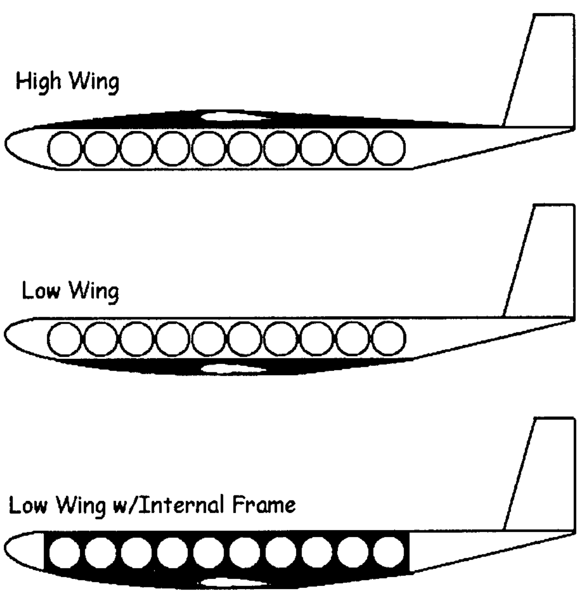


Figure 5-3: Tail Sizing Trades



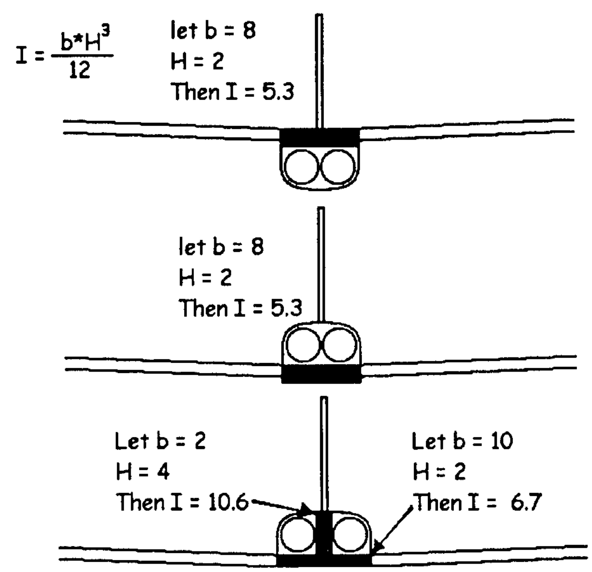


Figure 5-4: Fuselage Structure Considerations

Weight Breakdown

Fuselage	Weight (oz)	Empennage	Weight (oz)
Fuselage (w/o servos)	47.819	**Vertical**	
Engine Mount	1.650	Foam	0.345
Screws & Washers	0.575	Balsa	0.428
Aluminum Bar	5.370	Glue	0.086
		Monocoat	0.311
Nose Wheel		**Horizontal**	
Wheel	0.285	Foam	0.938
Strut	0.910	Balsa	0.326
Retractable Mechanism	3.910	Glue	0.065
Screws and Washers	0.175	Monocoat	0.528
Wings (without servos)		**Propulsion**	
Starboard Wing	19.560	Motor (Cobalt 90)	30.065
Portside Wing	19.290	Shaft	2.095
		Propeller	2.080
Main Landing Gear (Starboard)		Propeller Nut	1.330
Wheel (w/spring)	0.650		
Strut	0.840		
Retractable Mechanism	3.850	**Airframe Weight**	**156.12 oz**
Screws & Washers	0.190		**9.76 lbs**
		Empty Flying Weight	**235.80 oz**
Flight Systems			**14.74 lbs**
JR 8231	1.730	**Total Gross Weight**	**24.14 lbs**

Figure 5-5: Aircraft Weights

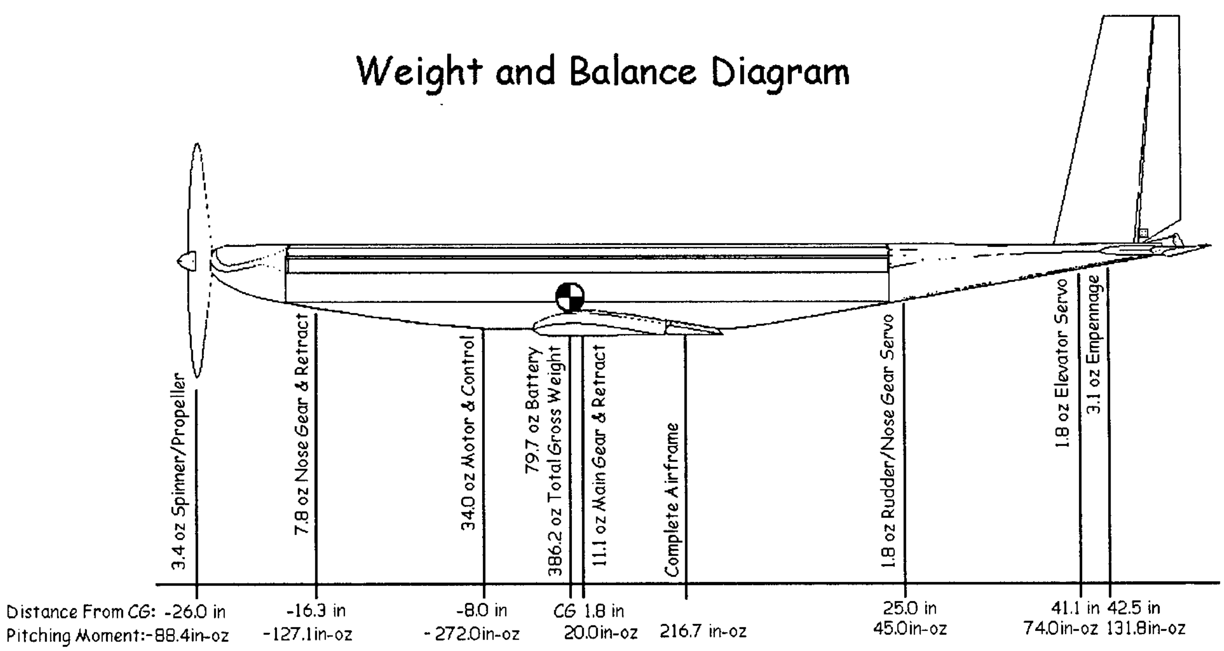


Figure 5-6: Aircraft Weight and Balance

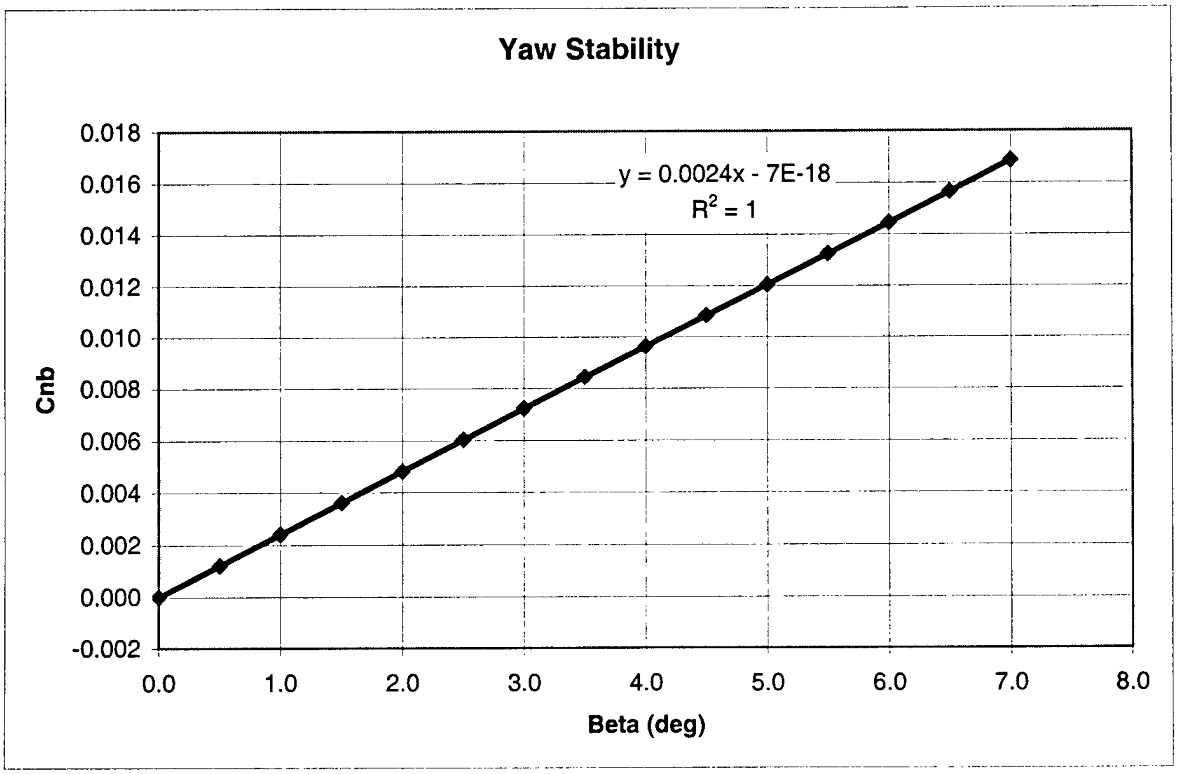


Figure 5-7: Yaw Stability

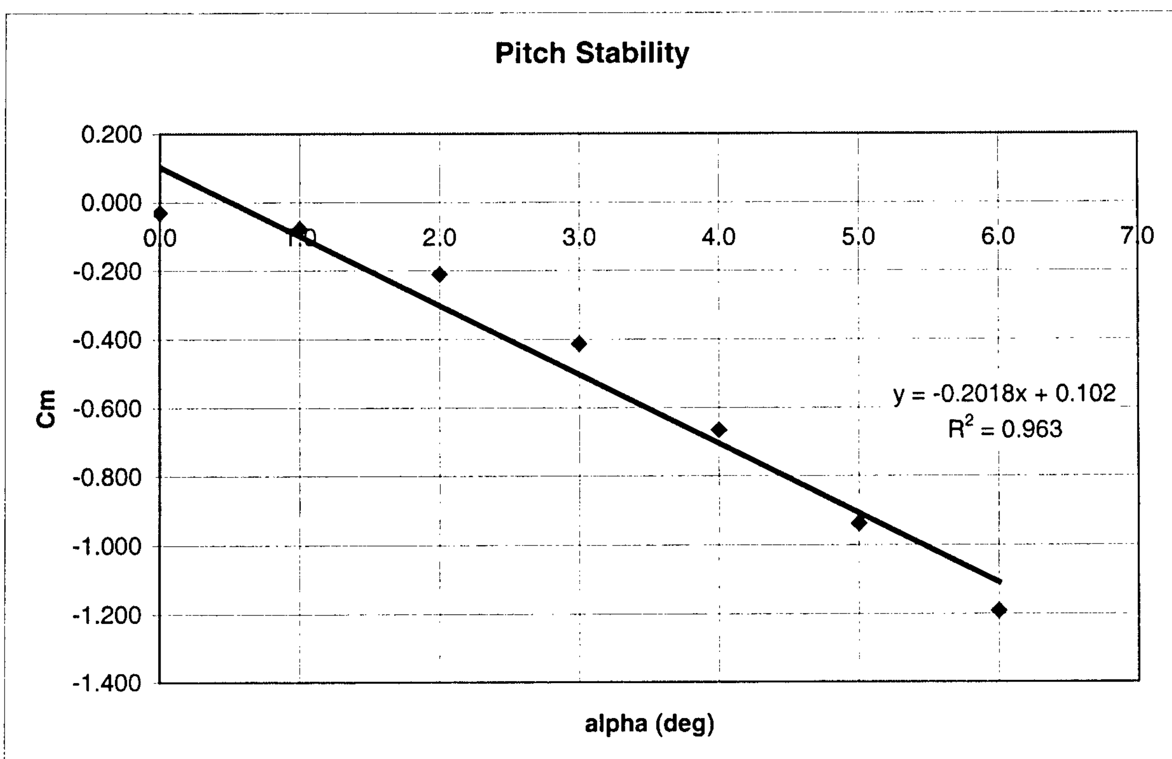


Figure 5-8: Pitch Stability

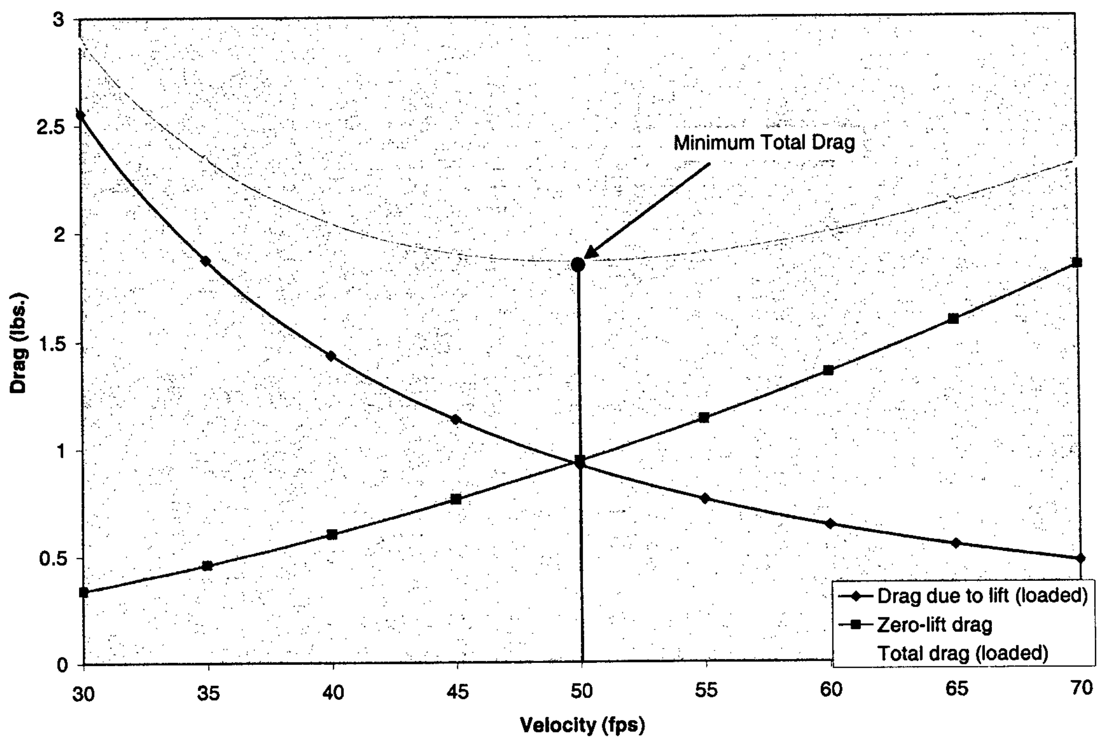


Figure 5-9: Aerodynamic Performance for Loaded Sortie

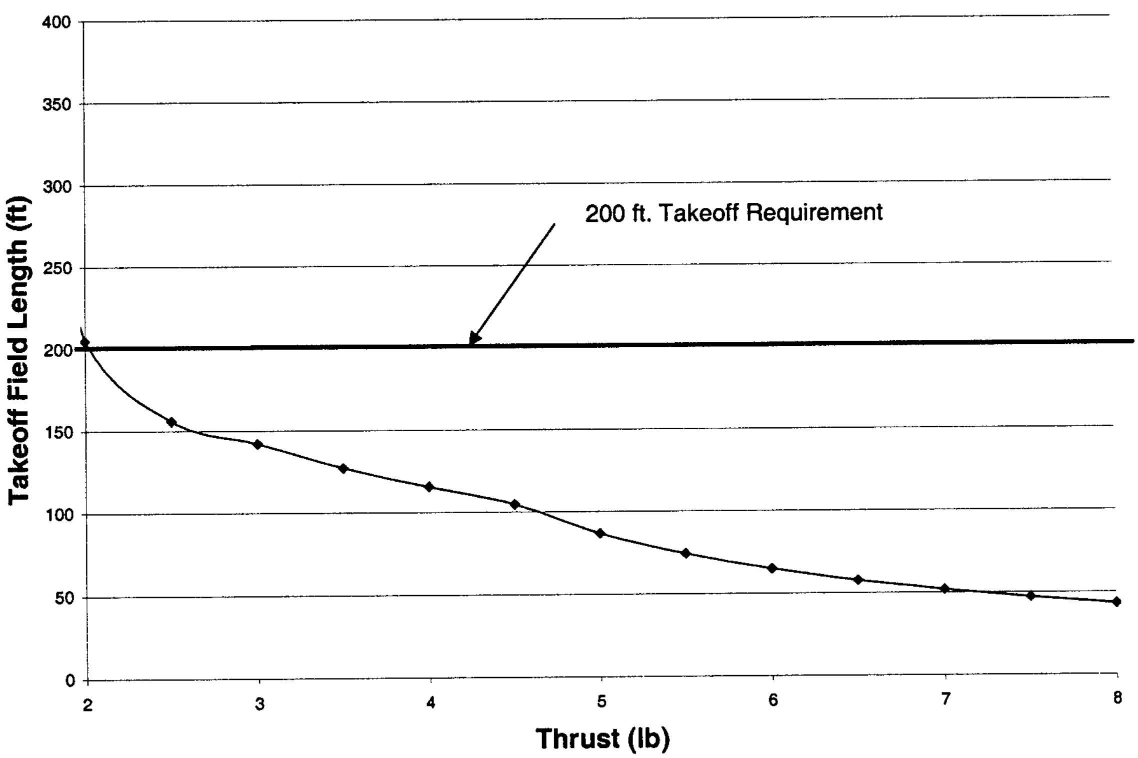


Figure 5-10: Takeoff Field Length for Varying Thrust

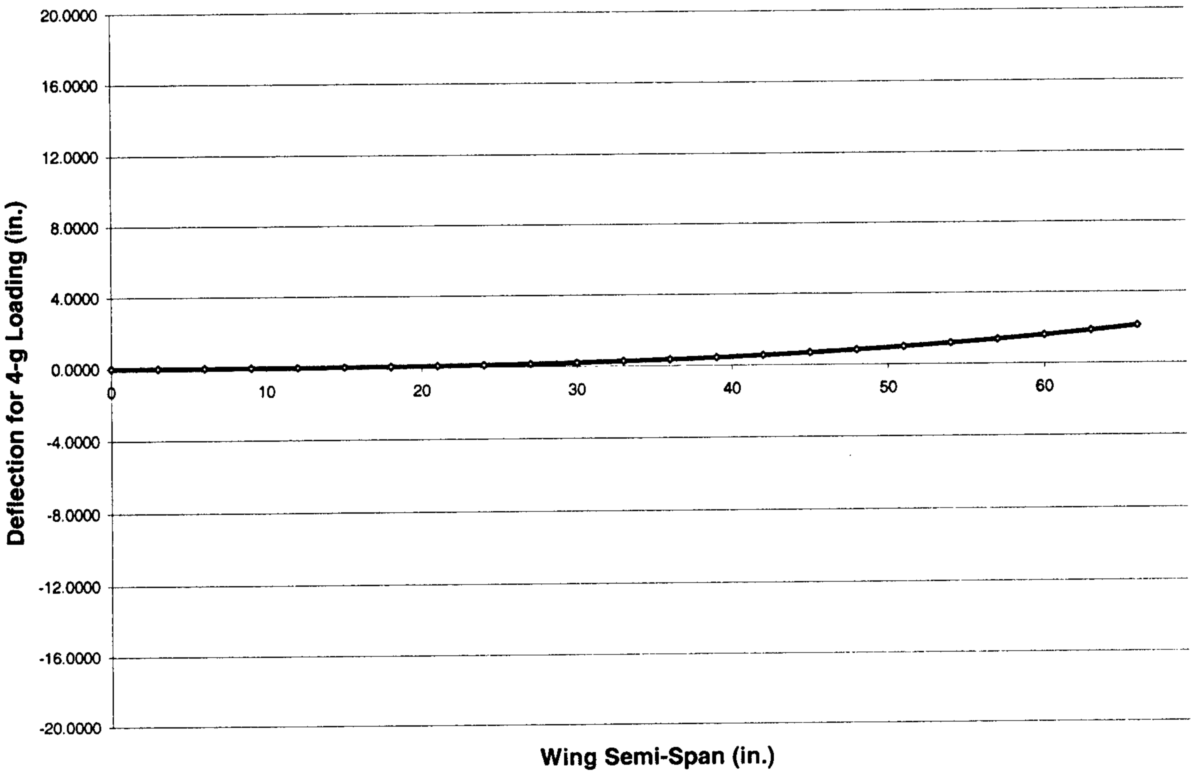


Figure 5-11: Wing Deflection for Maximum Loading Case

Georgia Tech - DBF 2001/2002 - Team Buzz Light
Rated Aircraft Cost

Description	Inputs	Hour Multiplier	Number of Hours: Component	Number of Hours: Total	Cost Multiplier	RAC
Manufacturer's Empty Weight (MEW)						
Empty Weight	15.000 lbs				$100.00	$1,500.00
Rated Engine Power (REP)						
Number of Motors	1 motor				$1,500.00	$7,500.00
Battery Weight	5.000 lbs					
Manufacuring Man Hours (MFHR)						
WBS Wing						
Span	10.00 feet	8	80			
Max Exposed Wing Chord	1.00 foot	8	8	100.00	$20.00	$2,000.00
Number of Control Surfaces	4 surfaces	3	12			
WBS Fuselage						
Body Maximum Length	6.00 feet	10	60.00	60.00	$20.00	$1,200.00
WBS Empennage						
Number of vertical surfaces with no active control	0 surfaces	5	0			
Number of vertical surfaces with active control	1 surface	10	10	20.00	$20.00	$400.00
Number of horizontal surfaces (span <25% of wing span)	1 surface	10	10			
WBS Flight Systems						
# Servos/Controllers	5 servos	5	25	25.00	$20.00	$500.00
WBS Propulsion System						
Number of motors	1 motor	5	5	10.00	$20.00	$200.00
Number of propellers or fans	1 propeller	5	5			
			Rated Aircraft Cost, $ (Thousands) =			**$13.30**

Figure 5-12: Rated Aircraft Cost Worksheet

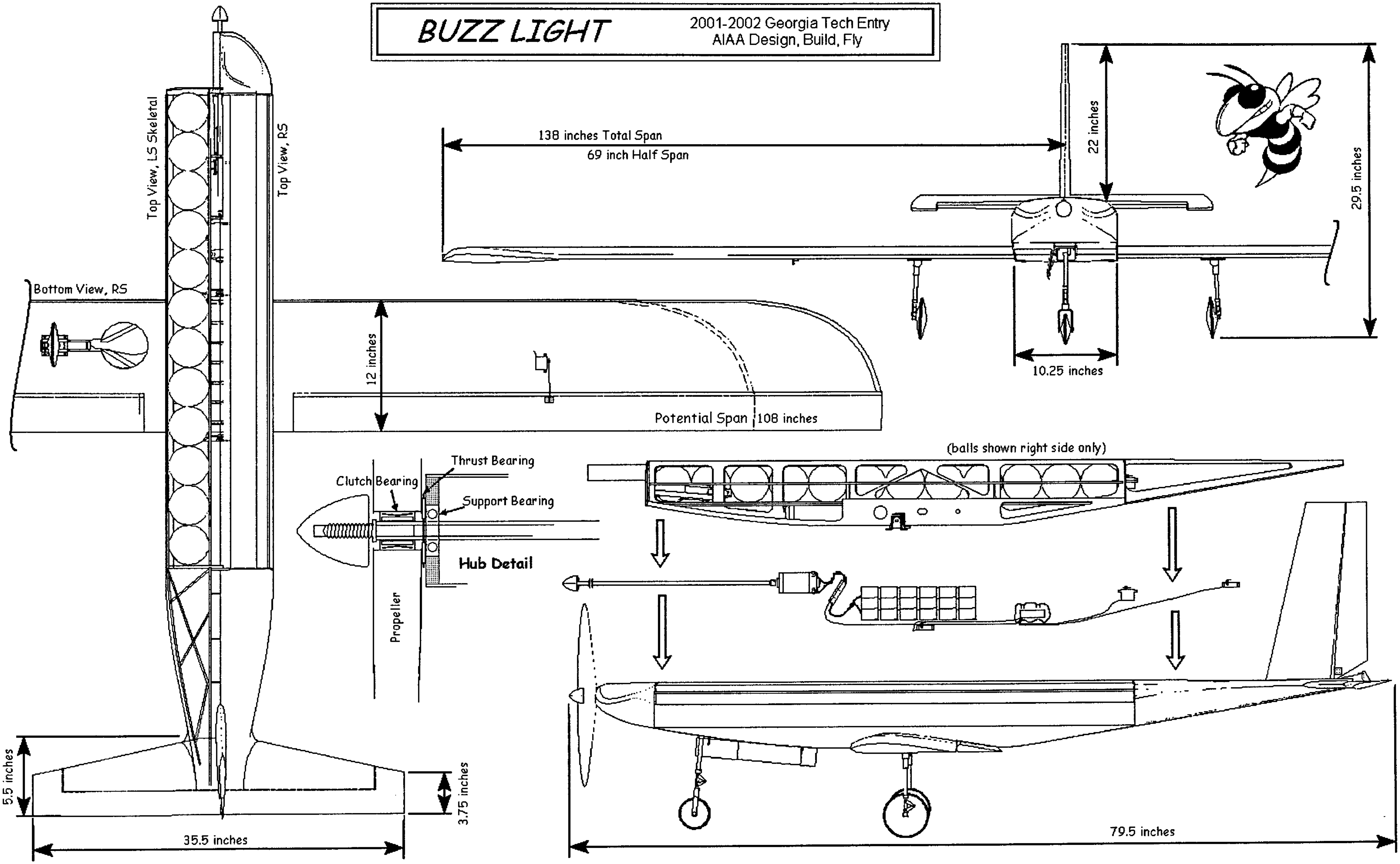


Figure 5-13: Three View

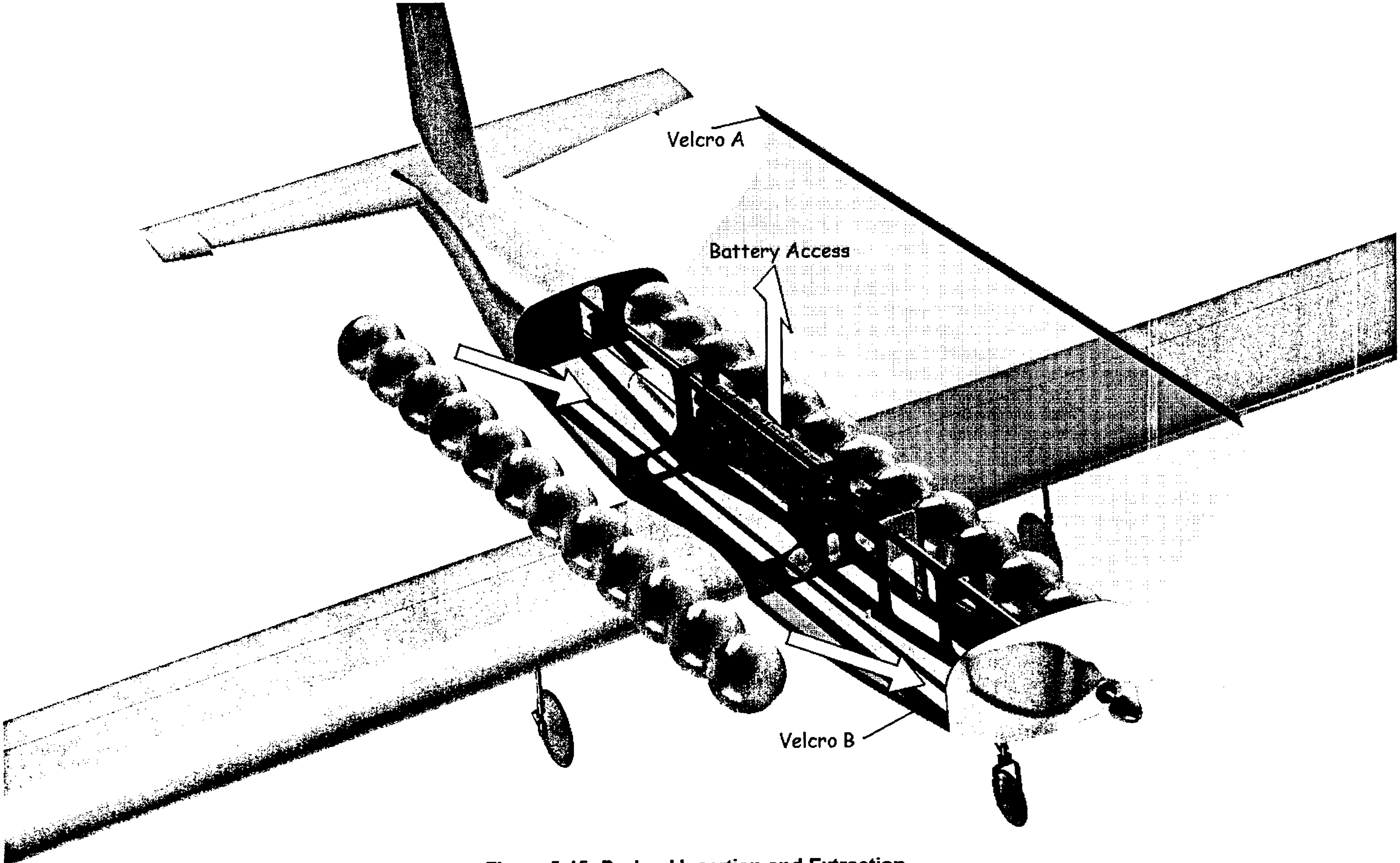


Figure 5-15: Payload Insertion and Extraction

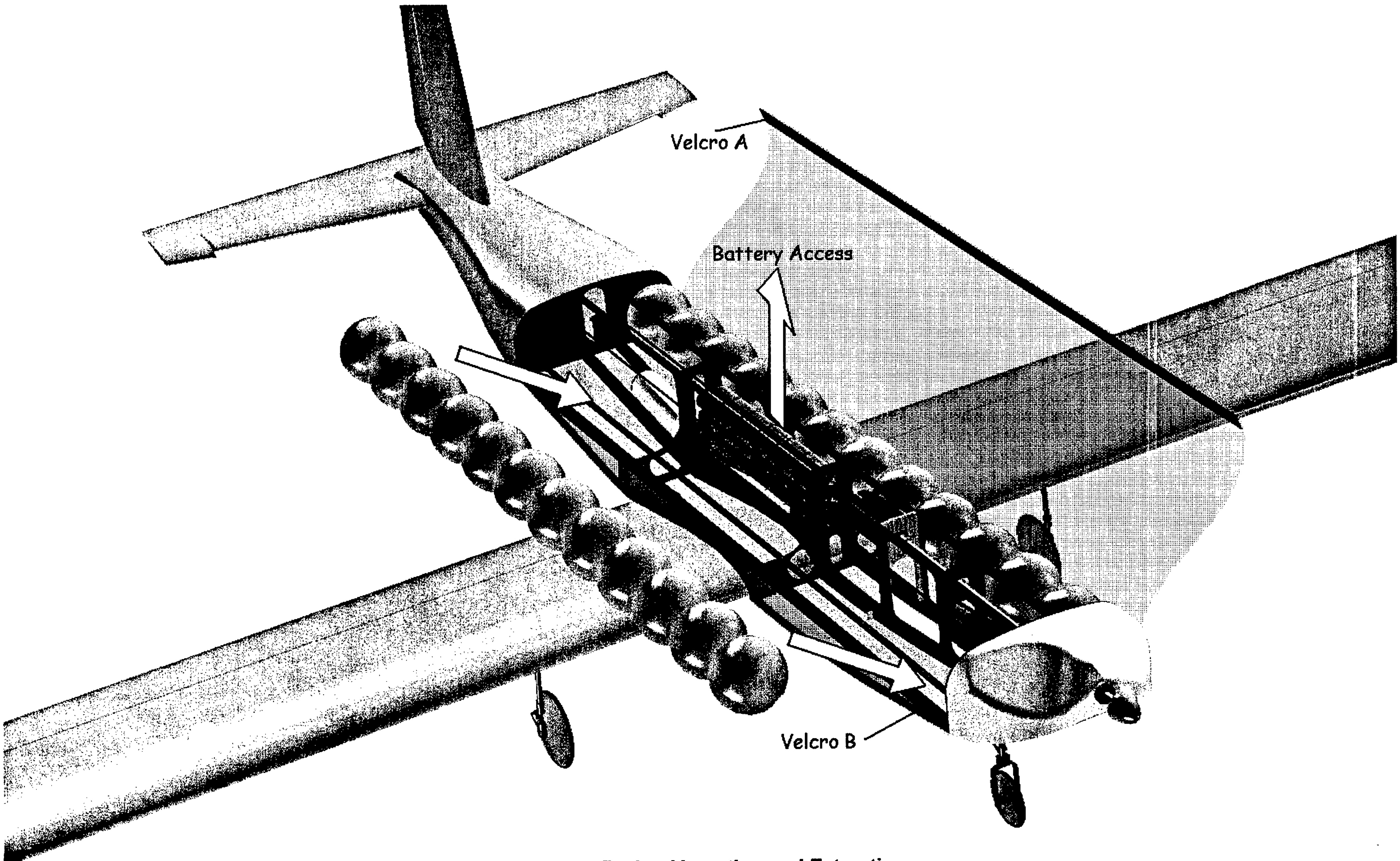


Figure 5-15: Payload Insertion and Extraction

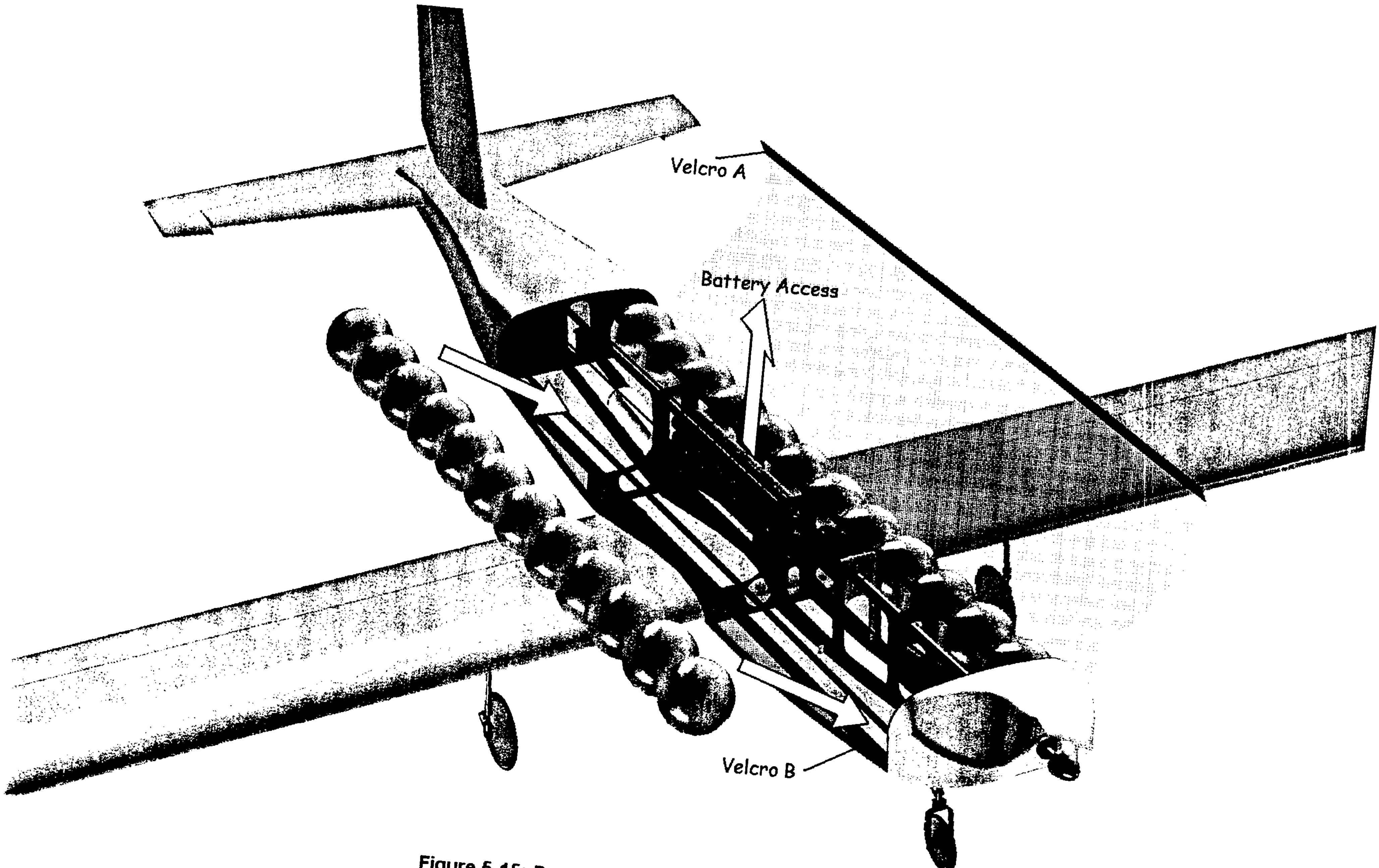


Figure 5-15: Payload Insertion and Extraction

These common practices found with light UAV construction are not necessarily the lightest, strongest structures. More advanced constructions, such as those using advanced composites, have been shown to have much higher strength and lower weight than more conventional constructions, and worth investigation. Most aircraft found in the RC hobby industry are made of balsa, poplar, and polystyrene, which have been used with great success. Careful arrangement of these materials may have a great effect on the overall strength of the structure. From analysis of RC competitions in the past, it was concluded that if done properly, results with more conventional materials can be comparable to that of advanced constructions, saving time and money. It was noted that limiting studies to these common materials would greatly simplify the task of the structures group and ensure that successful results were likely. Preliminary estimates of alternative or more advanced constructions would involve more time, skill, and money than more conventional constructions. It was argued that if these alternative constructions were truly superior, then they would be more prevalent in the RC hobby industry. These arguments led to the documentation of the primary Figures of Merit that the team felt were important to this task.

One of the primary Figures of Merit was knowledge. Without knowledge of material properties and structures, one can not successfully engineer a vehicle. Unfortunately, the team was composed mostly of undergraduates with little or no experience in these kinds of analysis. Although mastery of structural analysis was not expected, extensive efforts were made to define the proposed configuration in terms that the team could use to analyze the distribution of load.

Every component proposed in a construction was inspected for its contribution to overall strength. Studies of methods of adhesion were also conducted to see if the bond between two materials would fail at the contact face or if the adhesive itself failed. In the case of carbon fiber laminae, it was found that often times the medium joining the fibers would fail before the fiber itself. Carbon fiber's tensile strength (ultimate) is ten orders of magnitude higher than that of cold rolled steel, but the shear strength is weak by comparison. As long as the load on such a material was intended to be applied in tension, the team could expect to obtain good results. The other structural members were arranged as creatively to support the aircraft under normal flight loads and the three point load test. Through experience, study, and experimentation, the team was able to obtain satisfactory results by the time of this report.

The compilation of the report was done simultaneously with the design. Conceptual design was written at the end of the conceptual design phase, preliminary design was written at the end of the preliminary design phase, etc. The construction of the prototype vehicle began with the beginning of the detailed design phase. The experimental manufacturing methods were most prevalent at this point. This recursive approach to manufacturing ensured cohesion between the optimal design points obtained in theory, and what constructions were feasible. The completion of prototype components had to be done as quickly as possible after the theoretical design was defined in order to test the vehicle. Testing of the prototype had to be complete in time for the team to analyze test data, make changes to the design, build a final vehicle, and test (for validation) the final design.

6.1 Fuselage

A prototype aircraft was planned before the beginning of preliminary design. Having discussed the difficulties of performance prediction, the team decided that sufficient time needed to be made to produce a fully functional vehicle and make changes to it in time to completely build a second aircraft before the competition. Material selection is discussed below. Actual construction began as soon as final decisions were made about the design.

Construction planning for the prototype was fairly trivial. Material selection was made early to support quick, easy construction using cheap, common materials. Once again, Pugh diagrams were used to compare alternative materials. The team limited themselves to the materials commonly used for UAV wing construction, with the exception of Polycarbonate, which is known to have high strength, very malleable, and cuts easily. Any other materials were considered non-conventional and experimental. The Diagram is shown below:

Some properties not evaluated in the Pugh Diagram were adhesion properties and surface texture. Such favorable properties were considered only when choosing among top candidates. Density and strength figures were based on published information from Reference [9]. Other scores were based on the average evaluation from team members using these materials to compose the structural truss that would be the aircraft fuselage.

The layout of components was based on the analysis of moments of inertia for solid fuselage designs. The bending, shear, and torsional loads were estimated for 4g conditions. As shown in the Pugh diagram in Figure 6-2, Poplar light plywood and Balsa were the two leading candidates. In order to meet strength requirements, the more dense Poplar light plywood was used to construct the internal frame. Balsa strips were added to strengthen the frame, with placement based on Finite Element Analysis (FEA) of loads applied in bending, torsion, and vertical shear.

The process of placing members on the airframe began in the FEA program of the IDEAS CAD package. The team began with a frame of 1/8th inch thickness cut for a 1/2inch border. Material properties of American Yellow Poplar [9] were entered into the material properties section of the FEA. Estimated loads on the fuselage were the torsion of the advancing an 18 inch propeller from 0 to 6,000 RPM in 1 second, lift of the wing under a 4g load, and 3 lbs of downward force from the elevator. These loads placed on the frame showed that yield strength would be exceeded if the frame was unsupported for longer than 11 inches. Perpendicular frames were added near the wing to divide the payload into three sections, and a front and rear bulkhead were added to cap the payload bays. The FEA was run again, showing sufficient strength to prevent failure, but heavy twisting was suspected. By placing support members as needed to support the frame, the team continued this process to arrive at its final decision for constructing the prototype.

The team had originally planned to extract the payload by flipping the tail cone upwards, pushing down on the empennage to rotate the aircraft on its main gear, and the payload would roll out the rear of the aircraft. Carbon fiber rails were planned to support the balls in this configuration. Testing showed that

the team had not accounted for the friction of the balls rubbing against each other, and the balls would not roll out of the aircraft. The configuration was consequently changed to one in which the balls are extracted from the top of the vehicle.

The carbon fiber rails remained in the aircraft, and later it was noted that since professionally manufactured, provided a perfectly straight rod to which other members could be aligned. It was also noted that the rails were rigid enough to entirely support the balls, which was a simple solution to supporting the balls. The carbon fiber rods also provided a great deal of strength for little cost in weight.

It is important to note that carbon fiber materials did not score well in the Pugh diagram, but were ideal for this type of application. The reason for the low score is because "Carbon Fiber" as indicated above was the label used to identify laminated plates of carbon fiber used in the same manner as sheets of plywood. This description would imply choosing a cloth weight, determining the number and orientation of laminae, accurately forming the lay-up, and then finishing the material to fit the airframe specifications. The carbon fiber "rails" used above were prefabricated tubes, which are readily available from several suppliers. The team was careful to use such decision-making tools as clearly they are application specific. The inner frame of the vehicle was built to carry all the loads necessary to support the vehicle. This type of design simplified the structural analysis necessary to design components, as the outer "shell" was not considered. The outer structure's only purpose was to duct the air around the aircraft. Neither monocoque nor semi-monocoque, the outer structure was not load bearing, and could be built only strong enough to support its own weight. This was the fundamental idea behind all external structures.

The nosecone for the prototype was made from six layers of 3/4oz/yd fiberglass. The desired shape was formed by hand from a solid block of expanded polystyrene, and checked for symmetry with a contour gauge. The outside dimensions of the foam were sized precisely for the inside space of the nosecone. This would ensure that the nosecone shell would fit tightly over the existing structure. Each of the six layers were done individually, every 24 hours. In between each lay-up, the previous layer was sanded for smoothness before the next layer was applied. The final layer was allowed to cure under a heat lamp for two days, then painted. Once the paint had cured for another 24 hours, the foam on the inside was scrapped away. This technique was developed solely by the Georgia Tech DBF team, and has been shown to produce excellent results at a fraction of the time it would take to make a similar piece from a negative mold.

The outer skin for the main body of the fuselage had to serve two purposes. Not only did it have to duct the air around the vehicle, but it had to function as a hatch for exposing the payload. As described in the detailed design section, the hatch had to operate quickly, be lightweight, be functional and reliable, and be aerodynamically pleasing. The high fineness ratio of the aircraft led towards a long, straight body. A single sheet of flexible material was the natural solution to the design problem. Few alternatives were discussed, as this solution had many favorable characteristics with the team.

Material selection for the cover was done quickly. Polycarbonate was chosen because it cuts easily, accepts a wide range of adhesives, is cheap, and is readily available. Compared to other sheet

plastics such as ABS plastic, Nylon, Polyethylene, Polypropylene, Polystyrene, and Polyester, Polycarbonate has a higher shear modulus and tensile strength. The Polycarbonate in this application is fixed to one side of the fuselage with epoxy, and the other side is held on with Velcro.

For Brevity, the tail cone of the aircraft was made from Polyvinyl heat shrink over a balsa frame. A fiberglass tail cone was made after preliminary flight testing.

6.2 Flying Surfaces

The construction planning for the wing and empennage began with identification of the constraints and figures of merit. As discussed earlier, the span was limited more by structural limits than by aerodynamic qualities. Without any other true constraints, the accepted level of "constraint" was tentatively set to an arbitrary starting point for span and chord. These two figures were picked by using simple linear regression analysis done from previously gathered data. By using linear regression, it could be reasonably assumed that the chosen span and chord were feasible.

The team wanted flying surfaces that had smooth surfaces, did not take a tremendous amount of skill to build, did not take a tremendous amount of time to build, and were strong enough meet the strength requirements. With the exception of strength, these characteristics are easy to obtain using conventional building techniques.

The strength issue was analyzed by assuming a light core structure, such as expanded polystyrene, that would not be expected to carry the load on the wing. Then the team developed a spar using basic beam theory. From these analyses (refer to Detailed Design Phase) the material composition of the spar was decided to be carbon fiber cap strips with poplar plywood shear webbing. The difficulty for manufacturing the spar was to ensure that the final result was perfectly straight, and that the adhesion between the cap strips and the shear webbing was robust. After several trials, acceptable results were obtained using a precision-ground surface plate and laminating epoxy.

The empennage surfaces were constructed using the same foam core, balsa sheeted, polyvinyl heat shrink-covered technique as the wing. A slightly thicker balsa skin was used without a spar since normal loads are much smaller.

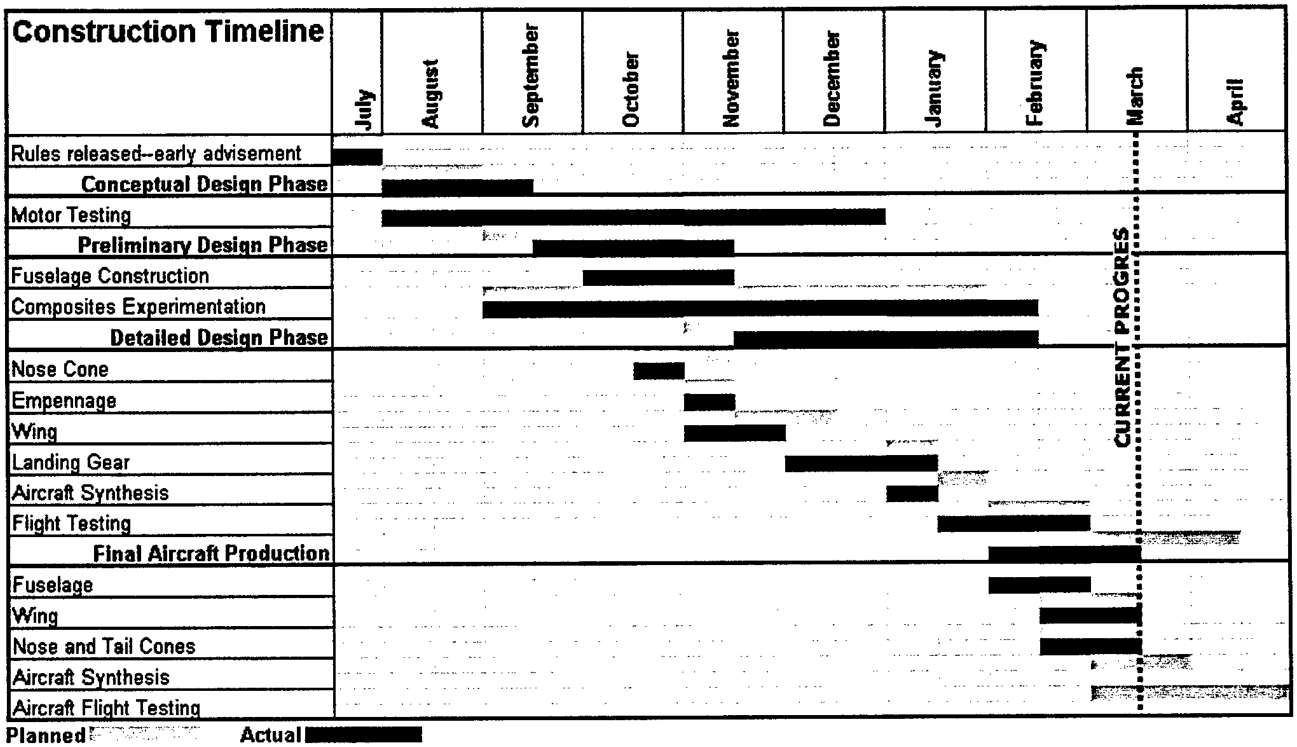


Figure 6-1: Construction Detail Timeline

Fuselage Materials

	Density	Strength	Workability	Availability	Directional Bias	Cost	Skill Required	Total Score	Rank
Index	8	9	9	5	6	3	7		
Balsa	10	10	10	10	2	10	10	422	2
Poplar Light Ply	8	8	10	10	8	10	10	424	1
Basswood Ply	6	7	7	9	8	4	7	328	7
Basswood	6	6	9	9	7	9	8	353	4
Spruce	9	9	7	2	7	7	8	345	5
Fiberglass	8	8	7	5	3	8	7	315	8
Carbon Fiber	9	10	7	5	3	7	7	338	6
Polycarbonate	8	9	8	8	10	8	8	397	3

Figure 6-2: Fuselage Materials Selection

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2002 AIAA DBF Competition Design Report

Queen's University at Kingston

"Negative Margin"

Department of Mechanical Engineering

Department of Engineering Physics

March 12, 2002

A	Parasite drag coefficient	T	Maximum airfoil thickness
A_{wetted}	Wetted area	V	Velocity
AOA	Angle of attack	V_{max}	Maximum cruise speed
AR	Aspect ratio	V_{min}	Minimum cruise speed (stall speed)
a_m	Average acceleration on ground roll	V_{mean}	Mean velocity on takeoff roll
Ac	Aerodynamic centre	V_{stall}	Stall speed
B	Induced drag coefficient	V_{TO}	Takeoff speed
C	Chord	W	Fuselage Width
C_D	Coefficient of drag	W	Weight
C_{Dpara}	Coefficient of parasite drag	X_{CG}	Position of CG
$C_{Dinduced}$	Coefficient of induced drag	X_{ACW}	Position of wing aerodynamic center
C_f	Skin friction drag coefficient	X_{ACH}	Position of stabilator aerodynamic center
CG	Center of gravity	X_{NP}	Position of stability neutral point
C_L	Coefficient of lift of wing	α	Angle of attack
C_{Lh}	Coefficient of lift of stabilator	β	Angle of bank
$C_{L\alpha}$	Derivative of C_L with respect to AOA	ρ	Air mass density
C_{Lmax}	Maximum coefficient of lift	σ	Maximum stress
C_M	Coefficient of pitching moment	ε	Downwash angle
c_p	Power coeffienct	η	Efficiency
c_t	Thrust coefficient	θ	Pitch Angle
D	Propeller diameter	μ	Dynamic viscosity
D	Drag	γ	Kinematic viscosity
d_c	Climb-out distance		
d_r	Ground roll distance		
d_{TO}	Take-off distance		
E	Wing efficiency factor		
G	Acceleration due to gravity		
FOM	Figure of merit		
H	Altitude		
I	Mass moment of inertia		
K	Form factor		
L	Lift		
M	Mass		
M	Pitching moment		
R	Turning radius		
Re	Reynold's number		
S_h	Stabilator planform area		
S_w	Wing planform area		
T	Thrust		
TOGW	Takeoff gross weight		

List of Nomenclature

1.0 Executive Summary

The Queen's 2002 entry to the AIAA/ONR Design/Build/Fly (DBF) competition marks a return to the traditional roots of model aviation. With a wingspan of 2.3m (9'), overall length of 2.1m (7') and an ambitious weight-saving design, only 3.6 kg (8 lbs.) unloaded, "Negative Margin" will prove to be a very competitive aircraft. Building on experiences of the past two years as well as team member's knowledge, we have revisited traditional modeling techniques. The design is light and efficient, allowing us to rely on a minimal but proven power system, which gives us confidence that 2002 will be our most successful year to date.

1.1 Major Development Areas

"Negative Margin" is a plane designed to "push the envelope" using a minimal power system and a light efficient frame. In an attempt to take full advantage of, and adapt to, the rule changes new construction techniques were employed, and a few very innovative design features were used. Variables such as power systems, loading/unloading mechanisms and new building methods were tested on a full-scale test section.

Early in the conceptual design phase the team studied the payload criteria, time dependency factors, and issues pertaining to the rated aircraft cost (RAC). After preliminary consultation, and the help of a simple spreadsheet the team concluded that a light airplane with minimal battery weight and a cargo capacity of 24 softballs would be the most competitive for this year's competition.

Due to the size of the cargo, a large plane would be designed leaving the team to decide on a power system. Shrouded propellers were considered as were used on last year's entry Absolute Toque but were not efficient with the chosen design. With limited resources on motors early in the year, and to keep the RAC to a minimum the team decided a single FAI-40 cobalt motor running on one 18 cell Sanyo 2400 mAh battery pack. The single motor allows for a more streamline design and affords the team the luxury of having extra motors. The Sanyo cells were chosen because of their superior energy density and efficiency. Using the single pack gives and incredible advantage in the rated aircraft cost but reduces the take-off power essential under loaded conditions. Ultimately this was tested and was deemed viable, giving the team a great benefit in overall score.

A traditional tractor configuration with tricycle landing gear was chosen. Design work began by incorporating the power system and a 24 softball capacity into a low-wing, conventional tail aircraft with an oval shaped fuselage, the minimum to encapsulate the payload. This design was incorporated into a full-scale prototype test section. Testing of this section revealed a plane, which satisfied all criteria the team set out, resulting in "Negative Margin" a light, low-wing, traditional airplane with a tailored power system.

1.2 Design Tool Overview

1.2.1 Conceptual Design

Throughout the conceptual design phase of Absolute Toque's development, a variety of tools were used to develop and evaluate different aspects of the aircraft's design. These ranged from researching old documents and studies on various components, making use of a spreadsheet detailing the *rated aircraft cost* penalty of the various designs, using the solid modeling package Solid Edge to look in 3D at various designs, to actually building a prototype test section to test the ideas under examination.

This research allowed the team to compare a variety of different designs with proven theory from several respected sources. Then actually perform our own testing to see if the technology could be applied to our design with limited resources. This allowed us to narrow down the permutations in design in preparation for the preliminary design stage.

1.2.2 Preliminary Design

Once the team entered this portion of the design process, the techniques used became more analytical than the qualitative methods used in the conceptual design. The *rated aircraft cost* spreadsheet was still used, but to screen slight variations of Negative Margin's design rather than the large structural variations examined in the previous section. Component weight was measured, estimated, or obtained from manufacturing specification sheets to allow for some first iteration values to be generated. Simple aerodynamic formulae were used to determine such things as the coefficient of lift and tail sizing, and to determine areas that needed further development.

Computer software in the form of ElectriCalc and MotoCalc were used to obtain an estimation of how much thrust the chosen electric motors would produce across their given flight envelope. These figures were used in the decision of battery pack weight and propeller selection to see if the system would produce the necessary power and to decide on the optimum propeller for the 40 Amp current limit.

1.2.3 Detail Design

The final stage of the design process meant a transition to purely analytical methods. Classical aeronautical theory was used to determine such things as drag, stall speeds, takeoff rolls, and turning radii. The use of a master spreadsheet reduced the hours of number crunching, and helped the team quickly see the effect of slight design changes.

The final values were used to produce the drawing package. "Negative Margin" was drafted into AutoCAD 14 to obtain the required construction drawings and to ensure that the components mated correctly.

2.0 Management Summary

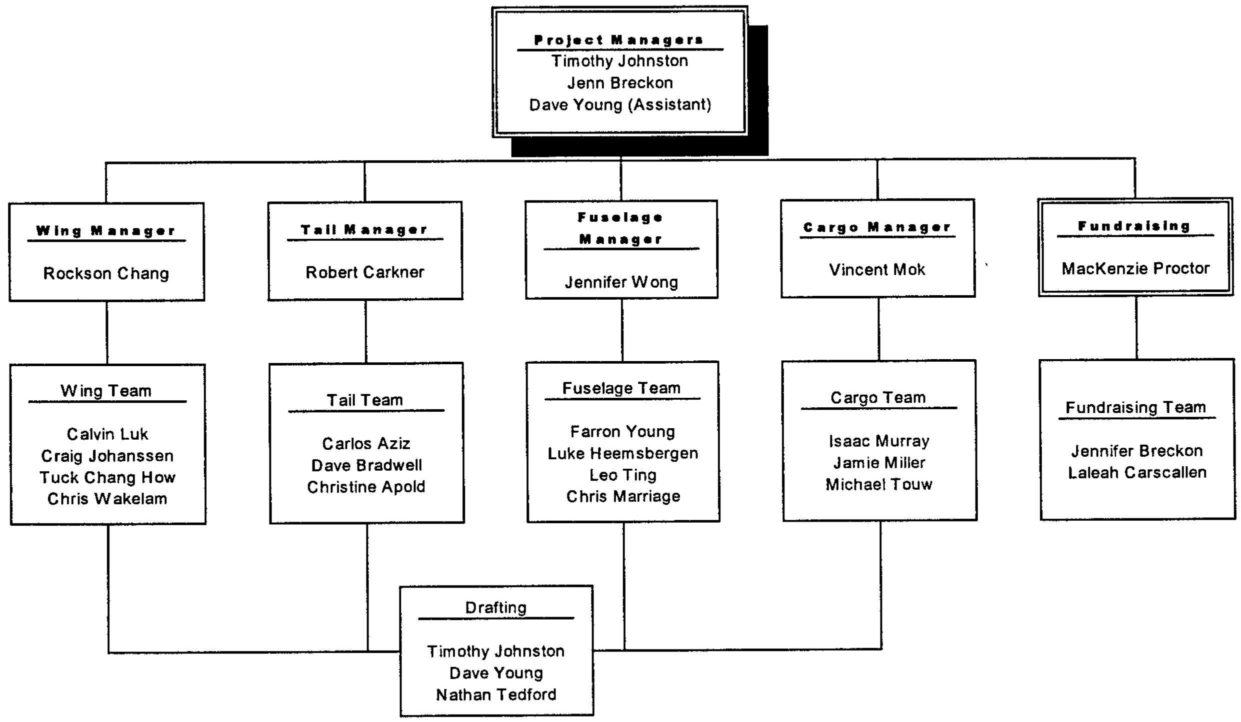


Figure 2.1 -Organizational structure

2.1 Personnel and Organizational Structure

The organizational structure chosen for this year's aero design team was based on the functional matrix plan. The design tasks during the initial portion of the year were broken down into four subgroups, with fundraising and finances being added as a fifth group. All team members were expected to be on at least two teams, to increase the knowledge each student would obtain about the various components of the aircraft.

Four of the subgroups were each tasked with the research, development, and design of one major component of the aircraft, with the fifth group being responsible for approaching outside companies to try and secure donations in the form of cash, product discounts, and access to tooling. Rough sketches and CAD drawings were submitted by each subgroup to Tim Johnston or Dave Young, who drew up the finalized working drawings of the aircraft.

In the initial part of the year, the entire team met once a week with either of the project managers, Tim Johnston or Jenn Breckon. The project managers would send an email to all members to announce the time and location. The purpose of these group meetings was to bring all of the five subgroups together to allow each team to report on their progress and to allow the entire group to offer suggestions to any

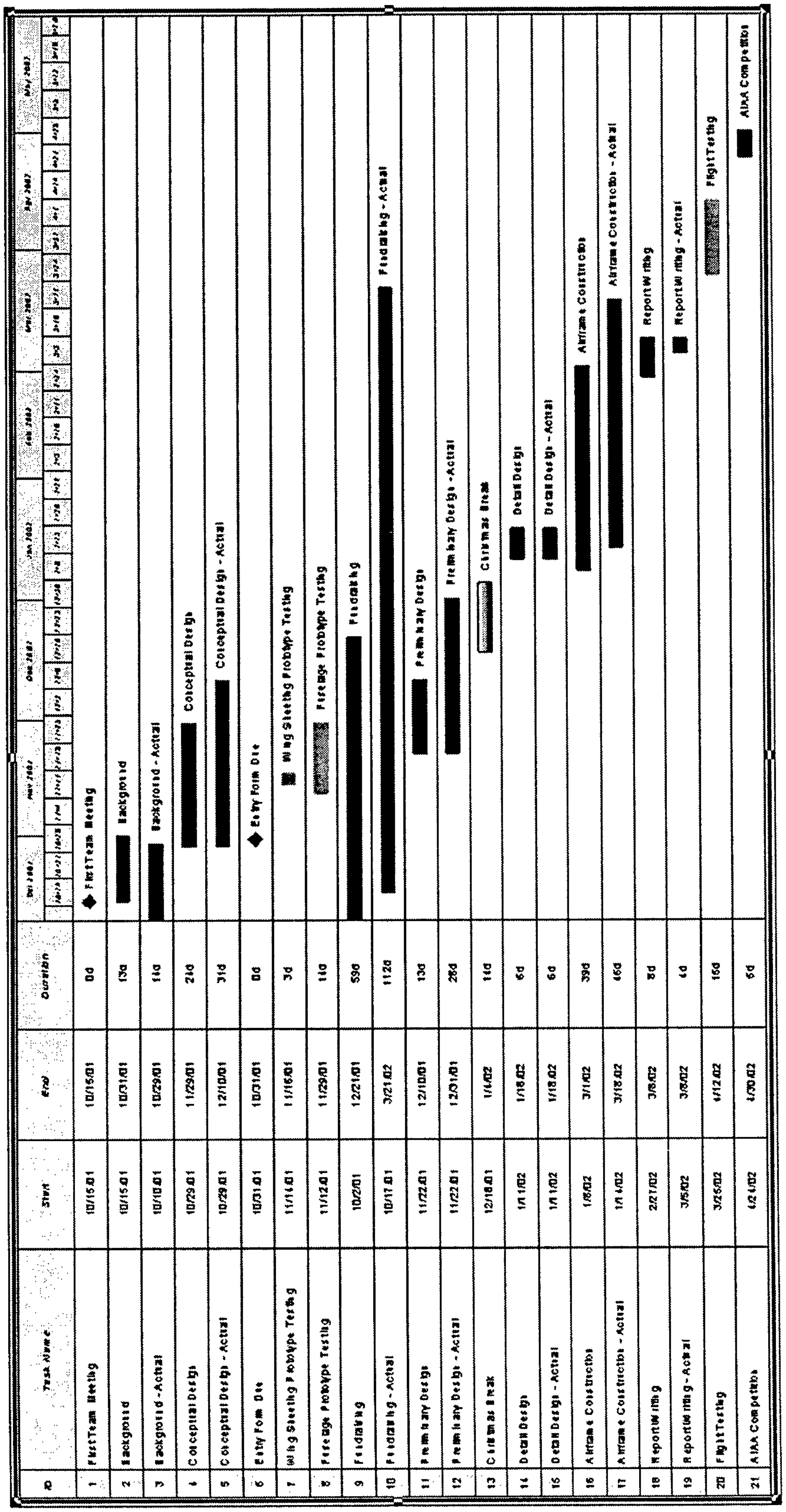
problems that arose. These meetings also helped ensure that there was adequate communication between the subgroups, and that each component would mate properly with the aircraft.

Also during this time period, separate meetings were held by the various subgroups. These meetings were called by the respective subgroup managers and were set at different times based on the schedules of the members in question. These meetings were also announced to the project managers, so that either Tim or Jenn could attend the meeting and help with the component design. It should be noted that during these meetings, the project managers were not the group leaders. It was felt that the aircraft configuration would benefit by leaving the younger students in charge of the various components, as this would ensure that everyone on the team had a chance to be an active participant in the design process. By making sure everyone was involved with the design, the actual construction of "Negative Margin" would progress much more rapidly.

After the design had been finalized, the subgroups were disbanded for the construction portion of the year. Shop hours set by the project managers, which were then emailed to everyone on the team every few days. This allowed team members to work around their own schedules and to decide when they would able to come in and build for a few hours. The shop hours were set to ensure that there would be a senior member of the team present to help the newer members learn the construction techniques required. This system ensured that every team member, regardless of prior building experience, was able to contribute to the assembly of "Negative Margin."

2.2 Scheduling

To ensure the project would be brought to completion within the time allotted, a Gantt chart timeline was produced in the team's second general meeting. Based on previous experience in the Design/Build/Fly Competition, the project managers were able to make reasonable assumptions to the length of time required to complete each stage of the project. As can be see in the Gantt chart, most estimates were very close to the actual completion time. Fundraising was a little optimistic, but sufficient funds were attained before the Christmas break to construct the test sections. Ongoing fundraising raised enough to complete the actual airplane and subsidize the trip to Kansas. There was enough slack built into the timeline to prepare for any problems brought up along the way to ensure timely completion of the project.



ID	Task Name	Start	End	Duration
1	First Team Meeting	10/15/01	10/15/01	0d
2	Background	10/15/01	10/31/01	13d
3	Background - Actual	10/10/01	10/29/01	14d
4	Conceptual Design	10/29/01	11/29/01	24d
5	Conceptual Design - Actual	10/29/01	12/10/01	31d
6	Entry Form Due	10/31/01	10/31/01	0d
7	Wing Sheeting Prototype Testing	11/14/01	11/16/01	3d
8	Fuselage Prototype Testing	11/12/01	11/29/01	14d
9	Fundraising	10/2/01	12/21/01	59d
10	Fundraising - Actual	10/17/01	3/21/02	112d
11	Preliminary Design	11/22/01	12/10/01	13d
12	Preliminary Design - Actual	11/22/01	12/31/01	28d
13	Christmas Break	12/18/01	1/4/02	14d
14	Detail Design	1/1/02	1/18/02	6d
15	Detail Design - Actual	1/1/02	1/18/02	6d
16	Airframe Construction	1/8/02	3/1/02	39d
17	Airframe Construction - Actual	1/14/02	3/18/02	46d
18	Report Writing	2/27/02	3/8/02	8d
19	Report Writing - Actual	3/5/02	3/8/02	4d
20	Flight Testing	3/25/02	4/12/02	16d
21	AIAA Competition	4/24/02	4/30/02	5d

Figure 2.2 –Gantt chart of the '01/'02 scheduling

3.0 Conceptual Design

This year's DBF competition continues the evolution of the event from one that is dominated by raw power to one that encourages efficiency and good design. The limitation on the number of scoring flights per mission, the addition of the time element, and the updated Rated Aircraft Cost formula have guided the team towards designing what we think is the most competitive aircraft for the given criteria.

Before any airframe design was started, the team examined the cargo to be transported this year. By graphing the effect of time and the amount of cargo on the overall flight score (see Figure 3.1), the team was quick to see that building a large plane with a 24 softball capacity that could fly the course as quick as possible was going to be achieve a higher score than a lightly loaded plane flying the course a minute or two faster.

With the decision made to build a large, quick plane, the team then turned to a figure of merit (FOM) to help narrow down the number of aircraft criteria. By examining wing position, tail configuration, cargo-loading configuration, and undercarriage design, the team was able to narrow the field down to three competing designs.

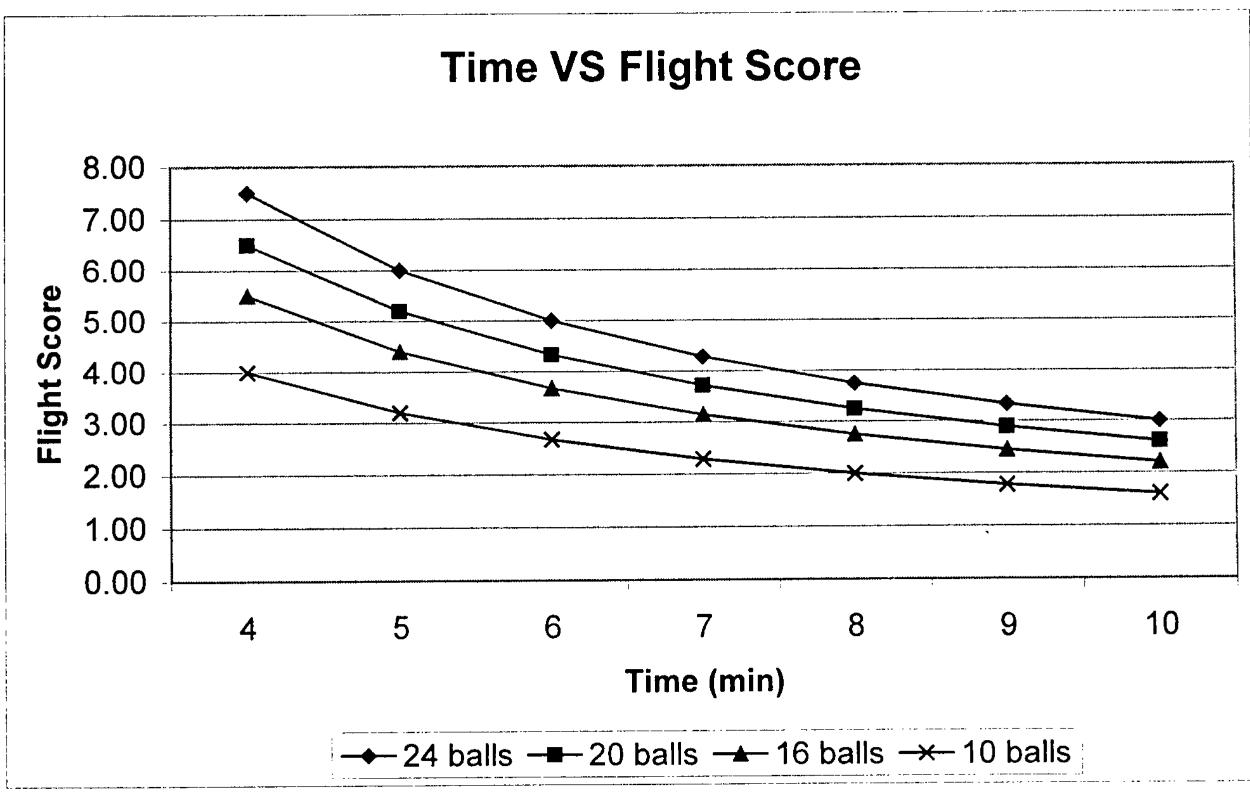


Figure 3.1 –Effect of time and amount of cargo on final flight score

3.1 Initial Figure of Merit

3.1.0 Figures of Merit

Wing Position:

The position of the wing on the aircraft was the first criterion examined. With the removal of the wingspan limit this year, the team felt that a monoplane would be a better choice over a bi-plane or tandem wing because enough wing area could be easily obtained though a single, efficient wing. This left the location of the wing to be determined through the FOM. A high-wing design offers a dual advantage of good stability and an undisturbed cargo area, yet can make loading and unloading the cargo awkward. A mid-wing provides less parasitic drag, but the cargo compartment must be designed carefully around the structural components of the wing. The third design examined, a low-wing, provides slightly less stability, necessitating dihedral to be built into the wing. However, it allows an undisturbed cargo bay with easy loading and unloading and provides a mount for a wider, more stable undercarriage.

Tail Configuration:

The tail configuration of the aircraft was also considered. The conventional tail is light, yet less efficient, resulting in a requirement for more area, which increases drag. A V-tail is light, yet has been known to create control problems in violent flight maneuvers. A T-tail is heavier, yet more efficient, producing less drag.

Cargo-Loading Configuration:

Due to the new rule stipulation that all cargo must be loaded and unloaded by hand, the cargo-loading configuration was the third criterion examined. A rear-loading tubular cargo bay employs an aerodynamic door under an elevated tail boom. It has several advantages, in that it does not interfere with the firewall mount of the motor and nose gear systems, nor is the size of the loading door is large enough to not pose any serious structural concerns. Unfortunately, working under the tail of the aircraft is awkward and slow. A front-loading tubular cargo bay, as featured on our '00-'01 entry "Absolute Toque", is much easier to load, but would be difficult to unload due to the new rule that prohibits lifting the rear of the aircraft to roll the cargo out. The top-loading design is by far the easiest to load and unload, but as it requires very large doors along the top of the fuselage, it can be difficult to build a system that provides the structural strength necessary.

Undercarriage Design:

The three types of undercarriage examined are tricycle, bicycle, and tail-dragger. The tricycle gear produces the most drag, yet offers the best ground handling. The bicycle gear is more complex as it requires out-riggers in the wings to give ground stability. The advantage of this design is that it produces a very low drag airframe for flight. A tail-dragger possesses better drag characteristics than a tricycle gear configuration, but has poor ground handling characteristics in windy conditions.

3.1.1 Figure of Merit Criteria

The following criteria were selected by the team to help identify the most promising aircraft configuration, with each design being assigned a score from 0-5, which is then multiplied by the weight of the specific criterion:

Drag: The drag of the component in question is important as it adds to the drag of the total airframe. In high wind situations, valuable battery power is lost fighting for headway, when a sleeker plane would have no trouble. This FOM was given a weight of 4.

Weight: Similar to drag, the weight of each component is added to produce the total plane weight. A wise man once said, "it's easier to take one gram from 100 parts than to take 100 grams from one part." The same principle applies here. This FOM was given a weight of 5

Performance: The design of the part is crucial to producing the best aircraft. If one option performs the same task a little better than the competition, then it receives a higher score. This FOM was given a weight of 4.

Ease of Manufacture: The best design in the world is useless if we can't build it. A higher rating indicates its relative ease of producing it. This FOM was given a weight of 4.

		Drag (4)	Weight (5)	Performance (4)	Ease of Manufacture (4)	Total
Wing Position	High	4	5	4	5	77
	Mid	5	4	3	3	64
	Low	4	5	5	5	81
Tail Configuration	Conventional	3	5	5	5	77
	V-Tail	5	5	2	2	61
	T-Tail	4	3	5	4	67
Cargo Loading	Rear	5	5	2	4	69
	Front	5	5	2	3	65
	Top	4	4	4	5	72
Undercarriage	Tricycle	3	4	5	5	72
	Bicycle	5	3	2	2	51
	Tail-Dragger	4	5	2	4	65

Table 3.1 –Initial Figure of Merit

3.2 Examined Designs

3.2.1 Flying Wing

The first possible design chosen by the team was a radical departure from anything Queen's has ever built. A flying wing with an oversized center section would house the 24 softballs in four rows of six balls. The wing would require an airfoil with a very small pitching moment, but this means that the center section must be very long to accommodate the reflexed trailing edge. To keep the wing loading to a reasonable level and the aspect ratio high to minimize drag, the outboard wing sections would have a reduced chord. The outer wing panels would also feature some dihedral to improve flight stability.

3.2.2 Blended Fuselage

The second design selected is very similar to the flying wing but it includes a conventional tail for more predictable flight characteristics. The wing would not require the reflexed trailing edge profile that produced such a large chord on the flying wing; instead a conventional flat-bottomed wing could be used. The tail would be mounted on a light boom structure to minimize weight and drag.

3.2.3 Conventional

The final design considered was similar to several designs Queen's has built and flown over the years. It features a box section fuselage with rounded corners to reduce drag. The wing fitted has a high aspect ratio and a constant chord, and the tail is located high up on the back of the fuselage. Cargo would be carried in two rows of 12 to minimize frontal area and to provide a fuselage long enough for an adequate tail moment. This design above all is predictable and easy to build.

3.3 Screening the Three Designs

Once the team had selected three designs to study further, another FOM was set up that studied how the various airframes compared with respect to drag, weight, performance, ease of manufacture, and their respective *Rated Aircraft Cost (RAC –see figure 3.4).* The weighting of the respective criteria remained the same as it was for the component selection process in 3.1. The FOM Rated Aircraft Cost was calculated by dividing 5 by the RAC, then multiplying by 10. This is a heavy rating as it was felt that this criterion had a huge effect on the team's final score and standing at the competition. See Table 3.2

Aircraft Concept	Drag (4)	Weight (5)	Performance (4)	Ease of Manufacture (4)	Rated Aircraft Cost (5)	Total
Flying Wing	5	3	4	4	4.02	87.1
Blended Wing	5	4	4	4	3.60	90.0
Conventional	3	5	4	5	3.77	87.9

Table 3.2 –Screening the three designs

3.4 Analysis of the Rated Aircraft Cost

Once the decision had been made to build the blended wing design as indicated by the FOM (see table 3.2), the team analyzed the RAC of the design by components (see figure 3.2). As the RAC has a direct bearing on the team's final score, reducing this penalty would improve the team's standings. When figure 3.2 is examined, it is apparent that the portion of the design that suffered the highest penalty is the rated engine power, which is based on battery weight. Reducing this weight would also reduce the overall weight of the aircraft, further lowering the RAC.

As the mission profile has changed for this year's competition, for the first time it is possible to design a competitive plane with less than the maximum weight of batteries. To make this possible, the plane's weight must be reduced dramatically. The team decided to re-analyze the three aircraft designs, but in a new FOM with the weight criterion given a multiplier of 10 (see table 3.3).

Aircraft Concept	Drag (4)	Weight (10)	Performance (4)	Ease of Manufacture (4)	Rated Aircraft Cost (5)	Total
Flying Wing	5	3	4	4	4.02	102.1
Blended Wing	5	4	4	4	3.60	110.0
Conventional	3	5	4	5	3.77	116.9

Table 3.3 –Screening the three designs for weight

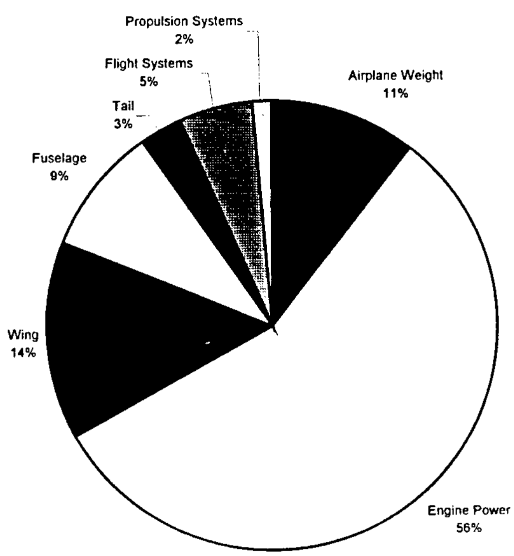


Figure 3.2 –Rated Aircraft Cost breakdown per aircraft component

3.5 Shrouds

Research performed at Queen's in previous years led to the utilization of shrouded propellers on the '00-'01 entry, "Absolute Toque". These units are ideal for increasing thrust at low speeds due to the low pressure created over the lip of the shroud from air accelerating into the propeller. Further efficiency is gained by the elimination of vortices from the tips of the propellers. Due to these advantages, the possibility of shrouding the propeller on this year's aircraft will be further analyzed in the preliminary design section.

3.6 Design Summary

The final aircraft "Negative Margin" (see figure 3.3 –Conventional) has the following specifications:

- 24 softball capacity, held in two rows of 12, centered over the plane's center of gravity
- Two top hatches, opening in a gull-wing configuration, with 4 linked mechanical latches per door
- Low-wing configuration, utilizing a flat-bottom type airfoil and a constant chord
- Tricycle undercarriage with a wide stance for excellent ground handling
- A single motor with a minimal battery pack
- A RAC of 9.30 based on preliminary weight estimates

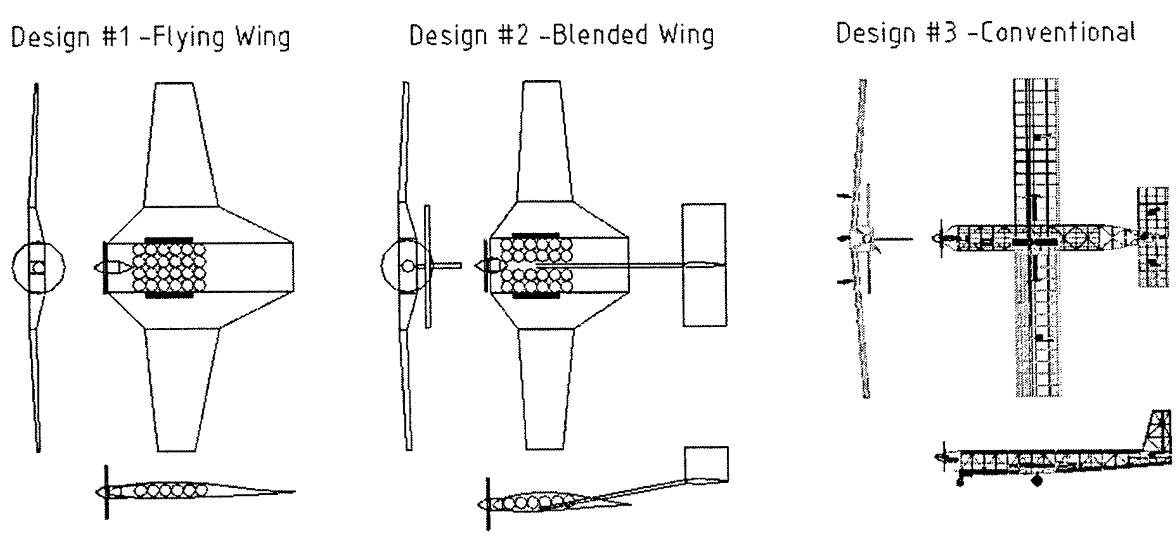


Figure 3.3 –Various Designs Examined

AIAA Design/Build/Fly 2001/2002					
	Rated Aircraft Cost Spreadsheet	Flying Wing	Blended Wing	Conventional 5lb battery	Negative Margin 2.4lb battery
Coeff.	**Description**				
A	Manuf. Empty Weight Multiplier ($/lb)	100	100	100	100
MEW	airplane weight (incl. batteries) (lbs.)	18	15	14	11.5
	Sum A*MEW ($)	**1800**	**1500**	**1400**	**1150**
B	Rated Engine Power Multiplier ($/Watt)	1500	1500	1500	1500
	number of engines	1	1	1	1
	total battery weight (lbs.)	5	5	5	2.4
REP	Rated Engine Power (Watt)	5	5	5	2.4
	Sum B*REP ($)	**7500**	**7500**	**7500**	**3600**
C	Manufacturing Cost Multiplier ($/hour)	20	20	20	20
	number of wings	1	1	1	1
	wingspan (ft.)	10	10	10	8.67
	maximum chord (ft.)	5	3.75	1.33	1.33
	number of control surfaces	2	2	2	2
	WBS 1.0 Wing:	**126**	**116**	**96.64**	**86**
	length (ft.)	0	6.25	6.67	6.67
	WBS 2.0 Fuselage:	**0**	**62.5**	**66.7**	**66.7**
	number of vertical surfaces	0	0	0	0
	number of vertical surfaces with control	0	1	1	1
	number of horizontal surfaces	0	1	2	2
	WBS 3.0 Empennage:	**0**	**20**	**30**	**30**
	number of servos or controllers	4	7	7	7
	WBS 4.0 Flight Systems:	**20**	**35**	**35**	**35**
	number of propellers or fans	1	1	1	1
	WBS 5.0 Propulsion Systems:	**10**	**10**	**10**	**10**
MFHR	Manufacturing Man Hours	156	243.5	238.34	227.7
	Sum C*MFHR ($)	**3120.00**	**4870.00**	**4766.80**	**4554.00**
	Total ($1,000s)	**12.42**	**13.87**	**13.67**	**9.30**

Table 3.4 –Rated Aircraft Cost of the four designs considered

4.0 Preliminary Design

4.1 Take-off Gross Weight (TOGW) Estimation

The first step in the preliminary design phase was estimating the gross weight of the aircraft, as this parameter is crucial in determining the size and performance. To determine the TOGW, it was necessary to compile the individual weights of all known components and estimate the weight of various airframe structures.

A cargo capacity of 24 softballs yields a cargo weight of 4.1 Kg (9 lbs.). With the payload specifications changed, having only one type of cargo for this year's airplane, the performance design was based on a full cargo complement of 24 softballs. A large airframe is required to accommodate the volume the softballs occupy and research was done investigating previous competition aircraft weights. With careful material selection, new construction techniques and engineered weight management an empty airframe weight of 4.8 kg (10.6 lbs.) was expected. Adding a 1.1 kg (2.4 lbs.) battery pack and payload, a **TOGW of 9.8 kg (22 lbs.)** seemed ambitious but reasonable.

4.2 Propulsion Systems Selection

4.2.1 Motor Selection

Regulations stating that brushed motors must be selected from the AstroFlight or Graupner lines remained in effect for the 2001/2002 season. After using AstroFlight Cobalt 40 motors last year, with excellent performance and reliability, the decision was made to continue employing this brand of motor. These motors have a solid reputation; capable of handling the voltages and high currents present in this competition.

A rule of thumb used in the electric flight community is that a minimum of 120 Watts/kg be used in rise of ground (ROG) flight. This approximation is marginal for smaller sport airplanes but with the large aircraft considered here this performance is adequate. Previous competition experience has verified this suggestion and provides an excellent staring point for motor selection. Using this information and knowing the approximate TOGW Negative Margin requires approximately 1200 Watts.

MotoCalc and ElectriCalc, commercial software packages, were used to compare various combinations of motors, gearbox, propellers and batteries. The software would produce information regarding each configuration's efficiency, thrust, propeller pitch speed, duration and current draw. In the conceptual design phase it was chosen that a single motor would be used. This decision limited the choice of motors to the AstroFlight FAI 40, FAI 60, or FAI 90. The FAI 90 was eliminated from the choices immediately because of the high weight and cost. One of the parameters the team decided on

early was the necessity for a lightweight aircraft. The FAI 40 and FAI 60 were analyzed. The FAI 60 was found to be the ideal motor in overall terms but had the drawback of requiring 36 cells for operation. In the quest for weight saving and the great advantage in RAC by halving battery weight the FAI 40 was chosen as the motor for Negative Margin. Though the 40 is slightly underpowered, producing approximately 900 Watts and 97 oz of thrust, all other performance criteria were acceptable with this motor. It is ambitious to push the power envelope; it is considered an acceptable risk considering the advantage in final scoring.

4.2.2 Battery Selection

Voltage and current define the power output of the motors. With the maximum current set at 40 Amps, the only way to increase motor output power is to increase the number of cells used. The AstroFlight FAI 40's are rated to 18 cells but have been known to run on 20 or 22 cells. The next requirement was to find the batteries with the highest power density available. No new NiCd batteries have arrived on the market since last year's competition so the selection was the **Sanyo RC2400** cells. As was the case last year these cells were chosen for their high power density after considering all factors including: weight, capacity and internal resistance. The cells are arranged in an **18-cell pack** and a **4-cell pack** if additional power is needed.

4.2.3 Propeller and Gearing Selections

To maximize the efficiency of the power system, one must carefully balance the thrust produced and the propeller pitch speed. Adjustments to this balance may be made by gearing the motor to throw propellers of various sizes. Based on previous experience and preliminary calculations an initial pitch speed of **31.3 m/s (70 mph)** was chosen. Using this information an initial propeller diameter was selected considering various factors such as gear ratios and pitch. It was found that operating an AstroFight helical gearbox with a 3.1:1 ratio a **13" by 8"** would produce a pitch speed of **31.3 m/s (70 mph)** and static thrust of **32.3 N (7.25 lbs)**. This results in a thrust to weight ratio of approximately 1:3 and maximum aircraft velocity of **28.6 m/s (64 mph)**. These estimations have been made assuming a drag coefficient of 0.03 the worst-case scenario based on the calculations of the airplane entered two years ago "Obsidian", which had a similar shape.

4.2.4 Shroud Sizing

A very successful experiment last year was the addition of shrouded propellers. The performance enhancements were significant enough to warrant a feasibility study into the use of shrouded propellers on this year's entry. The design of the shrouds was based entirely on an undergraduate thesis completed by 4 students, 3 of them members of the Queen's Aero Design Team. The thesis involved the design and

development of a variable geometry shroud for small scale UAV's. From this thesis, it was determined that fixed shrouds would provide a significant efficiency increase by reducing tip loses and straightening the efflux flow. Properly sized shrouds would also create their own thrust due to the low pressure created over the leading edge radius of the inlet lip, provided that the inlet area is greater than the exit area. This is achieved by using a cambered airfoil cross section. The inner diameter of the shrouds is of course dictated by the diameter of the propeller, and the outer diameter by the camber and thickness of the ring airfoil. The chord of the shroud is based on maintaining a smooth airfoil shape, about 0.3 m (12 in). Unfortunately, the constructions techniques employed, as well as the additional structure needed to mount the shroud would increase the weight approximately 15% over traditional motors. The added efficiency of the shroud is not high enough to offset the weight gained in the structure. Hence, the shrouds were dropped from the design at this point.

4.3 Wing Area and Airfoil Selection

The main characteristics driving wing area and airfoil type were stall characteristics, drag estimation and lift requirements. An FOM was used in selecting the desired wing attributes. All criteria are explained below, on a rating scale of zero to five, with five being the most desirable.

C_l at Best Lift to Drag (L/D) Angle of Attack (AoA)

The coefficient of lift at the best L/D was used to examine the amount of lift the airfoil would produce operating at peak efficiency. Inspecting the airfoil at this point is important because the greater the amount of lift generated the smaller the area can be, cutting down on weight and drag.

Maximum C_l

Maximum C_l was scrutinized because it is the key factor in stall speeds, take-off speeds and the maximum g-loading for a fixed area wing. Due to the limit on take-off distance and the energy advantages obtained by minimizing the amount of time in climb, a moderately high C_l was considered advantageous. Airfoils containing excessively high C_l's can create undesired induced drag.

Stall Characteristics

This FOM arises from the past experiences of too many modelers. An airfoil possessing a gentler stall increases the time available to react and increases the likelihood of recovery. The stalling characteristic scores were compared using published lift, drag and pitching moment data.

Induced C_d at Expected Cruise AoA

Due to the restrictions on battery power, drag reduction is of prime importance. All airfoils were compared at expected cruise C_l, where the drag will have the most influence on performance.

Ease of Manufacturing

The complexity of an airfoil shape has a direct effect on its ease of manufacturing and inherent structural integrity. Many intricate high lift airfoils possess thin trailing edges and concave undersides which are harder to build and require additional structure to maintain structural strength.

	C_l at best L/D AoA	Maximum C_l	Stall Characteristics	Induced C_d at Cruise	Manufacturing Ease	**Total**
Clark Y	4	4	4	4	5	**21**
Eppler 197	5	4	5	5	5	**24**
Naca 64_1-612	5	5	4	4	4	**22**

Table 4.1: Airfoil Figure of Merit

4.3.2 Airfoil Section and Wing Area

When sizing for wing area; wing loading, surface drag and induced drag must be considered, along with the parameters mentioned above. A larger wing area has higher parasitic drag, but lower induced drag. This arises from the fact that a large area does not require an airfoil with as high coefficient of lift to produce desired lift characteristics. The drawback of a large wing area is the added structure resulting in a slight increase in weight. It was decided that a desired wing loading under full payload conditions should be in the range of **97.3 g/dec² (35 oz/ft^2)**. Though the preliminary estimate of TOGW was approximately 22 lbs., past experience has shown that actual weights tend to be higher than estimates so the wing area estimations were based on a 10.3 kg (25 lbs.) airplane. Using this information, an area of about 1.07 m² (1650 in²) would be required. A flat bottom airfoil would be used to provide the necessary C_L. The flat bottom airfoil, being lighter to construct, would offset the weight of having the larger wing area and provide a lower induced drag. From this, the C_L at cruise was determined from the standard lift equation. Three-dimensional effects reducing the overall lift of the wing will be more thoroughly examined in the detailed design when the final wing configuration has been selected.

$$C_{L\min} = \frac{2L}{\eta \rho S_w V_{\max}{}^2}$$

Where η is the efficiency of the wing, assumed to be 0.9, and L is the total lift required, equal to the TOGW of 10.3 kg (25 lbs.).

This gives a required C_L of **0.28±0.1** at max a velocity **27 m/s (60 mph)** and a C_L of **0.32±0.1** at a of cruise speed of **24.0 m/s** (**54 mph**). The stall speed would be determined by the maximum coefficient of lift obtainable from the airfoil. Most flat-bottomed airfoils have a maximum C_l at approximately 1.3±0.1 with a well designed wing being able to achieve 90% of this maximum C_l. With a stall speed of approximately **11.1 m/s** (**25 mph**) can be achieved with a wing C_L of **1.22±0.1**. It was also desired to have an aircraft capable of maneuvering with a G loading of at least 2.5 at cruise speed, which gives a required maximum C_L of 1.05±0.1, well within that of a normal flat bottomed airfoil.

Maximum takeoff distance is one of the primary design considerations in determining the airfoils maximum C_l. If the C_l of the airfoil were not high enough to allow takeoff within the set distance, high lift devices would need to be incorporated. With the preliminary weight, thrust, and wing area known, with a set takeoff distance of 61 m (200 ft), a minimum C_l for take off was determined to be 0.52±0.1, achievable by many airfoils.

With all of the above preliminary calculations completed, the wing was required to have a **C_{Lmin} of 0.26 and C_{Lmax} of 1.22**. Airfoil and drag data were obtained from the UIUC Low-Speed Airfoil Test program and several airfoil texts. Their performance was examined with regard to the parameters and the desired values calculated above. The final selection made was the **Eppler 197**, with a C_L of 0.28 and a C_D of 0.0068 at a minimum drag angle of 0°, and a maximum C_L of 1.23 at the critical angle of attack of approximately 11°. The Eppler 197 airfoil has slightly less drag than the Clark Y used previously, but still possess similar high lift and gentle stall characteristics.

4.4 Aspect Ratio (AR)

A higher aspect ratio reduces the induced drag of the aircraft (ie. Gliders), thus allowing for a faster cruise velocity for equal power output. The wing span limit imposed in last years rules, has been lifted for the 2002 season leaving the span open to debate. With no limit on span it was decided that a wing no larger than 2.75 m (9 ft) would be used. Aside from the RAC advantages of having a smaller span, team felt it would allow sufficient speed and mobility for the competition. With the wing area previously estimated at 1.07 m^2 (1650 in^2), an average chord of 0.41 m (16 in) would be required for a constant chord wing as discussed in the conceptual design section. The final wing was 2.65 m (8 ft 8 in) with a 0.41 m (16 in) chord resulting in a wing area of 1.08 m^2 (1672 in^2). The subsequent aspect ratio was **6.5**, which is respectable for high lift aircraft with moderate speed and maneuverability.

4.5 Wing Platform

There are many choices of wing platform including rectangular, taper, double taper and elliptical, just to name a few. The main consideration of the wing platform this year was the rated aircraft cost. In the RAC calculation the longer the span, the more the penalty assessed, in efforts to keep the RAC as low as possible a shorter wing was desired. The "ideal" wing planform is an elliptical shape. It has the lowest induced AoA, induced drag and stalls evenly across the span (ie reduced tip stalling). The major drawback of an elliptical wing is the structural complexity of it. They are very hard to manufacture and repair if need be. A rectangular wing with an AR 6 requires an induced AoA approximately 17% higher and has about 5% greater induced drag. The major advantages of the constant chord wing are that they are easy to build, maintain constant Re across the span, reducing tip stalls and requires the shortest span to produce required area. The latter of concerns is the most important as to keep the RAC low. A double

taper wing was also considered; but because of the span required and the danger of tip stalling at low Rn's it was decided against. This year the main concern was keeping the RAC as low as possible and took precedence over the efficiency of the wing.

4.6 Horizontal Stabilizer Sizing

The design considerations used to determine the required tail surface dimensions are stability and control authority. The wing is capable of approximately 2.5g before stalling (see section 4.3.2), and the Eppler 197 has a moment coefficient of approximately -0.088 for a cruise angle approximated at 2°. The stabilizer must be capable of overcoming both the pitching moment of the wing and the moment caused by a finite separation between the center of gravity and the center of pressure on the wing (assumed for now to be within 0.0254 m or 1 in. of each other). The tail must then still provide enough torque for control. This leads to the inequality:

$$X_{ach}\tfrac{1}{2}C_{Lh}\rho S_h V^2 \geq \tfrac{1}{2}C_M \rho S_w V^2 c + 2.4X_{acw}W + I\ddot{\theta}$$

Using this expression, the product of stabilizer maximum coefficient of lift, surface area, and distance from the center of gravity ($X_{ach}C_{Lh}S_h$) can be found. In order to minimize its size and reduce drag, the tail is placed as far aft as feasible to give it a large moment arm on which to act. On Negative Margin this placement is achieved through the length of the payload section alone, though being slightly shorter coupled than and ideal design the distance is acceptable. It is common practice to use a stabilizer that is approximately 20 to 22% percent of the wing area. Further, the accepted and recommended elevator size is 40% of the stabilizer area. With the slow speeds and multiple takeoffs and landing in this competition, pitch authority was deemed very important. A new rule implemented this year restricts the span of horizontal surfaces to 25% of the wingspan or it will be deemed a "wing". A horizontal stabilizer conforming to these rules was designed and considered, but following a stability analysis was deemed unsafe and may not perform. The team decided that two horizontal surfaces be built and placed on either side of the fuselage. This keeps the span below 25% wingspan and each surface has a controllable "elevator". For the purposes of design and this report the two surfaces will be described as the horizontal stabilizer. It has been found that a **0.24 m^2 (370 in^2)** stabilizer located **1.02 m (40 in.)** from the CG would require a C_l of approximately 0.2±0.1. This would be easily provided by a **NACA0009 airfoil** (C_{Lmax} of 1.3) (ref. 7), which has the desired horizontal stabilizer qualities of low drag and zero pitching moment. These will be further quantified in detailed design.

The horizontal stabilizer platform is again designed in a rectangular fashion to save span distance. An aspect ratio close to 5 was also desired in order to reduce induced drag. This resulted in a constant chord of 0.27 m (10 in) and a 0.94 m (37 in) span revealing a 3.7 aspect ratio. This is slightly less than is recommended by Lennon (Ref 10), who recommends an AR of 4 or 5 with a constant chord, but is satisfactory in reducing drag.

As decided upon in conceptual design, a conventional tail configuration would be used. Construction of this design is light than a T-tail but less efficient. Having a low wing aircraft, it was felt that the tail would be sufficiently out of the downwash of the wing.

4.7 Vertical Stabilizer Sizing

With the use of a conventional tail no extra structure had to be added to the vertical stabilizer to support the loading of a T-tail. This allowed the design team to produce a very lightweight vertical stabilizer, which must provide yaw control and roll stability for the aircraft. It was decided that the rudder servo would be mounted in the horizontal stabilizer to save weight in the vertical tail. The tail is simple stick construction, using ¼" balsa stick, a common practice in the RC modeling field. For an airplane to be stable in yaw, the Center of Lateral Area should be about 25% back from the center of gravity. In the preliminary AutoCAD design, quick area moment calculations were done to show that the area selected was adequate for yaw control. A vertical fin area of approximately 40% of the horizontal tail area was decided upon therefore the vertical tail area **0.09 m^2 (144 in^2)** would be employed, resulting in a root chord of 0.25 m (10 in). The vertical fin was swept for aesthetics. Further analysis of the vertical fin height and width is completed in the detail design section.

4.8 Fuselage Design and Sizing

4.8.1 Fuselage Design Considerations

During the preliminary design stage, several design parameter and sizing trades were considered. While innovative design and construction methods were investigated, ease of manufacture, functionality and cost were considered. The decision of airframe design depended on compromises between simplicity of construction, strength, weight and reduction of drag. The following aspects of performance were considered in the proposals of fuselage geometry.

Efficiency

The layout of the fuselage should optimize the space required for the airframe structure and payload while limiting the fuselage's overall size. A restriction on the configuration of the softballs (ie. must be side-by-side, no staggering) was the main factor contributing to the design of the most efficient airframe.

Manufacturing Ease

The preliminary design of the fuselage should limit the cost and time required for its construction. The goal this year was to produce the easiest and cheapest airframe to manufacture in light of previous year's experiences. Another aspect to consider at this point is the shop tooling and facilities available to the team.

Functionality

The fuselage must act as the cargo-carrying portion of the aircraft. The design feature used for access to the cargo must be quick, rugged and reliable due to the rushed nature of payload handling. It must include provisions to support the softballs under varying flight load conditions.

Structural Rigidity

The airframe structure must be strong enough to account for the cargo loadings on the repeated take-offs and landings the aircraft will experience. It must also be rigid enough to prevent flexing, resulting in reduced effectiveness of the vertical and horizontal stabilizers. The types of materials and structures used for the primary components would depend on the ability to withstand the expected loading.

Drag Penalty

The design of the airframe should minimize the amount of parasitic and interference drag created by the fuselage. This will reduce the power required for cruise and increase overall top speed, increasing motor run time. All junctions between airframe components should be smoothly faired in order to reduce interference drag. Any fuselage upsweep should be less than 15° to prevent separation drag.

Weight

The weight of the fuselage is of prime importance because of the reduced power plant. The materials and structures must all be designed with weight considerations.

4.8.2 Fuselage Sizing

The design of the fuselage was based on the parameters set put above. The length of the fuse was dictated by the size of the cargo bay and the distance required for the tail moment arm. The rules state that softballs must be no more than one high and a minimum of two balls across. Working within these rules the cargo is configured in two bays, sized for the diameter of one softball. The payload bay is 48" in length to accommodate 12 balls on each side of the fuselage, thereby allowing a total capacity of 24 balls. This hold is centered over the wing 1/3 chord (predicted center of gravity), with the addition of a faired aft end there was sufficient length for an appropriate tail moment arm. The fuselage has a rectangular cross-section with rounded corners of radius 1 in to reduce drag. The motor is mounted on a rigid firewall at the nose of the aircraft. The final dimensions came out to be an overall length of **2.03 m (80 in)** with a frontal cross-sectional area of **0.22 m (8 5/8 in).**

4.8.3 Cargo Access

Access to the cargo hold is facilitated through two doors mounted on the top of the fuselage and hinged in the middle. The doors are constrained by a tooth and latch system designed by the team. All latches are attached to a single handle allowing quick release of all teeth, which permits the doors to be

opened simultaneously for loading/unloading. When loading or unloading is complete the handle is pulled once more, doors are closed and secured into place.

4.9 Undercarriage Design

The tricycle landing gear design used for Negative Margin must be able to absorb the landing shock and provide excellent ground handling capabilities. The main wheel stance is slightly wider than ¼ of the wingspan, chosen to prevent upset in high crosswinds. The nose gear is located as far forward on the fuse as possible. This provides a very stable base for the aircraft. The propeller is mounted at the top of the firewall, which measures approximately 8 inches. Because the maximum size propeller that will be used is 14 inches there is no trouble with prop to ground clearance. Therefore the length of the landing gear was only based on providing adequate clearance for rotation and the wing tips. A gear length of 0.11 m (4.5 in) allowed slightly over 15 degrees of rotation and 0.2 m (8 in) of wing tip to ground clearance. The main landing gear was positioned 0.05 m (2 in) behind the center of gravity so that when the aircraft rotates on takeoff, the wheels remain behind the center of gravity.

Dual ball bearing supported aluminum wheels, 0.089 m (3.5 in) in diameter, are used for the main gear. Because the aluminum does not deform, the rolling resistance is very low. However, these wheels are prone to sliding sideways in crosswinds, necessitating a rubber coating to increase friction. A plastic rubber has been found which coats the running surface of the wheel and provides sufficient friction to prevent sideways motion but not hindering the low rolling resistance. The nose gear supports less weight, so a standard 0.063 m (2.5 in) model aircraft wheel was used. The higher friction of this wheel provides better directional control on takeoff.

Gear struts are formed from bend ¼ in diameter music wire. The main gear uses a torsion bar design for shock absorption, where as the nose gear uses a single coil.

Item	Imperial	Metric
Propulsion		
Motor	AstroFlight	
	FAI 40 G	
Speed Controller	Astro 204D	
Cells	18, 2400mAh	
Gearing	3.1:1	
Propeller	13 by 8	
Static Prop Thrust	6.1 lbf	26.7 N
Run Time (100%)	3.3 min	3.3min
Motor Efficiency	80%	80%
Wing		
Airfoil	Eppler 197	
Wing Area	1672 in^2	1.079m^2
Span	104.5 in	2.65 m
Chord	16.0 in	0.406 m
MAC	16.0 in	0.406 m
Wing Thickness	2.17 in	0.055 m
Aspect Ratio	6.5	6.5
Taper	None	
Aileron Area	134.1 in^2	0.086 m^2
Aileron Length	33.5 in	0.851 m
Aileron MAC	4.00 in	0.102 m
Fuselage		
Length	80.0 in	2.03 m
Max Height	7.75 in	0.197 m
Width	8.63 in	0.219 m

Item	Imperial	Metric
Horizontal Stabilizer		
Airfoil	NACA 0009	
Area	373 in^2	0.241 m^2
Span	37.4 in	0.949 m
Chord	10 in	0.254 m
MAC	10 in	0.254 m
Thickness	1.0 in	0.025 m
Aspect Ratio	3.75	3.75
Taper	None	
Elevator Area	91.9 in^2	0.059 m^2
Elevator Span	30 in	0.762 m
Elevator MAC	3.00 in	0.076 m
Tail Moment Arm	40.0 in	1.02 m
Vertical Stabilizer		
Airfoil	¼" Stick	
Area	114 in^2	0.074 m^2
Root Chord	10.0 in	0.240 m
Tip Chord	6.0 in	0.152 m
MAC	8.167 in	0.207 m
Height	14.0 in	0.356 m
Rudder Height	13 in	0.330 m
Rudder MAC	3 in	0.076 m
Rudder Area	39 in^2	0.025 m^2
Velocities		
V_{max}	64 mph	28.6 m/s
V_{cruise}	55 mph	24.4 m/s
V_{stall}	22 mph	9.78 m/s

Table 4.2. Summary of aircraft geometry based on both preliminary and detail design calculations

5.0 Detail Design

Drawings of the final aircraft design are presented at the end of this section. These drawings include detailed two-dimensional drawings of the airplane along with a three-dimensional isometric close-up view of the cargo bay to illustrate how the softballs are constrained.

The design of any airplane is a highly iterative process, involving many changes to the initial design before arriving at the final configuration. To accommodate this process, a master spreadsheet was developed that performed the necessary calculations while only requiring some airfoil properties and several key dimensions of the aircraft geometry. This spreadsheet allowed the team to make large or small design changes without having to re-calculate all parameters of the design.

5.1 Weights

The actual weights of all electrical equipment, control equipment and hardware was calculated and tabulated. The airframe is constructed entirely of balsa and plywood. This allowed a very accurate weight estimate by knowing exactly the amount of wood needed for the AutoCAD drawings and the wood properties. These were also tabulated. Care was taken not to underestimate the weight of the aircraft, as all primary design features require an accurate approximation of weight in the calculations. A summary of this data is presented below.

DESCRIPTION	WEIGHT (lbs)	WEIGHT (N)	NUMBER	SUBTOTAL (lbs)	WEIGHT (N)
1-12" .47 Softball	**0.375**	**1.669**	**24**	**9.000**	**40.047**
Battery Pack	**2.400**	**10.679**	**1**	**2.400**	**10.679**
Receiver	**0.125**	**0.566**	**1**	**0.125**	**0.556**
Servos	**0.125**	**0.556**	**6**	**0.750**	**3.340**
Landing Gear	**0.750**	**3.337**	**1**	**0.750**	**3.337**
Wheels	**0.125**	**0.556**	**3**	**0.375**	**1.670**
Wing	**3.500**	**15.574**	**1**	**3.500**	**15.574**
Tail	**1.000**	**4.450**	**1**	**1.000**	**4.450**
Fuselage	**2.500**	**11.124**	**1**	**2.500**	**11.124**
Motor	**0.688**	**3.059**	**1**	**0.688**	**3.059**
Speed Controller	**0.075**	**0.334**	**1**	**0.075**	**0.334**
Gear Box	**0.094**	**0.417**	**1**	**0.094**	**0.417**
Shroud	**0.000**	**0.000**	**0**	**0.000**	**0.000**
Receiver Battery	**0.250**	**1.112**	**1**	**0.250**	**1.112**
Hardware	**0.250**	**1.11**	**1**	**0.250**	**1.112**
Prop	**0.200**	**0.890**	**1**	**0.200**	**0.890**
Take Off Gross Weight with 24 softballs				**21.956**	**97.699**
Empty Flying Weight				**12.956**	**57.652**
Airframe Weight				**10.556**	**46.972**

Figure 5.1 –Weight breakdown by aircraft component

5.2 Payload Fraction

Payload fraction is a measure of the payload's contribution to the take-off gross weight of the aircraft. The payload fraction of Negative Margin is **0.410.** Much of the airframe weight is being put towards creating a voluminous cargo bay and reinforcement to maintain the airplanes' integrity. This leads to a slightly below average payload fraction.

5.3 Drag

In order to make accurate predictions of flight speed and accelerations, the drag of the aircraft must be calculated. In this calculation, the total drag is considered to be the sum of the parasitic drag and the induced drag from the wing. This approximation does not take into account the interference drag caused by the junction of various parts. This is dealt with in section 5.3.3.

5.3.1 Parasitic Drag Coefficient

Parasite drag was estimated using the "component build-up" method. A flat-plate skin friction drag coefficient (C_f) for fully developed turbulent flow is calculated for each major component of the aircraft and then multiplied by a "form factor" (k) that estimates loses due to form drag:

$$C_{dPara} = \sum \left[\frac{k \times C_f \times A_{wetted}}{S_w} \right]_{componet}$$

$$\mathrm{Re} = \frac{V \times L}{\gamma} \qquad C_f = \frac{0.455}{(\log_{10} \mathrm{Re})^{2.56}}$$

Component	A_{wetted} (m^2)	Re	C_f	t/c	k	$C_{dparasitic}$
Wing	2.157	723362	0004902	0.13625	1.07	0.011960
Fuselage	1.305	3255129	0.003744	8.2286	1.08	0.004892
Wheels	0.036	180841	0.006463	0.0625	1.13	0.000247
Gear Struts	0.002	11303	0.012554	1.00	1.80	0.000039
Horizontal Stab.	0.482	452101	0.005366	0.0201	1.10	0.002639
Vertical Stab.	0.198	369216	0.005584	0.0254	1.03	0.001055
Total						0.020832

Table 5.2: Parasitic Drag Estimate

The total parasitic drag estimate is 0.021, which seems reasonable for an aircraft of this shape and size. In order to help reduce drag, the entire airframe is covered with Monokote, a commercial covering material used in the modeling community. It is applied loosely on the airframe and heated with a heat gun shrinking the material to a taut, low drag finish.

5.3.2 Induced Drag Coefficient

The induced drag coefficient is dependent on a proportionality factor "K" and the square of the C_L (for moderate angles of attack). K is dependent on an Oswald efficiency factor "e" and the aspect ratio of the wing. In order to reduce the induced drag, Negative Margin was designed to increase the aspect ratio to a satisfactory number, trying not to extend the span too much, increasing RAC. Winglets and endplates were considered as ways to minimize the induced drag. However, the rules state that the length of the winglets be added to the wing span in determining the overall span. Wing tip endplates have been known to add more interference drag than any reduction in induced drag they may provide.

$$e = 1.78\left(1 - 0.045 \times AR^{0.68}\right) - 0.64 = \mathbf{0.853} \quad K = \frac{1}{\pi A e} = 0.0488 \quad C_{D_{induced}} = KC_L^2 = \mathbf{0.0571C_L^2}$$

5.3.3 Interference Drag

Interference drag arises from the various geometries created by the mating of various aircraft components. At these junctions sharp edges or overlaps add to the overall drag of the aircraft. By utilizing proper drag reduction techniques this form drag may be reduced to a point where it can be considered negligible. Some of these techniques employed on Negative Margin are wing-fuselage fairings, a molded tail surface and wheel pants. The team feels that with these measures in place the interference drag is small enough to be ignored.

5.3.4 Total Drag

The total drag on an aircraft is a direct result of the velocity the aircraft is traveling. While the Reynolds number term in the parasitic drag equation does vary with velocity, its effects are negligible for a calculation of this accuracy. However, the induced drag is heavily dependent on the aircraft velocity due to the C_L term. Thus the coefficient of drag at a known airspeed and C_L is given by the equation $\mathbf{C_D = 0.021 + 0.0571C_L^2}$. With the results of the last equation the total drag force experienced by the airplane is given by $D = 0.5C_D\rho V^2 S_w$.

5.4 Wing Sizing and Performance

5.4.1 Wing Planform

The dimensions proposed in the preliminary design phase were further refined in an attempt to increase the accuracy of the wing performance calculations. The wing area was recalculated using the detailed AutoCAD drawings for sizing, accounting of fuselage carry-over. The span was finalized at **2.65 m (104.5 in)** and a constant chord of **0.41 m (16 in)**. The slight change in span arose from the concern that there would not be enough lift created with a smaller wing using the very marginal power plant chosen. It also is easier to cut the wingtips if necessary rather than add length if needed, the weight added was shadowed by its added lifting ability. With these changes in place the final wing area is **1.08 m^2 (1672 in^2)**. The maximum wing loading with a full complement of 24 softballs is **988 g/dec^2 (30.3 oz/ft^2)**.

5.4.2 Aileron Sizing

The ailerons were sized based on historical data for the effective control of an aircraft. The ailerons were sized using a 50% semi-span and 25% wing chord. The ailerons do not extend all to the wing tip, this prevents flutter caused by wing-tip vortices. The ailerons uses are Frise type ailerons, these create drag when deflected upward, helping to yaw the plane into the turn. The area of each aileron is **0.086 m^2 (134 in^2).**

5.4.3 Stall Velocity

An airfoil efficiency factor must be chosen to relate 2-dimensional airfoil data to a 3-dimensional wing. This efficiency factor, η, was chosen to be 90%. This value accounts for lift losses at the tip and inaccuracies in the airfoil construction. This reduces the max coefficient of lift from 1.32 to about 1.22. Thus the stall speed can be calculated using the following equation:

$$V_{STALL} = \sqrt{\frac{2Mg}{\eta C_{l\max} p S_w}} = 12.8 m/s$$

This results in a stall speed of 11.1 m/s (25 mph), slightly faster than desired but reasonable.

5.4.4 C_L at Cruise and Maximum Velocity

In order to determine the maximum velocity and resulting C_L, several iterations must be carried out. By iterating velocity and C_L values, a final cruise and maximum velocity were found. The cruise velocity was taken to be 90% of the maximum velocity.

	Velocity	Velocity	C_L
V_{CRUISE}	24.0 m/s	54 mph	0.36
V_{MAX}	27.0 m/s	60 mph	0.32

Table 5.3. Maximum and Cruise Velocities

5.4.5 Wing Incidence

With the desired cruising velocity known (calculated to conserve battery power), the angle of incidence for the wing was determined. This angle provides the model with the correct lift in level flight at the cruise velocity. The equation below calculates the angle of incidence accounting for aspect ratio and wing planform.

$$a = \frac{a_0 + 18.24 \times C_{Lcruise} \times (1.0 + T)}{AR} = \mathbf{1.1^\circ}$$

Where, a_0 = the angle of attack of the wing at $C_{Lcruise}$

T = planform adjustment factor for aspect ratio (Fig 4., pg 6, Lennon, Andy, "Basics of Model Aircraft Design")

The pitching moment of the wing at the cruise velocity and angle of incidence was calculated to be **-22.0 N/m (-1556.3 oz/in).**

5.5 Tail Sizing and Performance

5.5.1 Horizontal Stabilizer Platform

The horizontal tail has an area of **0.241 m^2 (373 in^2),** which is 22% of the wing area. The horizontal stabilizer has a rectangular planform, a span of **0.95 m (37.3 in)** and a chord of **0.254 m (10 in)** resulting in an aspect ratio of **3.7**. Again historical guidelines suggest that an elevator area of between 30 and 40 percent for effective pitch control on sportier airplanes. An elevator area of **0.06 m^2 (91.9 in^2)** was used, representing 25% of the stabilizer area, though slightly lower than suggested for a sport plane this elevator will suffice.

5.5.2 Horizontal Stabilizer Lift

The basic guide lines for aircraft modeling state that the horizontal tail be placed at 2.5 times the MAC of the wing from neutral point to neutral point. The neutral point of both wings is considered to be located at the ¼ chord. This guideline would suggest a separation distance of 1.22 m (48 in) for the tail moment arm. However, due to the fuselage sizing influenced by the cargo bay and considerations of RAC the tail moment arm was shortened to 1.02 m (40 in). This leaves the aircraft short coupled and slightly less stable but is well within previous model standards.

With the pitching moment of **-22.0 N/m (-1556.3 oz/in)** and a tail moment arm of 1.02 m (40 in) the horizontal tail must produce a downward force of 30.5 N (42 oz), assuming 85% efficiency of a conventional tail. This down force is provided by the negative lift of the stabilizer airfoil, requiring a

$\mathbf{C_{Lstab} = -0.15}$.

5.5.3 Horizontal Stabilizer Incidence

The angle of incidence to provide this C_{Lstab} was taken from published airfoil data and was shown to be -1.8° for two-dimensional flow. Using the same equation as the wing to account for an aspect ratio of **3.7,** the tail incidence becomes -2.4°.

Another factor that must be considered in the angle of incidence is the downwash from the wing and it's effect on the horizontal stabilizer. The horizontal moment arm along with the vertical distance between stabilizer and wing were calculated and compared to charts (figure 2, pg 40, Lennon, Andy; "Basics of RC Model Aircraft Design" ref 10.) to give the proper tail incidence. A final angle of incidence of –0.82 ° was used.

5.5.3 Vertical Stabilizer Platform

The vertical stabilizer remained unchanged from the preliminary design. A rudder area of 35% vertical tail area was set. This is slightly higher than historical recommendations of about 25% but was desired to provide stability to the pilot in high banking turns. The resultant rudder area was 0.025 m^2 (39 in).

5.6 Propulsion

Motor and propeller selection remained as they were in the preliminary design. A single AstroFlight FAI 40 will provide the aircraft with marginal but sufficient power. An AstroFlight helical gearbox with a ratio of 3.1:1 would turn a 13" by 8" propeller with an approximate pitch speed of 28.4 m/s (64 mph) and provide approximately 27.5 N (6.1 lbs.) of static thrust. At a predicted current draw of **41 amps**, full throttle duration will be of the 18 cell battery pack is 3.3 minutes. The computer radio will control the amp draw to below 40.0 amps to prevent a fuse blowing under static draw. As the prop unloads in the air the

current draw will be reduced. Also, because the majority of the mission will be flown at a partial throttle setting the duration of the battery will be improved.

5.7 G-loading

Two major parameters were considered when predicting the maximum g-load on the airframe. Firstly, the aircraft's structural capabilities were estimated with a calculation of the spar's maximum allowable bending stress. This led to predictions of the accelerated stall characteristics of the wing, aided by published lift data on the Eppler 197 airfoil.

5.7.1 Structural Loading

A **G-load rating of 4** was assigned as a design parameter of the maximum loading the airplane would experience under normal flying conditions, while at the maximum TOGW. From here the size and strength of the wing could be determined.

For calculations, the assumption was made that the wing spar carried all wing loads and that the spar would experince heavier loads than any other aircraft part. Thus, the maximum bending stresses the spar can handle will determine the G-load capability of the plane.

The wing manufacturing plan calls for an "I" beam spar to be used, with a carbon laminate for each flange and a vertical grain balsa shear web. The maximum bending stresses these elements can handle will determine the G-load capability of the plane. As nearly all G-loadings placed on the airframe would be positive, this puts the upper flange of the spar into compression and the lower into tension. Thus the number of carbon fibre laminates would be different for each spar flange. From these assumptions, the cross sectional area of these respective materials could be calculated.

The maximum bending moment experience by the wing was caculated using basic force and moment analysis and found to be **36.8 kN/m (2102 lbf/in)** at the root, dropping to zero at the wing tip. The second moment of inertia (I_y) was calculated for the area of the carbon fiber cross section. With the known distance to the neutral plane (z), the bending stress could be calculated using the equation:

$$\sigma_x = \frac{M \cdot z}{I_y}$$

The dimensions of the upper spars (stock sizes) were then iterated to achieve a safety factor of approximately 1.5 (to account for lamination defects) with positive 4 G loading. It was found that 4 layers of carbon fiber on a 0.0381 (1.5) in by 0.0064 m (0.25 in) piece of balsa was required for the bottom spar. Because the compressive properties of carbon fiber are much lower than its tensile strengh, 7 layer of fiber were necessary for the top spar. In order to be sure that the spar did not fail in shear, vertical grain shear webs were used to join the two spar halves together.

Because the stress in the spar goes to zero at the tip, a tapered spar was used to reduce the overall weight. The spar tapers from 0.0381 m (1.5 in) to 0.019 m (0.75 in) at the tip.

5.7.2 Accelerated Stall Characteristics

The controlled level turning radius of an airplane is determined by the maximum radial acceleration the wing can sustain before and accelerated stall condition occurs. Also, the maximum G-load that can be placed on an aircraft given by the ratio between maximum lift available from the airfoil and the lift generated in steady level flight. It is necessary to determine the angle of bank at which the wing can still produce the required lift for the airplane. It is calculated using the equation below:

$$Cos\beta = \frac{C_{Lcruise}}{C_{L\max}}$$

With the coefficient of lift information know, the **maximum angle of bank is 73.9°.** The maximum radial acceleration before an accelerated stall sets in is given by:

$$Tan\beta = \frac{a_{stall}}{1 \cdot g}$$

This gives a loading of **3.47g's.** Hence, if the aircraft sustains a load more than 3.47g lateral acceleration, the maximum lift of the wing will be exceeded and an accelerated stall will ensue.

From this data the minimum radius of turn with no loss of altitude will be **14.7 m (48.1 ft)**. This was calculated using the following equation.

$$R = \frac{V_{cruise}^2}{a_{stall}}$$

5.8 Take-off Performance

Take-off performance is broken into three segments: ground roll, rotation and climb-out distance. The rotation distance for the small, high lift airplanes considered here is very small and will be assumed negligible in this calculation.

5.8.1 Ground Roll

The ground roll distance is given by: $d_g = \dfrac{V_{TO}^2}{2 \cdot a_{mean}}$

Where, $M \cdot a_{mean} = \left[T_{mean} - \left(A + B \cdot C_{Lcruise}^2\right) \frac{1}{4} \rho V_{Mean}^2 S_w - \mu\left(W - C_{Lcruise} \frac{1}{4} \rho V_{Meam}^2 S_w\right)\right]$

Take-off speed (V_{TO}) is taken as 15% above the stall speed, that is: $V_{TO} = 1.15 \cdot V_{stall}$

C_L and C_D used in this equation are the values at cruising speed. This assumption is valid in that the plane can be considered in level flight while on the ground with its angle of incidence relative to the runway. The static thrust term is estimated from ElectriCalc. This information yielded a ground roll acceleration of **2.15 m/s^2 (7.06 ft/s^2).** This acceleration leads to a ground roll distance of **38.3 m (125.5 ft)**. These distances are given under a no wind condition.

5.8.2 Climb Out Distance

The equation to calculate climb-out angle for an aircraft is: $Tan\vartheta = \dfrac{T}{W} - \dfrac{D}{L}$

Using this equation a **climb angle of only 7.5°** is achieved. If this climb angle was sustained as to keep velocity constant in flight, a distance of **79 m (260 ft)** would be required to reach an altitude of 10 m (32.8 ft). Of course, all calculations made here are for a heavily loaded aircraft. To increase the climb rate, the airplane should be brought to the minimum drag velocity in level flight. The excess power, which exists at this point, may be used to accelerate the aircraft or to aid the climb.

5.9 Endurance and Range

5.9.1 Minimum Drag Velocity

Both induced and parasitic drag are dependent on the velocity of the aircraft, with induced decreasing and parasitic increasing. Therefore there is a velocity at which a minimum drag force is achieved. At this velocity is the point of minimum required thrust, as thrust is equal to drag. The minimum drag velocity is given by:

$$V_{\min drag} = \left(\frac{B}{A}\right)^{1/4}\left(\frac{2Mg}{\rho S_w}\right)^{1/2}$$

In this equation A and B refer to the parasitic and induced drag, respectively. The minimum drag velocity for Negative Margin is **15.64 m/s (35 mph)** at which only **6.74 N (1.52 lbs)** of thrust are needed.

5.9.2 Endurance

An airplane achieves maximum endurance when it is flying at its minimum throttle setting. This setting is attained when the aircraft is flying at $V_{mindrag}$ where the lowest thrust is required. The ElectriCalc software package is used the estimate the endurance of the plane with the selected power plant. It was found that the velocity at which drag is minimized occurs at a throttle setting of 43% maximum. At this setting there is an estimated running time of 24.7 minutes. Due to losses present in the wires, leads, speed controller, motors and the difficulty maintaining a constant velocity, a conservative estimate of 90% system efficiency was used. With a 90% efficiency the endurance of Negative Margin was found to be **22.2 minutes**. This estimate does not account for the energy drawn on take-offs, ground roll, climb, maneuvers and landing. All of these factors significantly reduce the endurance of the power system.

5.9.3 Range

The maximum range characteristics of an electrically powered aircraft differ from those of a gas-powered plane, as motor efficiency drops at increased throttle settings. To calculate the range, the endurance prediction of 22.2 minutes is multiplied by the endurance velocity of 15.64 m/s (35 mph). This method produces a maximum range value of **20.86 km (13 miles).** This range is assuming zero wind conditions and neglects the power needed for takeoff, climb, landing and energy loss maneuvers.

5.10 Stability

Static stability is achieved when the forces on the aircraft created by a disturbance to the flight path push the aircraft back in the **direction** of its original state. Dynamic stability is achieved when the dynamic motions of the aircraft will eventually return the aircraft to its original state (motions are damped). Without the software to perform a dynamic stability analysis to any degree of accuracy, the stability analysis presented here has been limited to static stability. However, for most modes of flight, static stability analysis will be sufficient for an aircraft with a conventional layout such as Negative Margin.

5.10.1 Longitudinal Stability

The maximum allowable distance between the center of gravity of the aircraft and the location of the ¼ chord neutral point of the wing was determined using the following stability criterion:

$$\frac{dC_{M(CoG)}}{dL} = \frac{x}{c} - \eta_H\left(\frac{S_H}{S_W}\right)\left(\frac{l_H}{c}\right)\left(\frac{a_H}{a}\right)\left(1-\frac{d\varepsilon}{d\alpha}\right) + \frac{dC_{Mf}}{dC_L} \leq 0$$

The marginally stable case value of x, the distance from the ¼ chord, was found by setting the above inequality to zero and evaluating. This yielded a value of 44.4%, which means that for the aircraft to be longitudinally stable, the center of gravity can be located at no more than 44.4% of the wing chord from the leading edge. The difference between the actual center of gravity and the maximum allowable center of gravity is termed the static margin. With the predicted center of gravity at 30% of the mean aerodynamic chord, a **static margin of 0.144** is achieved. This is a sufficient value for good longitudinal stability.

5.10.2 Lateral-Directional Stability

In order to provide directional stability, the moments about the center of gravity must be such that the derivative of the yawing moment is less than zero. In order to determine the yawing moment derivative about the center of gravity, the pitching moment derivative of the fuselage and vertical fin must be evaluated. The derivative of the pitching moment of the fuselage in the case of "Negative Margin" will be negligible because the fuselage areas ahead of and behind the center of gravity are equal. The derivative of the vertical fin pitching moment can be expressed by:

$$\frac{dC_{mfin}}{d\theta} = \left(\frac{dC_{Lfin}}{d\alpha}\right)\left(\frac{S_{fin}}{Sw}\right)\left(\frac{I_H}{c}\right)$$

Where $dC_{Lfin}/d\alpha$ is the slope of the lift curve of the vertical fin, S_{fin} is the vertical fin area, S_w is the wing area, l_H is the tail moment arm and c is the mean aerodynamic chord of the wing. The value of the vertical fin pitching moment derivative was evaluated to be 1.6. This value is subtracted from the fuselage pitching moment derivative to determine stability.

$$\frac{dC_{mCG}}{d\theta} = \frac{dC_{mFuselage}}{d\theta} - \frac{dC_{mfin}}{d\theta}$$

The evaluated derivative was found to be 1.6, which is greater than zero indicating that the aircraft was directionally stable.

This results in a **lateral-directional pitching moment derivative of –1.6**, meaning the aircraft exerts a restoring force in the direction opposite the disturbance and is hence statically stable.

5.10.3 Roll Stability

Based on historical data, **6° of wing dihedral** would provide sufficient roll stability for an unswept low wing. As an aircraft with dihedral banks, it begins to slide slip towards the lowered wing, effectively increasing the wings angle of attack and increasing its lift. This provides a restoring force to level the wings.

5.11 Control Systems

5.11.1 Receiver and Programming

Control of the aircraft is provided by a **Futaba 8 channel PCM** transmitter and receiver with its failsafe programmed as per competition rules. Each control surface has its own servo and receiver channel, allowing programmable mixing to be incorporated. Ailerons are programmed to provide differential control for more up throw than down. Slight rudder application is also programmed to operate in conjunction with the ailerons to provide coordinated turns. Also because no flaps were incorporated into the design if more lift or drag is needed on take-off or landing the ailerons may be drooped giving the effect of flaps.

5.11.2 Flight Pack Battery

A five cell, **600mAh, NiMh** battery pack is used to power the receiver and servos. This pack size is large enough to complete several missions, but will be peak charged again after each ten-minute flight. The **6.0V** system increases power to all servos for a minimal weight gain.

5.11.3 Servo Selection

All servos must provide quick, accurate and strong control authority to the flight surfaces. Any slop in the control set-up could lead to flutter, and possible departure of that control surface. To reduce this risk, all servos are dual ball bearing supported. While many mini servos have static load torques capable of moving the flight surfaces, their fragile gear sets are unable to handle sudden dynamic loads. For this reason, standard sized servos with metal gears are used. In order to select the proper servo, the load on each control surface must be calculated. The torque on the servo can be calculated with the equation below.

$$T = 8.5\times10^{6}(C^{2}V_{\max}{}^{2}L\sin(S_{1})\tan(S_{1})/\tan(S_{2})) = oz/in$$

Where T is the torque in oz/in, C is the control surface chord, L is the control surface length, S_1 is the max control surface deflection (assumed to be 20°), S_2 is the maximum servo deflection from center (30° each way for most servos). The maximum torque on each flight surface was calculated. The aileron servos experienced the highest flight load of all the surfaces with a load of **49.5 N/cm (69 oz/in)** each. After viewing many servos and comparing their torque, weight, speed and price, the **Hitec 645 MG Super Torque Metal Servo** was selected. At 6.0V, this servo provides 94.2 N/cm (133 oz/in) at a speed of 20°/sec in 57.5 g (2.1 oz) package. While the power output of this servo may seem like overkill in a static loading, the extra torque provides a factor of safety against dynamic loads. It was decided to use this servo exclusively for the entire aircraft, as the weight differences between other servos were minimal. It also meant that only one type of spare would be needed for the entire aircraft.

5.12 Aircraft Safety

5.12.1 Fuses

In compliance with the contest rules, a fuse will be placed on the motor power line will be used to keep the currents below 40 amps. This protects the motor and speed controller from drawing too much current in the event that the propeller gets jammed. These fuses are the maxi blade type fuses and will be removed from the aircraft to disarm the motors while loading and unloading cargo.

5.12.2 Data Acquisition System

One of the difficulties of flying an electric powered RPV is monitoring the capacity left in the batteries. Low batteries can result in an airplane losing power suddenly in the air. In order to prevent this, an AstroFlight Whatt meter has been incorporated into the aircraft structure. This device will measure the voltage of the battery pack, the amperage draw and the amount of charge that is left the batteries. By recording the number of mAh that were put into the motor when charging, the capacity left in the batteries

can be quickly determined. The ground crew will check this value while changing cargos to determine if there is enough capacity left to perform another sortie. (See Fig 5.1)

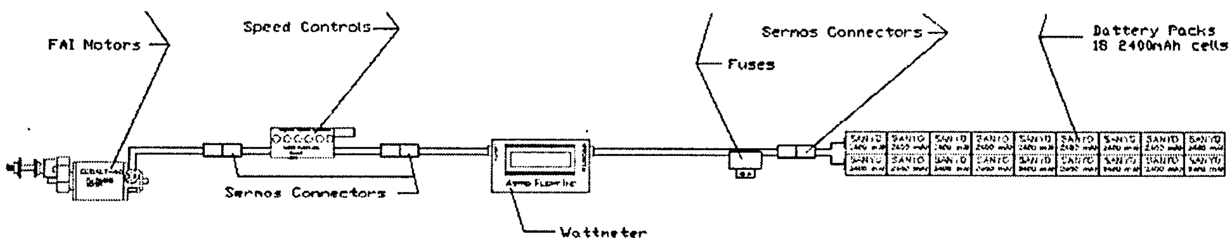


Figure 5.1: Negative Margin's Wiring Diagram

5.13 Estimated Mission Performance

It is important to look at the estimated mission performance to judge how well the plane will perform under actual conditions. This will be looked at in three sections: Payload, time and overall performance.

5.13.1 Payload Performance

The airplane has been designed to carry the full complement of 24 softballs. This was decided upon to maximize the flight score. A spreadsheet was created comparing the main parameters regarding flight scoring: # of softballs and time. It was found that the plane must complete all laps to attain maximum score. For the purpose of the spreadsheet it was assumed that 6 laps were flown so only the time and amount of balls was compared. It was found that time was small factor in the final total score with RAC playing a major factor. From this information the payload carried in Negative Margin will be 24 softballs on the passenger flight. Also, the team felt that because RAC was a large factor we would minimize it. Knowing that battery weight was a very key factor, Negative Margin is carrying only 2.4 pounds of batteries giving it a large advantage over most other teams, which will be competing with bigger motors and heavier batteries. The advantage comes in the final score, where Negative Margin has carried an equal amount of balls, granted it may have been slightly slower but has a lower RAC giving it the edge. See table 5.4 for comparison.

Total Laps Flown	# of Balls	Total Time	Single Flight Score	Report	RAC	SCORE
6	24	7	4.29	92.2	8.61	45.89
6	24	8	3.75	92.2	8.61	40.16
6	24	9	3.33	92.2	8.61	35.69
6	24	10	3.00	92.2	8.61	32.13
6	24	6	5.00	92.2	12	38.42
6	24	7	4.29	92.2	12	32.93
6	24	8	3.75	92.2	12	28.81
6	24	9	3.33	92.2	12	25.61
6	24	10	3.00	92.2	12	23.05

Table 5.4: Comparison of Total Aircraft Scores

NOTES:

The times in the table are in minutes.

No prejudice is meant in the Report scoring it is simply a relative number based on 2000/2001.

The shaded portion is where Negative Margin is expected to stand.

5.13.2 Time Performance

Early into the design it was decided that time was not the most critical factor in the competition. The plane would not be designed to be slow but was not going to give up weight saving or other penalties in the rated aircraft cost to become faster. With Negative Margin designed and the final flight speed estimations carried out, time estimations were made. Video coverage of previous competitions was analyzed and the flight times of various aircraft were recorded. The most interesting fact that came out of this analysis is that slick, sleek designs and heavy, non-aerodynamic designs completed laps only 10 to 15 seconds apart. That is the sleek designs completed a portion of the mission (2 laps) only about 10 seconds faster. Putting this into context of this years competition where a set amounts of laps is required this would result in only a 30 second difference at the end of six laps. All other things being equal, mainly loading/unloading time, the sleek, fast designs with heavier battery packs only gain a small amount in the single flight score. Because of this the rated aircraft cost of this year's plane was minimized.

The actual flight time estimate was made as shown below in table 5.5.

Task	Fast Time (s)	Slow Time (s)
Position Laps (2) (360 turns)	110	130
Loading	30	45
Passenger Delivery (2) (360 turns)	110	160
Unloading	30	45
Return (2) (no 360)	75	90
Total (seconds)	**355**	**470**
Total (minutes)	**5.92**	**7.83**

Table 5.5: Time Estimations for Mission Profile

As is shown in the table the best estimated flight time is about 6 minutes and the worst estimate is approximately 8 minutes. This gives a single flight score range of 5.07 to 3.8. The team feels this is very acceptable as the plane carries the maximum amount of cargo and it is assumed that no team can complete the mission profile in less than 5 minutes.

5.13.3 Overall Performance

Negative Margin is a competitive aircraft. One the team feels has the ability to finish among the top in the competition. The flight times are comparable to the expected competition and the aircraft is capable of carrying the maximum amount of cargo resulting in a high single flight score. It is anticipated that the minimized rated aircraft cost will prove to be the edge Negative Margin has over the opposition.

5.14 Weight and Balances

The center of gravity (CG) is considered to be at the spar of the wing, or located at 30% of the wing chord. Using this the weights of all components were tabulated and their distance to the CG was calculated. Many components were fixed in position restricting them from moving to balance the aircraft. One major component not fixed was the battery pack. The batteries are located above the wing but can be shifted forward and backward in the fuselage as much as six inches to ensure the CG is located at the spar. The cargo bay is configured in such away that the cargo weight is centered above the 33% wing chord. That is that 12 softballs are located fore of the spar and 12 aft. The table below shows that Negative Margin will balance at the expected location.

Description	Weight (lbs)	Weight (N)	Distance to CG (m)	Moment about CG (Nm)
Softballs Fore of Spar	4.500	20.017	0.146	2.922
Softballs Aft of Spar	4.500	20.017	-0.146	-2.922
Battery Pack	2.400	10.670	0.295	3.148
Servos Fore of Spar	0.125	0.556	0.160	0.089
Servos Aft of Spar	0.375	1.668	-0.250	-0.417
Landing Gear Fore	0.250	1.112	0.290	0.322
Landing Gear Aft	0.500	2.224	-0.050	-0.111
Tail	1.000	4.448	-1.020	-4.537
Motor	0.688	3.060	0.325	0.995
Speed Controller	0.075	0.334	0.200	0.067
Reciever Battery	0.250	1.112	0.120	0.133
Prop	0.200	0.890	0.360	0.320
Totals	**14.863**	**66.108**		**0.009**

Table 5.6: Calculations of Moments about CG

As can be seen in the last column of table 5.6 the total moment about the CG is 0.009 Nm, for all intensive purposes there is no moment about the center of gravity. Also, the fuselage is balanced about the CG, the added weight of the reinforced motor mount and firewall offsets the length behind the CG.

5.15 Rated Aircraft Cost

The Rated Aircraft Cost is a model to approximate the cost of the aircraft in dollars. It provides a way to compare the different design costs within the competition, as each team will undoubtedly have different procurement costs. The Rated Aircraft Cost is given by the formula below, with the variables being described in subsequent sections.

*Rated Aircraft Cost, $ (Thousands) = (A*MEW + B*REP + C*MFHR)/1000*

5.15.1 Manufacturers Empty Weight (MEW)

The Manufacturers Empty Weight is the weight of the airframe in pounds, including all flight and propulsion batteries but no payload. In order to convert this weight to a dollar value, the coefficient "A", which has a value of $100, is multiplied by the MEW. Thus for every pound of airframe weight, the Rated Aircraft Cost increases by 0.1. The Manufacturers Empty Weight of "Negative Margin" is **12.25 lbs**.

5.15.2 Rated Engine Power (REP)

The Rated Engine Power is a measure of the amount of battery power being used. It takes into account the amount of engines used and the weight of batteries used. This is given by the equation:

REP = (1+0.25(# engines-1)) * Total Battery Weight*

For Negative Margin this calculation is:

REP = (1 + 0.25*(1-1)) * 2.4 lbs = **2.4**

The REP is then multiplied by the coefficient "B" which has the value of $1500.

5.15.3 Manufacturing Man Hours (MFHR)

The Manufacturing Man Hours value is used to approximate the number of hours it takes to construct the airframe based on its size and complexity. The MFHR is broken down into several sections as dictated by the Work Breakdown Structure (WBS). The sum of all the hours tallied from the WBS is the total MFHR. The coefficient "C" is used to assign the value of $20 per manufacturing hour.

The breakdown of the MFHR is contained below (table 5.6)

WBS	Assigned Hours	Aircraft Dependant Parameter	Hours Subtotal
1. Wings	8 hr / ft wing span	104.5 in (8.70 ft) span	69.6
	8 hr / ft max exposed chord	16 in (1.33 ft) chord	10.6
	3 hr / control surface	2 control surfaces	6
WBS Wing Total			86.2
2. Fuselage	10 hr / ft body	80.0 in (6.67 ft)	66.7
WBS Fuselage Total			66.7
3. Empenage	5 hr / Vertical surface (n/control)	0 Vert. surface n/control	0
	10 hr / Vertical surface (control)	1 Vert. Surface control	10
	10 hr / horizontal surface	2 Horizontal surfaces	20
WBS Empenage Total			30
4. Flight Systems	5 hr / servo	6 servos	30
	5 hr / motor controller	1 motor controller	5
WBS Flight Systems Total			35
5. Propulsion Systems	5 hr / motor	1 motor	5
	5 hr / propeller	1 propeller	5
WBS Propulsions Systems Total			10
TOTAL MFHR			**227.9**

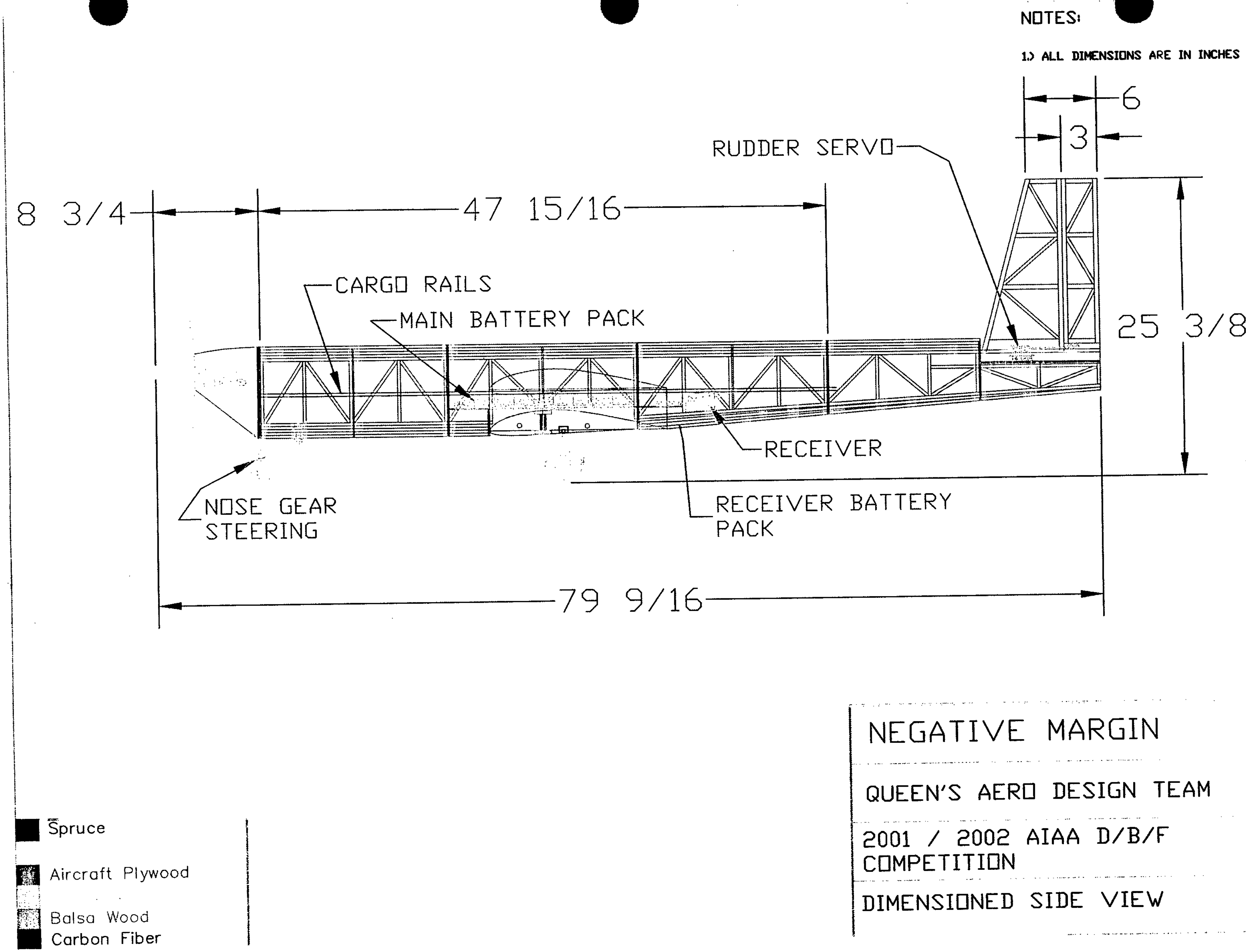
Table 5.6 –Rated Aircraft Cost

5.15.4 Final Rated Aircraft Cost

The Total Rated Aircraft Cost for "Negative Margin" was calculated to be:

*Rated Aircraft Cost, $ (Thousands) = ($100/lb * 12.25 lbs + $1500/lb * 2.4 lbs + $20/hr * 227.9 hrs) / 1000*

Rated Aircraft Cost, $ (Thousands) = 9.383



NEGATIVE MARGIN

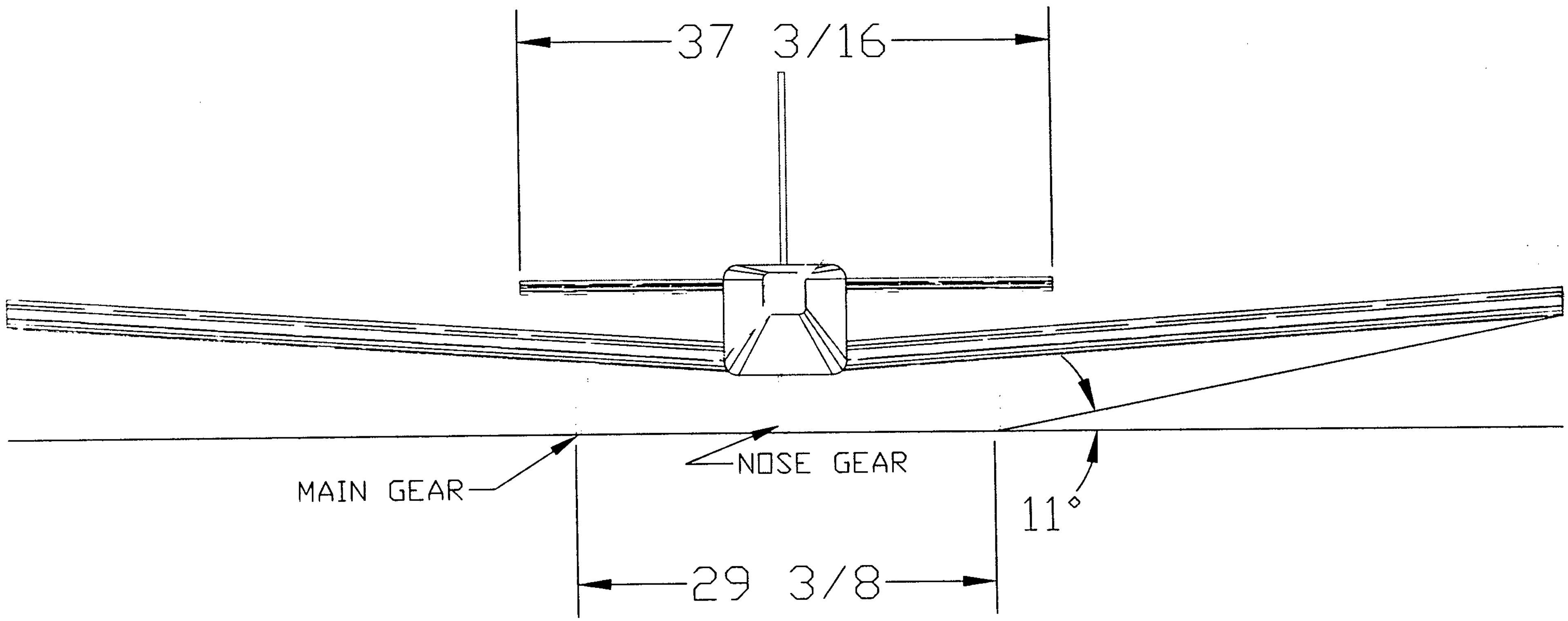
QUEEN'S AERO DESIGN TEAM

2001 / 2002 AIAA D/B/F COMPETITION

DIMENSIONED SIDE VIEW

NOTES:

1.) ALL DIMENSIONS ARE IN INCHES



Spruce
1/16" Balsa Sheeting
Aircraft Plywood
Balsa Wood
Carbon Fiber

NEGATIVE MARGIN

QUEEN'S AERO DESIGN TEAM

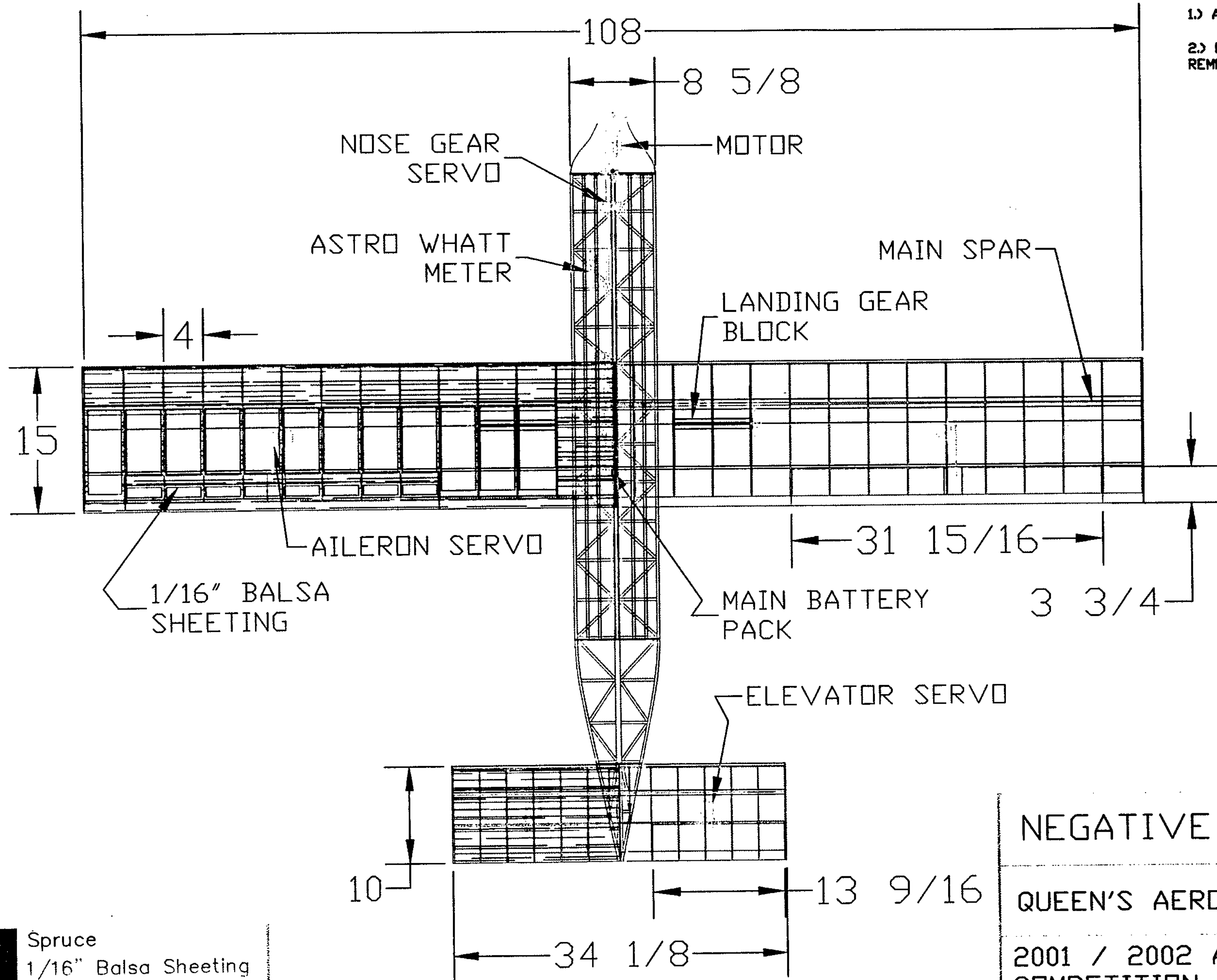
2001 / 2002 AIAA D/B/F COMPETITION

DIMENSIONED FRONT VIEW

NOTES:

1.) ALL DIMENSIONS ARE IN INCHES

2.) RIGHT HAND 1/16" BALSA SHEETING REMOVED FOR CLARITY.



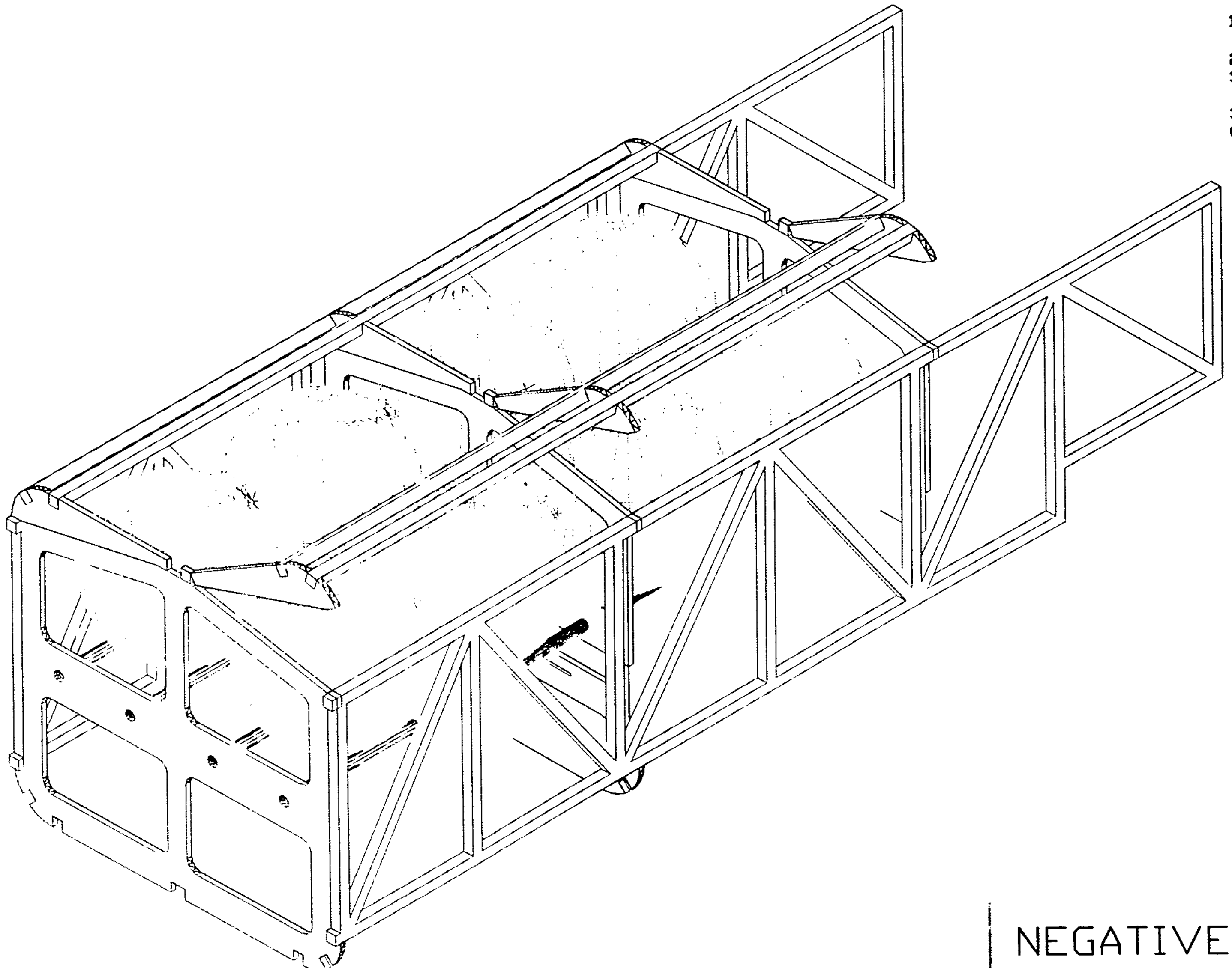
Spruce
1/16" Balsa Sheeting
Aircraft Plywood
Balsa Wood
Carbon Fiber

NEGATIVE MARGIN

QUEEN'S AERO DESIGN TEAM

2001 / 2002 AIAA D/B/F COMPETITION

DIMENSIONED TOP VIEW



NOTES:

1.) ALL DIMENSIONS ARE IN INCHES

2.) VIEW OF FIRST 2 CARGO SECTIONS ONLY.

3.) COVERING AND WING OMITTED FO[R] CLARITY OF VIEW.

- Spruce
- Aircraft Plywood
- Balsa[w]ood
- Carbo[n F]iber

NEGATIVE MARGIN

QUEEN'S AERO DESIGN TEAM

2001 / 2002 AIAA D/B/F COMPETITION

DETAILED CARGO ISOVIEW

6.0 Manufacturing Plan

Negative Margin's design can be broken down into three separate components, each of which employs a different construction technique. Different methods of building the wing, fuselage, and tail appendage were analyzed and the best choices were determined with a figure of merit matrix. The timing of the construction process is documented in a Gantt chart (see figure 6.4) at the end of the section.

6.1 Wing Construction

6.1.1 Foam and Fiberglass

This technique involves cutting a wing from low density foam using a hot-wire cutting apparatus. The wing cores are then strengthened by adding balsa or carbon fiber spars along the top and bottom surfaces. Provision is then made for flap and aileron actuation installation, and then the entire surface is coated with one or two layers of fiberglass. This results in a structurally strong wing without too much effort. The chief disadvantage of this technique is that the weight can become prohibitive.

6.1.2 Built-up Construction

Built-up construction is the oldest and most traditional form of building a wing; unfortunately, it is also the most time consuming. Ribs are cut in an airfoil shape from thin balsa or aircraft plywood and are then positioned on a jig so that there are 4 to 6 inches between each one. The spars, made of balsa, spruce, carbon fiber, or of some combination, are glued in and a shear web of cross-grained balsa is positioned to form the web of the I-beam structure. Leading and trailing edges are formed by gluing balsa to the front and back of the ribs, and the whole structure is then sanded to ensure a streamlined shape. Thin balsa sheeting is applied from the leading edge back to the spar on both the top and bottom of the wing, forming a strong D-tube structure, which is good in torsion. The whole structure is then covered with a thin plastic film to form an airfoil. This technique forms a light, rigid structure.

6.1.3 Carbon Fiber Monocoque

This technique is the most technically demanding of the three choices presented here. An airfoil is drawn up in a 3-D modeling computer program and is transmitted to a computer controlled milling machine. The machine must mill two female molds, one for the top of the wing, the other for the bottom of the wing, from a temperature stable material. The molds are then prepared and pre-impregnated carbon fiber is laid up into the cavity. The mold is then placed under vacuum in an autoclave and baked at approximately 120 degrees Celsius for three hours. Once cooled, the wing halves are released from their molds and are carefully sanded and glued together. This technique requires very complex and expensive facilities, materials, and expertise; however, it results in a very light, strong wing.

6.2 Fuselage Construction

6.2.1 Foam and Fiberglass

This construction technique is something Queen's discovered two years ago while preparing the tail booms of our '99/'00 entries "Obsidian" and "Minnow" and was refined enough to be the chief construction technique for last year's entry, "Absolute Toque". A foam block is shaped and hollowed with a hot-wire foam cutter, then a layer of fiberglass is applied to the exterior. This structure is tough, light, and produces an excellent surface finish. Unfortunately, weight can be a problem on larger structures.

6.2.1 Built-up Construction

This form of construction stretches back to the first days of both full-sized and model aircraft flight. Many thin strips of wood (balsa on model planes) connect several wooden formers to produce the fuselage frame. This frame is then covered by doped paper or silk, or in more recent times, by a shrinkable plastic film. This method is labor intensive, yet produces good results. This type of structure requires extensive repair after a crash, yet can be repaired quickly if necessary.

6.2.2 Carbon Fiber Monocoque

This technique makes use of expensive composite materials to produce a very strong and lightweight fuselage. A mold of the required fuselage shape is made up of a heat resistant material. Several layers of pre-impregnated carbon fiber are laid up onto the mold, a sheet of thin structural honeycomb (which acts as a shear web for the carbon) is placed into the lay-up, and then more carbon fiber is laid up on top. The assembly is vacuum bagged and then heated until the epoxy cures. This technique requires access to expensive materials, equipment, and expertise, but can produce excellent results. The chief disadvantage is that it is nearly impossible to repair after a mishap.

6.3 Tail Construction

6.3.1 Foam and Fiberglass

A strong and smooth airfoil can quickly be made by cutting the required shape from medium density foam and then adding a single layer of fiberglass. The fiberglassed surface is then covered by a sheet of thin plastic and the whole assembly is placed in a vacuum until the epoxy has hardened. The plastic sheets can then be peeled away, leaving a perfect tail surface. While this method of construction can result in a perfectly sculpted complex airfoil, it tends to be heavier than the other options available.

6.3.2 Sheet Balsa

By far the easiest way to construct a tail, thick sheet balsa can be cut in the required plan form and then the edges can be rounded with a sanding block. Although the tail does not take a proper streamlined shape, the extra drag is usually accepted for the ease of construction. This method of construction is durable, but is heavy and prone to warping with changes in temperature and humidity.

6.3.3 Built-up Construction

A built-up tail is the lightest, but most fragile option under consideration. Construction is very similar to a built-up wing, with a set of evenly spaced ribs joined by a double spar and shear web, and the leading and trailing edges. Sheeting is sometimes extended right to the trailing edge to give a slight increase in torsional stiffness. The chief disadvantage of this design is that it is very time consuming to construct. Also worth considering is that the tiny balsa structure that makes up a built-up surface is vulnerable to damage, especially on a portion of the plane that is often accidentally banged and knocked during storage and transportation.

6.4 Figure of Merit

To choose the best combination of manufacturing processes for Absolute Toque, a qualitative figure of merit was conceived to evaluate each technique's merits and weaknesses in an easily interpreted chart. Five criteria were selected, weight, structure, time, skill, and expense, and each construction method was given a qualitative score that illustrates its performance in each category. The separate categories are described in detail below.

6.4.1 Weight

In a high performance competition aircraft, flight performance dictates who will win, and who will be defeated, or worse, who will crash. If the aircraft design is effective and well planned out, then building weight is the one element that can seriously affect every aspect of the flight envelope. Where it is

reasonable, a builder should always strive to make the components as light and efficient as possible. Thus, this was selected as the first criterion in the FOM.

6.4.2 Strength

Structural failure is expensive and can be dangerous under the wrong conditions. To ensure that the aircraft will be able to withstand the loads experienced in flight and on the ground, structural integrity was chosen as the second criterion in the FOM.

6.4.3 Skill

To produce the required components of the aircraft, the selected construction technique must either be known to the team or it must be easy to learn. Also worth considering is that more experience with the specified building technique produces a more accurate final product and less waste, thus it is desirable to choose methods that are familiar to a larger number of team members. Skill was selected as the third criterion in the FOM.

6.4.4 Expense

Among the various construction techniques discussed, there is a huge difference in cost. This is because some techniques use exotic materials or machining, while the more mundane and traditional techniques make use of the builder's individual skill rather than a complex mould or machine. As Negative Margin was built with our meager budget in mind, the cheaper option is often worth pursuing due to fiscal necessity. As such, the expense of the construction technique was chosen as the fourth entry into the FOM.

6.4.5 Time

The final item worth considering when evaluating the construction choices is the length of time that the method requires. Negative Margin was designed and built on a 100% volunteer basis because Queen's does not offer course credit towards participation in a design competition. As all design and construction must be made around the demands of a full engineering course load, time is a precious commodity and was given a place in the FOM.

6.5 Evaluation and Selection

6.5.1 Analytical Method

Each construction technique was evaluated in terms of each of the five criterions listed above. The weight of the method was estimated in ounces. The strength of the method was given a rated on a scale from 0 to 10, with 10 being the most robust choice. The FOM's "skill" category indicates the relative skill level required, rated on a scale from 0 to 10, with 10 requiring the most skill. The cost of the method was estimated using a Canadian dollar value with American funds also indicated ($1.59CDN=$1.00US). The final entry, time, was given a value of the estimated construction hours required for each method.

Total scores were tabulated with the following equation, which weights the relative importance of each of the criterion.

Total = 200/weight+strength/2+20/skill+200/expense+100/time

	Weight (oz)	Strength	Skill	Expense		Time	Total
				Can$	US$		
Wing							
Foam and Fiberglass	96	6	5	180	113.21	30	12.03
Built-up Construction	56	5	6	150	94.34	35	12.35
Carbon Fiber Monocoque	40	5	10	3000	1886.79	100	9.32
Fuselage							
Foam and Fiberglass	160	6	5	150	94.34	35	10.94
Built-up Construction	48	5	6	200	125.79	42	12.13
Carbon Fiber Monocoque	72	5	10	2000	1257.86	60	7.79
Tail							
Foam and Fiberglass	20	6	5	6	3.77	5	68.83
Sheet Balsa	24	7	3	6	3.77	3	83.42
Built-up Construction	16	5	6	4	2.52	5	87.08

Table 6.1 Manufacturing Process Evaluation

6.5.2 Selection

The Figure of Merit (table 6.1) indicated that built-up construction techniques would be suitable for the flight surfaces (wing and tail), and that a foam and fiberglass composite would be the best choice for the fuselage. The methods we employed during the construction of these components are described below.

6.5.3 Testing

Once the decision had been made to employ a built-up wing and tail, the weight-saving campaign was extended to these structures. The area under scrutiny was the leading edge sheeting, which acts to provide torsional stiffness to the wing and to maintain a proper airfoil shape on this critical portion of the wing.

Balsa is an anisotropic material, with much greater axial strength in tension and compression along the grain than across it. As the principle direction of shear that must be resisted in a torsional loading situation occurs at a 45 degree angle, it is logical that if the balsa is oriented so that its strongest axis is coincident with this loading, then less structure could be used to resist the same forces. The conventional leading edge sheeting is oriented axially with the wing to ease construction, so the team felt it would be worth investigating if this sheeting could be replaced by a series of 6.35mm (1/4") square balsa sticks, forming a truss design between the wing's ribs.

The team drew up four simplified models of various wing designs; full leading edge sheeting, top sheeting only, front stick bracing, and full stick bracing (see figure 6.1) and then analyzed them in ANSYS, a commercial finite element analysis program.

Further verification was provided by the construction of four 12" span test panels. These panels were loaded in torsion, and the results were compared to the ANSYS results. As the computer models were simplified to minimize the computer resources required to simulate the loading, the two pools of test data were correlated by their relative performance to the fully sheeted wing as this type of construction is by far the most common built-up wing design. A graph comparing the relative stiffness divided by the relative weight shows that the computer simulation agrees with the real world testing in all cases, and that the conventional fully-sheeted wing provides the stiffest structure per weight of all the designs studied. For this reason, the team chose to remain with the conventional wing sheeting for Negative Margin's flight surfaces.

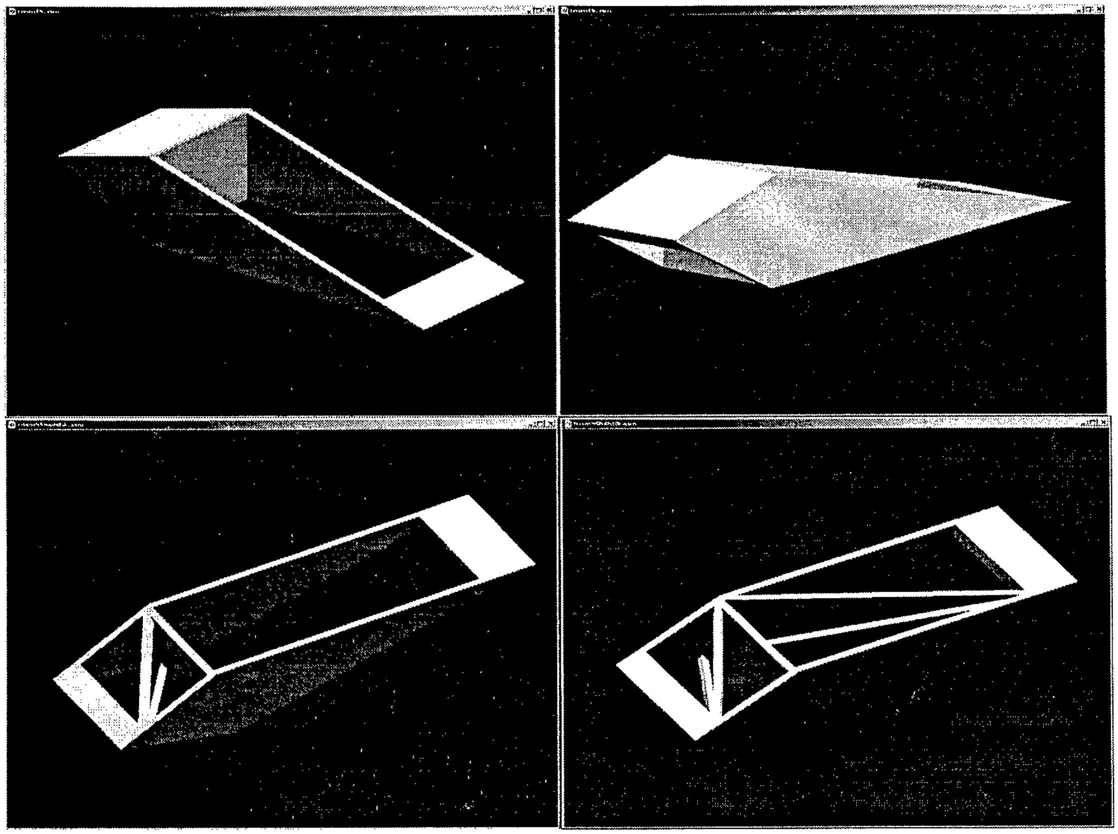


Figure 6.1 –Four solid models tested using finite element analysis

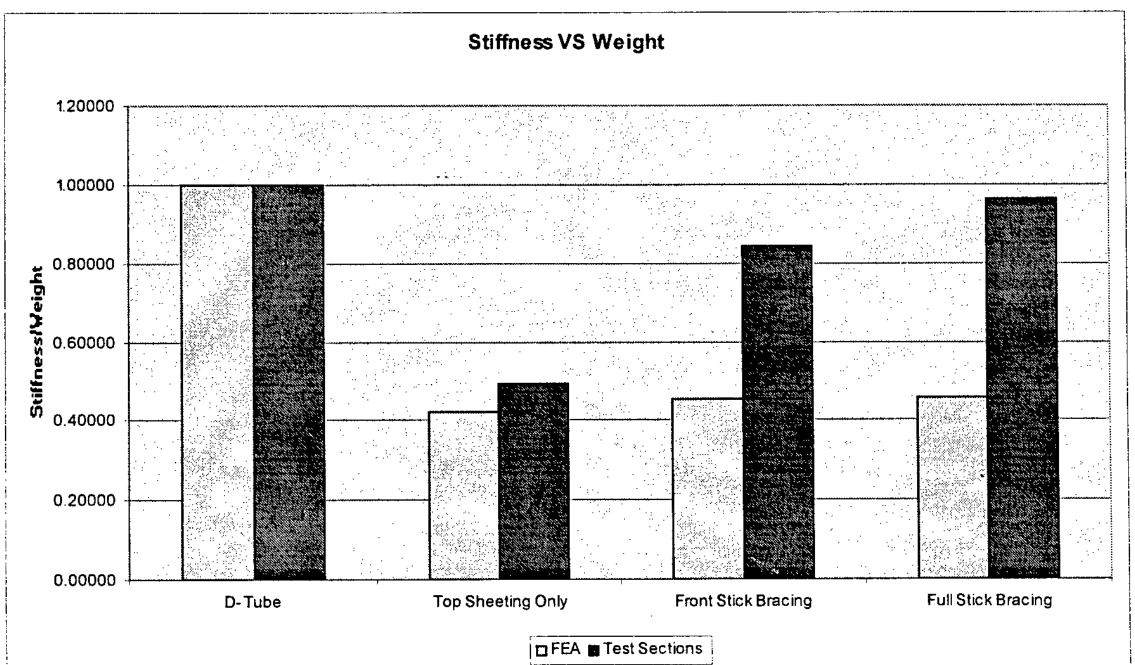


Figure 6.2 –Test results showing relative stiffness/relative weight of the four designs

6.6 Description of Construction Techniques Employed

6.6.1 Flying Surfaces

Once the Figure of Merit indicated that both the wing and the tail surfaces would utilize built-up construction techniques, work began with drafting software to produce the set of working drawings necessary for this type of construction. In a built-up surface, be it a wing or a vertical fin, the airfoil shape is created by cutting out pieces of balsa wood with the proper profile, which are subsequently known as 'ribs'. The exact shape of the ribs depends on their spacing along the flight surface. This spacing can range from 10cm (4inches) between ribs as on Negative Margin's wing, or can be less for the smaller flight surfaces.

The CAD program produces a set of outlines, which can then be cut from balsa with a sharp knife. Once the ribs are cut, it's time to prepare the spars. The spars are made from a laminate of balsa wood and linear carbon fiber, with the carbon fiber being applied in a wet lay-up process and then vacuum bagged until the epoxy sets. The spars are then trimmed to the correct dimensions and then set aside until required.

The next stage of the wing construction is setting up the ribs. The ribs must be aligned on a pair of threaded rods to ensure that their position and spacing is correct. Each rib is held in place by a washer and nut on each side, with a total of four washers and nuts per rib. This arrangement allows minute adjustments to be made before any glue is applied, but can still be removed once the wing has enough structure in place to support its own weight.

With the ribs in place, the bottom spar is glued into its prepared position. The shear webs (their task is to act like the web of an I-beam) are then prepared and glued in their position between the ribs. It's this step where having a rib spacing of 10cm (4inches) becomes important. Balsa is sold in either 7.62cm (3inch) or 10cm (4inch) widths. By spacing the ribs accordingly, time can be saved by not having to trim each shear web. Weight can be saved at this stage by tapering the shear webs' width to account for the reduced loading conditions at the wing tips.

Once the shear webs have been installed, the top spar can be glued in place. This forms the final section of the wing's I beam, and the increase in wing rigidity is apparent as soon as it is in place. The leading and trailing edges are then glued in place, increasing the torsional rigidity of the unfinished wing until the wing sheeting can be installed. The threaded rods of the wing jig are removed, and the resulting wing panel can be handled to install servo mounts, hinge points, landing gear mounts and wing dowels.

"Negative Margin" features a wing built with 3 degrees of dihedral to improve the aircraft's stability. Once the required wing hardware is installed, the left and right wing panels are joined with the aid of a specifically designed wing brace at the spars. The assembly is wrapped in carbon fiber for strength and left to cure.

The newly joined wing is now ready to be sheeted. 1/16" balsa sheet is glued from the leading edge to the top and bottom spar, producing a "D-tube" wing structure. This structure gives the wing its required

torsional rigidity and helps to maintain the wing's proper airfoil contour in flight. The center section of the wing is fiberglassed at this point to help ensure that the wing joint retains the strength for the required mission profile.

Once the fiberglass has cured, the wing is filled and sanded to prepare for covering. Servomotors are installed in their bays, wired, and tested. The wing is then covered with a self-adhesive heat-shrinkable Mylar, which protects the balsa structure as well as bridging the open bays between ribs. Other than the installation of the control surfaces, covering is the final stage in constructing a built-up flight surface.

6.6.2 Fuselage Construction

The construction of the built-up fuselage for Negative Margin began by first constructing a 0.914m (36") long test section. This section was full scale, and helped the team get a feel for how much structure would be required to house the softballs and to maintain a rigid fuselage. Based on our experience with the test section, work began on the full-size fuselage in January.

The main components of the fuselage were designed to be built separately on a flat surface and could then be assembled into a strong, but light, fuselage. Five full-length trusses are built in this fashion; two for the top doors, two for the fuselage sides, and one for the fuselage bottom. All truss joints are then gusseted with 0.4mm (1/64") plywood to distribute the load present at these points. Once completed, the fuselage bulkheads are cut out of 3.2mm (1/8") balsa and aircraft plywood and then they are glued to the bottom truss in the appropriate position.

At this point, a piece of 9.5mmX4.8mm (3/8"x3/16") spruce is glued along the top of the fuselage to further secure the bulkheads. The top and bottom pieces of spruce are connected with a series of diagonal balsa I-beams to form a truss design in the center portion of the fuselage. This truss serves as the backbone of the fuselage, adding strength and stiffness at a minimal weight increase.

With the center truss in place, the fuselage sides are installed by gluing them to the bulkheads in the appropriate place. The tapered tail is formed by carefully gluing the rear section of both side trusses together and then gluing the plywood stabilizer mount in place. 1/4" square balsa longerons are installed at this point to form the rounded corners of the fuselage.

With the fuselage now framed, it is strong enough to handle without fear of damage. Components like the motor mount, battery tray, and electronics cradle can be installed where appropriate. The softballs are supported on four 4.8mm (3/16") dowels, which can be slid into position through the pre-drilled holes in the fuselage bulkheads.

The top doors can now be built right on the fuselage to ensure proper sizing and alignment. Once completed, the hatch latching mechanism is installed and adjusted to ensure a secure and reliable fit and operation.

Further strengthening is added at this point by the addition of two 1.6mm (1/16") plywood plates from the firewall back to the wing saddle. These plates are glued to the outside of the side trusses and are fiberglassed directly to the firewall, strengthening the structure against the loading expected from motor vibration and nose landing gear shock in the case of a less than perfect landing.

With all the structural components in place, the fuselage can be sanded in preparation for the addition of the mylar covering material. The mylar is then applied, providing an aerodynamic shell over the light, but strong, balsa structure.

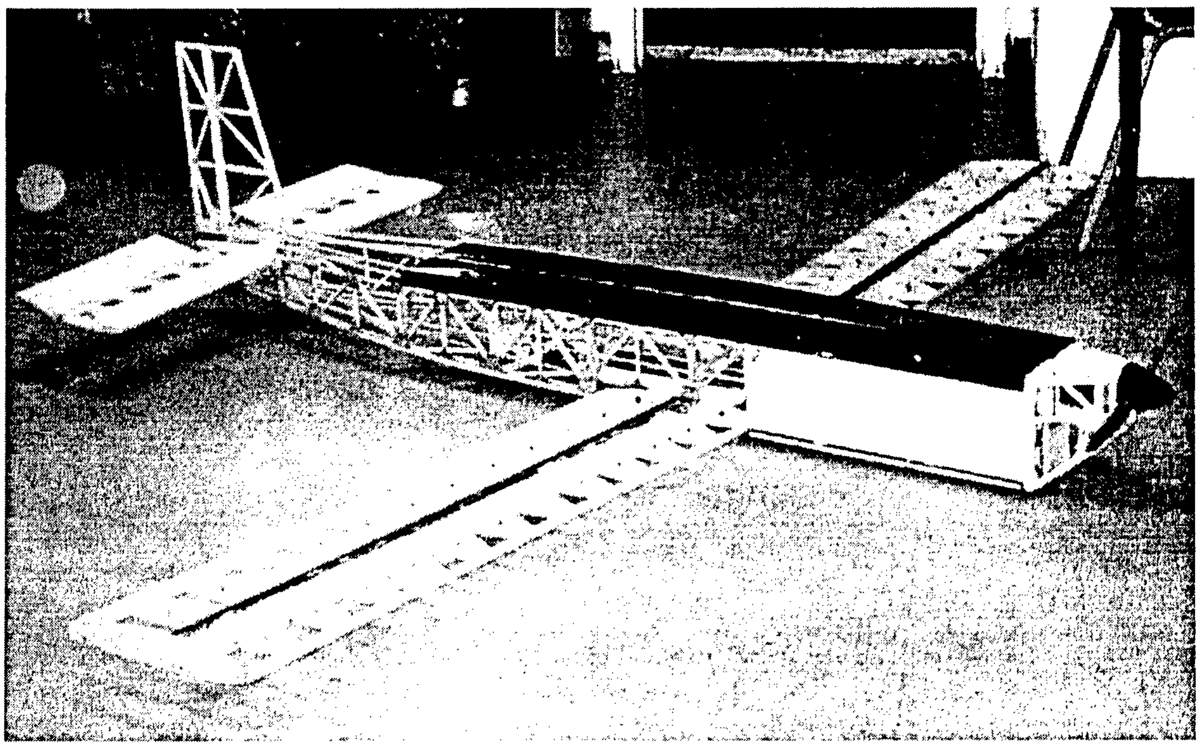


Figure 6.3 –Negative Margin during construction

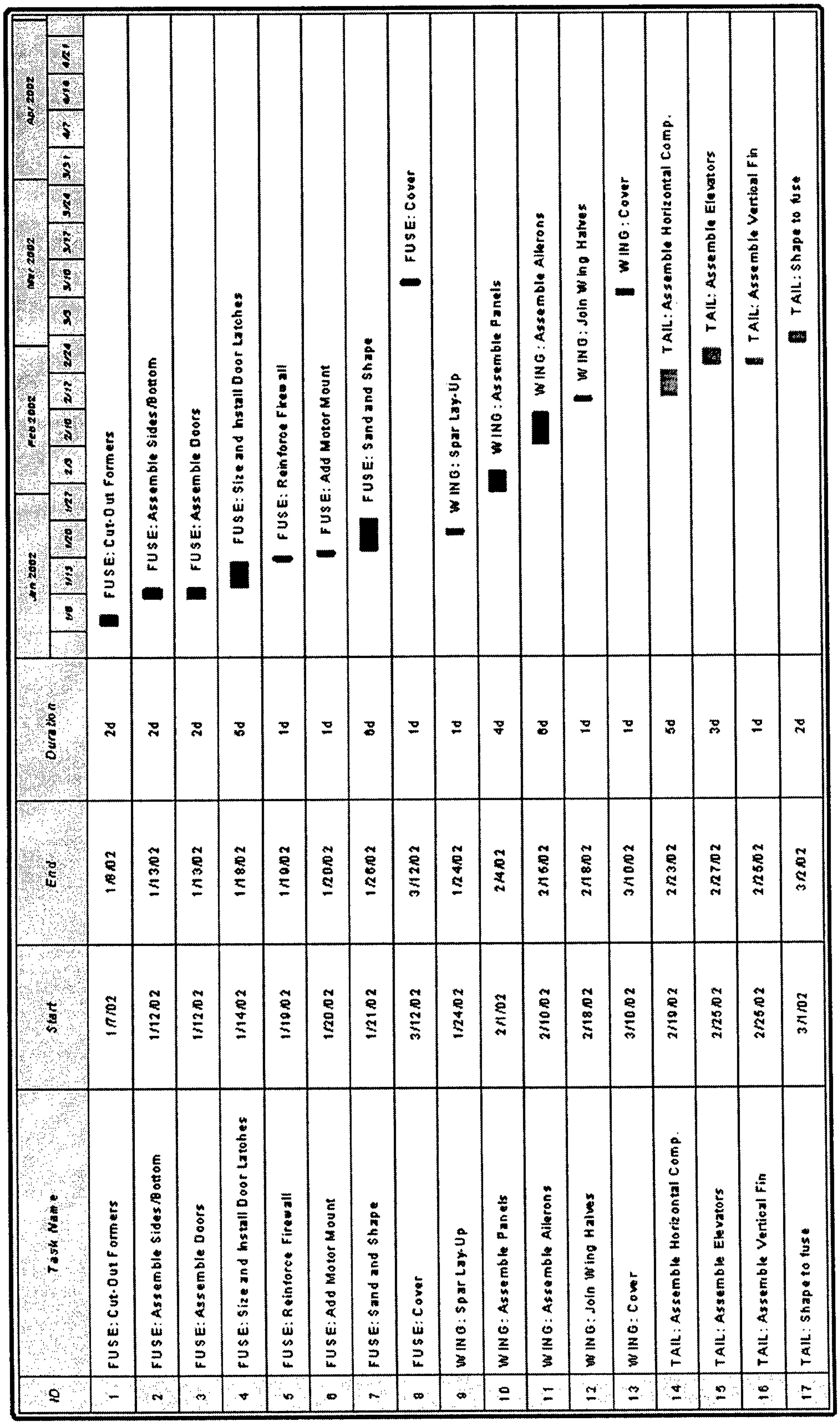


Figure 6.4 –Gantt chart showing construction schedule

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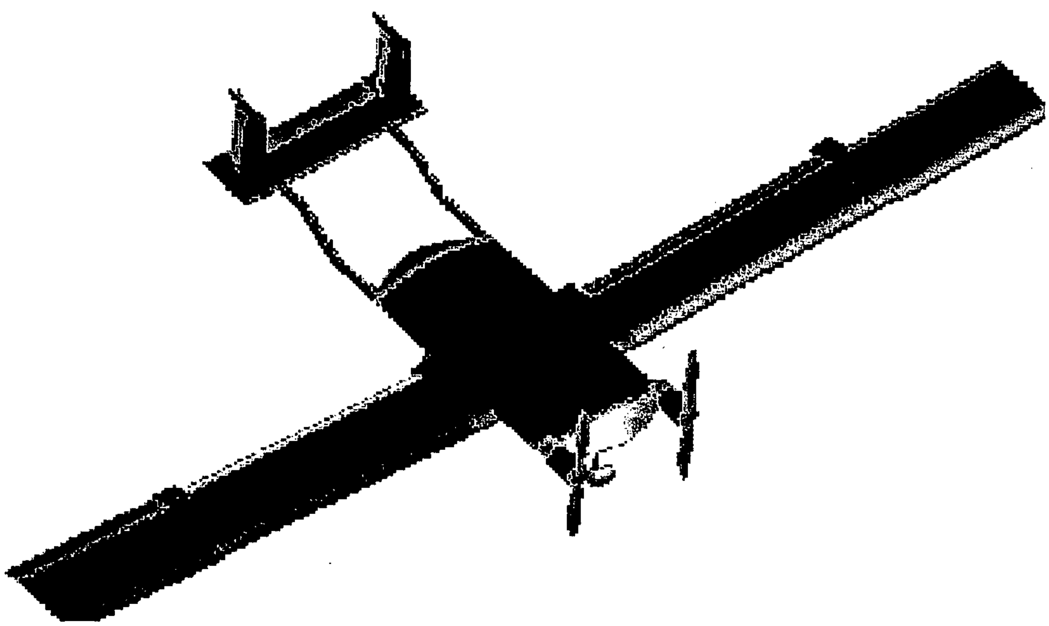
UNITED STATES MILITARY ACADEMY

2001/02 Cessna/ONR Student Design Build Fly
<u>Competition Design Report</u>



USMA

TEAM 1

WEST POINT, NEW YORK
MARCH 12, 2002

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1.0 Executive Summary

The United States Military Academy's department of Civil and Mechanical Engineering sponsored the creation of two student teams to compete for the privilege of participating in this years Cessna/ONR Student Design/Build/Fly competition to be held at Wichita, Kansas on April 27-28th. The purpose of the competition is for students to design an electrically powered, propeller driven, unmanned R/C aircraft capable of carrying a maximum of 24 softballs around a circular course in three 10-minute flight periods.

The goal of this design team is to design, build and demonstrate an aircraft capable of achieving the highest competition score. The competition score is based on the ability of this design teams product to achieve three successful flights, produce a report effectively documenting the design and construction process and meet the lowest rated aircraft cost. AIAA officials will judge all aspects of the competition.

The phases of the engineering design process can be broken down into three parts, conceptual design, preliminary design, and detail design. Descriptions of the three major phases are as follows.

1.1 Conceptual Design

Before analyzing any concepts we broke our team down into five sub-teams in order to facilitate the development of concepts for all aspects of the aircraft. These design teams were; wing, fuselage, landing gear, control surfaces, and propulsion. Each team was tasked to become an expert in their particular field and present the team with the most viable options for combining different ideas into a final concept. Each sub-team's concepts were evaluated based on customer requirements developed by the team leadership. The major aspects of these customer requirements were contest limitations, rated aircraft cost, payload capacity, and construction feasibility. The goal of this phase was to select several configurations of our aircraft for further analysis.

1.1.1 Design Alternatives. Design alternatives can be broken down along the lines of the five major sub-teams. We considered a single wing with either high, low or mid placement. Tapering from the front, back and not at all was analyzed. Fuselage designs were built around the need for maximum payload (set by a historical contest precedence) and designs of many different geometry's and sizes were investigated. The effects of the payload placement on stability and loading were also important design considerations. Parameters relating to the development of landing gear for a tail-dragger or tricycle landing configuration were studied and retractability, braking and ground maneuverability were also considered. Control surface factors that were investigated were the trade off's between separate aileron/flap configuration, blended aileron/flap and no flap configurations. T-tail and conventional tail configurations were researched. Of primary concern was propulsion, we considered 1 and 2 engine configurations spinally mounted and wing mounted. The contest rules severely limiting engine output was a dominating design consideration.

The options investigated were for a plane with an empty weight of between 10 and 20 pounds and a wing span of around 10 feet due to shipping constraints. All options were considered on the basis of their own merit and then in their merit in relation to the other sub-teams options.

1.1.2 Design Tools. The design tools used in the development of our design are those espoused by Mr. David G. Ullman (Oregon State University) in his book the *Mechanical Engineering Design Process*. These consist of design methods such as how/why analysis, mind mapping, functional decomposition, Quality Function Deployment (QFD) and finally relative comparison matrices.

The relative comparison matrices consisted of a table evaluating each concept based on its ability to meet specific engineering requirements drawn from the QFD. Each sub-team developed such a matrix and developed a rank ordered list of viable concepts. The top concepts of each team were then combined to develop several configurations for further analysis.

1.1.3 Design Results. The final concepts selected consisted of front mounted dual engine, single wing, and tricycle landing gear design. Several configurations based on this overall concept were developed. These configurations were developed based on combinations from the relative comparison matrices and were intended to yield as broad a sampling of designs as possible.

1.2 Preliminary Design

During the preliminary design phase we conducted an analysis of the designs selected in the conceptual phase of this design process. The purpose of this phase was to further narrow the scope of the design configuration, develop the spatial dimensions for each part, and select materials out of which to ultimately construct each aircraft component. Analysis in this phase was based upon more extensive numeric and graphical analyses.

1.2.1 Design Alternatives. Many of the airplane characteristics still needed to be determined based upon the results of the conceptual phase. In order to determine these characteristics a thorough analysis of sizing tradeoffs was conducted in order to establish the parameters for refinement. This resulted in new figures of merit with which to analyze the concepts gleaned from the conceptual phase of the design process. With a more thorough understanding of the design of the plane serious consideration was given to the nature of the materials that would be needed. The most important consideration was the strength to weight ratio of the material in relation to the strength required by the design. Once an understanding was gained of the materials which might be used a numerical and graphical analysis of the remaining concepts was conducted in order to final spatial dimension, configuration and materials. Notable figures of merit used to investigate possible configurations in this section included the aircraft lift to drag ratio, thrust to weight ratio, and predicted competition scoring to rated aircraft cost ratio.

1.2.2 Design Tools. During this phase we once again utilized some tools of Mr. Ullman's design process in order to refine our concepts. A second run of the relative comparison matrix was used to select materials and evaluate remaining concepts to further narrow the field. Cosmos Design Star was used to conduct approximate material analysis based on conceptual designs in order to determine feasibility. Iterative processes were conducted using Mathcad and Excel in order to further analyze design concepts and possible component configurations.

1.2.3 Design Results. The result of this phase was the final shape of the airfoil, wing planform area, payload capacity, number of engines, engine brand, as well as dimensional and material specifications.

The end state of this phase left us with a design constrained to a single wing, dual engine, tricycle landing gear, maximum payload blended wing body configuration. With this design it was possible to move on to the detail design phase in order to determine specific performance parameters.

1.3 Detail Design

The purpose of this design phase was to develop performance and handling analyses and optimize based on the mission profile. This section would take values of certain performance parameters (such as wing aspect ratio, tip to tail length, etc.) that had been assumed in previous analyses, and optimize them based on performance and handling characteristics. The calculations done up to this point were then updated with the optimized performance parameters forming an circular process. Therefore, in actuality, calculations from the preliminary and detail design phases were done in tandem and the ultimate parameter design was dependent on iterating between the analyses done in both sections. Finally, in the detail design phase, the final concept was conveyed through an extensive drawing package.

1.3.1 Design Alternatives. Alternatives in this phase of design dealt mainly with the finer point of the aircraft design. The payload securing device and the nature of component connections were considered and the assumptions we had been using for performance oriented dimensions were to be refined slightly to achieve specific performance specifications and stability requirements. All connections were designed and refined given construction abilities as well.

1.3.2 Design Tools. In the detail design phase extensive numeric comparisons were conducted. Mathcad, as well as hand calculations, were used extensively to determine performance comparisons for optimization.

1.3.3 Design Results. The result of this phase was a specific design that was ready to be built. Every connection had been finalized and every sub-teams design had been checked for compatibility with the overall design. Performance and handling characteristics were predicted and the aircraft's ability to complete the given mission was assessed.

2.0 Management Summary

2.1 Introduction

In August of 2001, ten undergraduate engineering students at the United States Military Academy were formed into a project team with the goal of designing and manufacturing a remotely controlled aircraft to compete in the Cessna/ONR Student Design/Build/Fly Competition to be held in Wichita, Kansas 26-28 April 2002. The original ten students (nine seniors and one junior) were responsible for the creation and design of the project and consisted of eight students majoring in Aerospace Engineering, one in Mechanical Engineering, and one majoring in Electrical Engineering. These students comprised four sub project teams which were supplemented as the year progressed and underclass assistants were recruited. The entire project was overseen by an instructor in USMA's Mechanical Engineering Department.

2.2 Architecture of the Design Team

A project manager was appointed from the design team in order to organize the project, direct the team's efforts, and take on the responsibility of the project success. In order to ensure the members of the engineering design team were given focused, specific assignments, the project manager decided to divide it into specific sub entities based on their expertise and personal interests. The organization of the design team is summarized in Table 2.1.

Project Manager		Patrick Schmidt
Project Executive Officer		Brad Townsend
Wing / Dynamic Control	Team Leader / Wing	Kevin Andreson
	Stability / Control Surfaces	Andrew Johannes
Fuselage	Team Leader / Fuselage	Nate Thompson
	Payload Considerations	Ryan Kelly
Propulsion	Team Leader	Sam Jin
	Electronics / Power Plant	Nick Ryan
	Thrust	Brian Schapker
Landing Gear	Team Leader	Jerrod Adams

Table 2.1: Organization of the engineering design team.

2.3 Configuration and Schedule Control

In configuring the engineering design team, the project manager ensured that each member possessed a well-defined role and an area of the project to focus on. Team leaders were responsible for supervising the progress of their respective team members, overseeing construction of their components, and coordinating their efforts with the leaders of the other teams. As for all other team members, Table 2.2 summarizes their specific duties and responsibilities.

Project Manager	Responsible for accomplishment of the overall project mission
	Plans and organizes the project
	Ensures communication and cooperation between sub project teams
	Coordinates full group meetings
	Serves as liaison between team and USMA faculty advisor
	Prepares the project report
	Primary supervisor of aircraft design
Project XO	Assumes control of project in the absence of the project manager
	Manages team budget and finances
	Assists the project manager in preparing the project report
	Primary supervisor of aircraft manufacture
Wing	Oversees design and manufacture of lifting surfaces
Dynamic Control	Designs and constructs means of stabilizing and controlling aircraft flight
Fuselage	Oversees design and manufacture of the fuselage
Payload	Responsible for planning the loading and transport of the mission payload
Electronics	Designs the electrical configuration and responsible for overall power plant design
Thrust	Designs means for converting power plant energy to aircraft propulsion
Landing Gear	Designs means for aircraft recovery and taxi

Table 2.2: Duties and responsibilities of the engineering design team.

All members of the engineering design team were essential to the overall project success. Figure 2.1 illustrates the contributions made by each member and the team effort that characterized the project as a whole.

In order to ensure project organization, a detailed timeline was developed by the project manager in early September. This schedule consisted of a chart detailing the timeline of the engineering design process and outlined key steps along the way. Figure 2.2 shows the overall project schedule including the project design timeline. Both the planned and actual dates of each project milestone is illustrated in this chart. Members of each sub project team developed a more detailed schedule laying out the project building timeline which is presented later in the manufacturing summary of this report.

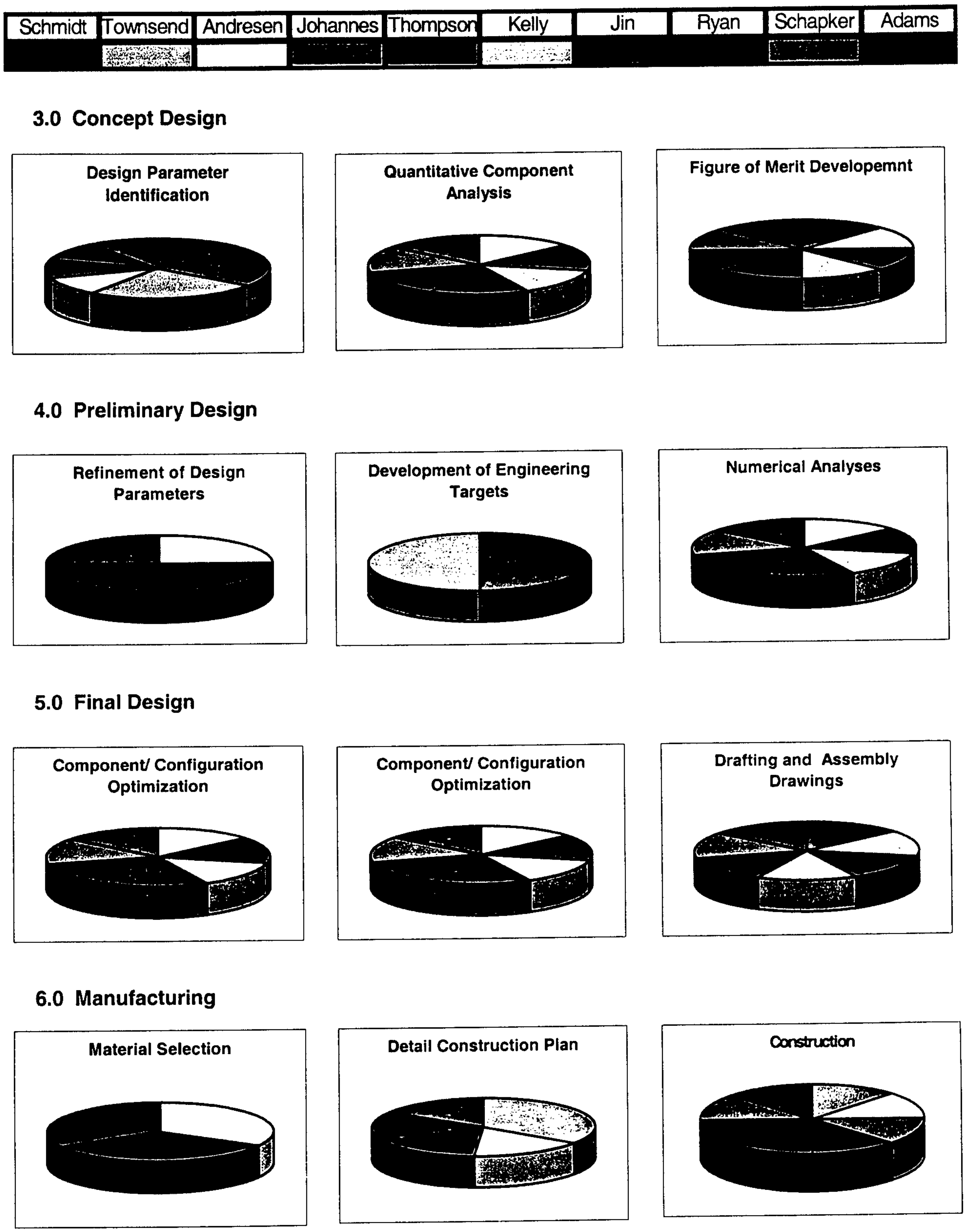


Figure 2.1: Contributions of each design team member to the overall project success.

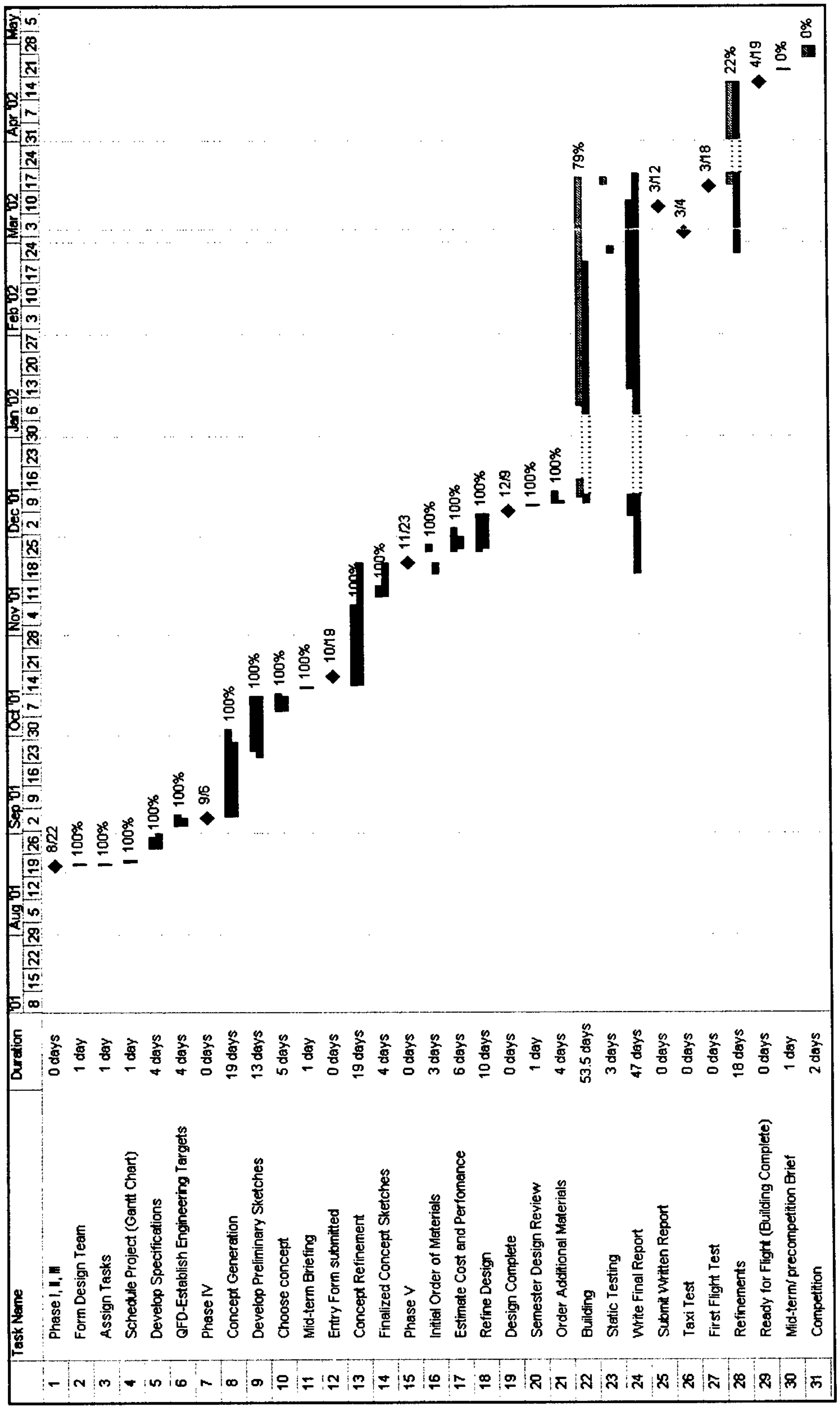


Figure 2.2: Overall project schedule and timeline of the design process. The blue bars denote the planned timeline while the red bars represent the actual timeline of events. Events in progress are shaded light red up to the present time and light blue up to the future completion date.

3.0 Conceptual Design

3.1 Introduction

The purpose of this phase of the design process is to develop several concepts that can be further analyzed in the following design phase. The first step in developing concepts for analysis is to determine the exact nature of the problem to be solved, understand the scope of the problem, determine what device must be built to design the problem and finally to develop specifications for the device that meet customer requirements. The final step in this design phase is to use the engineering specifications to develop a broad range of concepts and then compare them to the customer requirements (figures of merit) in order to determine several tentative designs for further analysis in the preliminary portion of this design.

3.2 Understand the Problem

The first step in understanding the problem was to conduct a How/Why analysis. A How/Why analysis (also called an objective tree analysis) consists of a tree diagram that builds around answering why a problem must be solved and then how we must solve that problem. This tool gave us a better understanding of the problem and helped to develop a problem statement. The result of this step was an initial problem statement that would be used to guide the rest of our design process. This problem statement is: Develop an R/C aircraft capable of carrying the largest payload in the shortest amount of time.

The next step in the design process is the construction of a mind map. This consists of a process very similar to brainstorming in which the scope of the problem is analyzed. This analysis helps to recognize aspects of the problem which need to be considered throughout the design process. These included interrelationships between aspects of the design, possible customer requirements as well as possible functions.

The culmination of these first two steps is the functional decomposition, ref. Figure 3.1. Here the overall design is broken down into the five major subgroups (wing, landing gear, fuselage, propulsion and control surfaces) for a more detailed analysis of the specific functions that must be performed by each device in order for it to be successful in the overall design. For each major functional group associated with the aircraft a functional decomposition was developed which allowed for the determination of specific sub-functions for each functional group. This completion of this phase results in a complete understanding of the problem and the specific tasks that must be met in order to successfully complete the task.

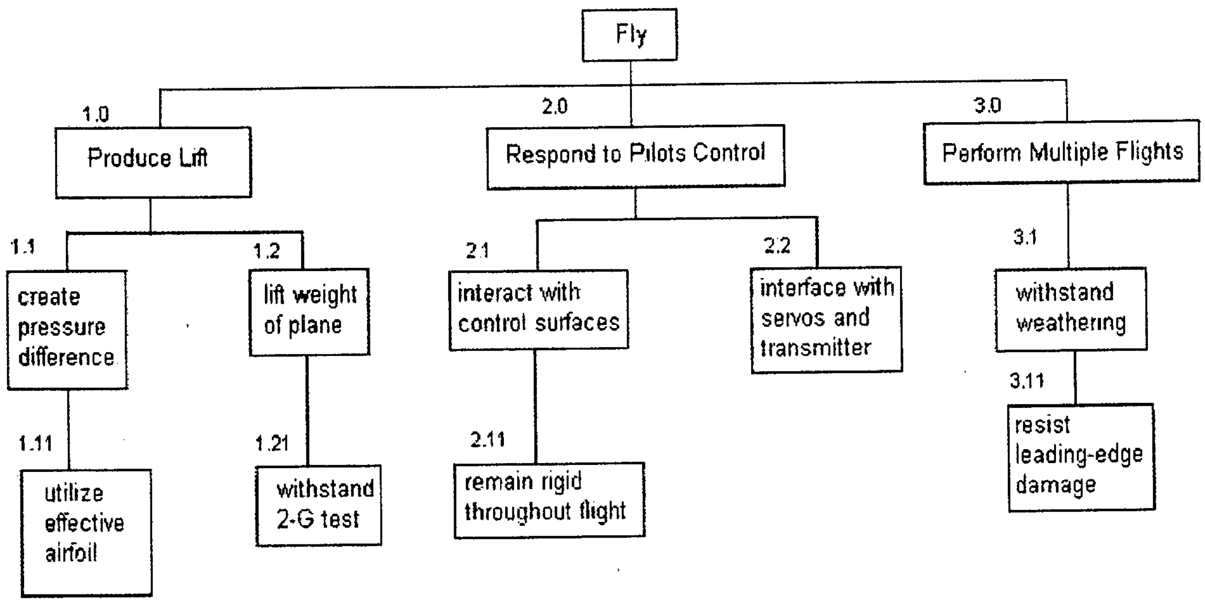


Figure 3.1: Functional Decomposition for the wing sub-team

3.3 Figures of Merit

The goal of our design is to satisfy our customer requirements. The first step in satisfying these customer requirements was to develop a set of engineering requirements (figures of merit) based off of our customer requirements using a tool called a Quality Function Deployment (QFD). Once we have developed figures of merit we can then use these to develop concepts that we can compare to our customer requirements in order to determine success.

3.3.1 Identification of Customers. Determining customers requires us to identify all individuals or organizations who will use or review our design. In this situation we are designing a product for a competition. Therefore, our primary customers are the judges of this competition. They determine the rules of the competition and measure our success or failure.

3.3.2 Determination of Customer Requirements. In order to determine our customer requirements we examined the competition and its rules for performance requirements. Then we developed some of our own customer requirements based on what we saw as needs for the aircraft based on our goal, to ultimately win the competition. These customer requirements included performance parameters, construction requirements and cost. Refer to the QFD, Figure 3.2, for a complete listing.

3.3.3 Relative Importance. It is not possible to design something that meets all customer requirements equally. Trade-off's in design are almost always a necessity. Therefore an importance must be assigned to each customer requirement so that we can better develop a successful design. This was done by rank ordering our customer requirements from 1 to 15.

3.3.4 Benchmarking. Before developing a new design it is always a necessity to evaluate existing designs. This was done primarily by analyzing the two teams that competed from the United States

Military Academy last year. We ranked their planes based on their success in meeting our customer requirements in order to develop a better idea of where we needed to improve on their designs.

3.3.5 Figures of Merit. Once the benchmarking is complete and the customer requirements are selected it is possible to develop figures of merit. These are specific engineering requirements that represent qualitative aspects of each of the customer requirements. Each engineering requirement was correlated to a specific customer requirement.

3.3.6 Assessment of the Competition. Through a review of last year's designs, we developed an idea of the level of performance that our aircraft needed to achieve. In this we looked at specific design parameters of last years aircraft in relation to our engineering requirements in order to help develop a better design. There is no point in designing an aircraft that does not improve on previous designs.

3.3.7 Establishment of Engineering Targets. Comparing each engineering requirement to each customer requirement allows us to establish targets for our design. The weighted and relative importance of each engineering requirement was then determined by its ability to meet the most customer requirements successfully. Once this was established the hard values for the competition were entered. With all of the above values determined we could then establish a final set of engineering targets for our design that would allow us to successfully meet our customer requirements.

AIAA: Design Build Fly Project		Customer Importance	Propulsion: Thrust to Weight Ratio	Propulsion: Battery Charge Time	Propulsion: Number of Flights Before Battery Recharge	Propulsion: Time of Flight	Propulsion: Engine Weight	Fuselage: Time to Load / Unload	Fuselage: Drag Profile	Wing/Lift: Wing Structural Strength	Wing/Lift: Drag Coefficient	Wing/Lift: Lift Coefficient	Overall Structure: Absorb Shock	Overall Structure: Aerodynamic Efficiency	Overall Structure: Number of Flights To Failure	Customer Assessment (⊗ USMA '01 Modular; ✱ USMA '01 Balsa Bandit) 0 1 2 3 4 5
Modular Structure	Low Weight	8.0	◉	○	○	○	◉			○		◉	○	○		⊗✱
	Transportable Wing Spz	7.0								○				○		✱
	Durability / Survivability	6.0								◉			◉		◉	⊗ ✱
	Easy to Repair	4.0								◉			◉		○	✱ ⊗
	Simplicity of Manufacturing	6.0					△	△		○	△	△				✱ ⊗
	Effectiveness of Subcomponent Integration	10.0					△	△	△	◉			○	○	△	✱ ⊗
Payload / Capacity	Maximum Number of Softballs	7.0	○		△	△		◉				○				✱
	Ease of Loading / Unloading	8.0				◉		◉								✱ ⊗
	Security of Payload in Flight	5.0				△		○								✱
Cost	Low Cost	9.0		◉	○											✱ ⊗
Engine	Electric Motor															✱
	Lightweight Powerplant	5.0	○	◉	◉	○	◉					○			△	✱ ⊗
	Provide Maximum Speed	6.0	○	◉	○	◉	○		○	△	◉	○		◉		✱
Flyability	Minimum Takeoff Distance	3.0	△	○		○	△		△	○	○	◉		◉		✱
	Fail Safe Switch															✱
	Easy to Maneuver	6.0				○	△		○	◉	○	○		◉	◉	✱ ⊗
	Standard Controls	4.0				△				○					○	✱
Manufacturability	Availability of Parts	8.0					○			○					△	⊗ ✱
Weighted Importance			129.0	213.0	121.0	208.0	184.0	166.0	49.0	348.0	87.0	177.0	144.0	210.0	195.0	
Relative Importance																
USMA '01 Balsa Bandit			388	20	1		2.5	25		54	108	1.1	140	12.2	N/A	
USMA '01 Modular			37	20	1		2.5	25		55	1.56	1.0	55	7	3	
Units			N/A	min	# flights	min	lbs	sec	N/A	lbf	N/A	N/A	lbf	L/D Ratio	# flights	
Engineering Targets			.45	20	1	8	2.5	60	.024	55	.080	1.25	130	14	N/A	

Figure 3.2: Quality Functional Deployment. Customer requirements or figures of merit are listed along the side of the chart while engineering requirements are listed along the top of the chart.

3.4 Concept Development

Once the engineering targets have been developed the next phase of the design can begin. With the overall engineering targets in hand and knowledge of the customer requirements each sub-team broke down to develop as many concepts as possible. Once multiple concepts had been developed a 3-step analysis of each concept was conducted in order to narrow down the field to a reasonable number overall number of concepts.

3.4.1 Feasibility Judgment. In the first step of concept development the designer needs to decide if the design concept is reasonable for our design. The primary factor that needs to be considered here is whether or not it is reasonable. Fanciful, unnecessarily complicated or prohibitively expensive concepts were removed from consideration.

3.4.2 Technology Readiness. An important aspect to consider in all concept developments was the ability of the design team to make the concept reality. Once again designs that would require far too much time investment or were simply impossible to build with the design teams limited experience were removed from consideration.

3.4.3 Go/No Go Analysis. The final step in concept development was to compare the concepts against our engineering and customer requirements. In order to be considered as a concept a concept had to meet these requirements or it was simply removed from consideration. The goal of all of these steps is to eliminate frivolous designs and simplify our decision matrix which would determine the final design concepts.

3.5 Concept Determination. The final and most important phase of the conceptual design phase is the determination of the final concepts. The method used for concept determination is known as a decision matrix or Pugh's method. This method consists of a matrix for relative comparison of concepts on which the customer requirements are on the y-axis and the concepts themselves are listed on the x-axis. This is the point where the concepts developed using the engineering requirements (figures of merit) are measured against the customer requirements for success. The measurement process consists of ranking each concept's success at meeting each relevant customer requirement on a scale of 1 to 5. Where:

1- little success at meeting customer requirement
2- weak success at meeting customer requirement
3- moderate success at meeting customer requirement
4- successful at meeting customer requirement
5- very successful at meeting customer requirement

The score for each concept in each customer requirement was then compared against that customer requirement importance. The overall score for each concept was summed and the top concepts were selected using this method.

3.5.1 Wing. The wing is the most important aspect of any aircraft. The competition rules limited the concept development for the wing to two basic variations, multiple or single wings. Multiple wings offer

several advantages when rated aircraft cost is determinant on wing span. However, the rules this year base rated aircraft cost for the wing on planform area. Because of this and the inherent difficulty involved in positioning and controlling the wings in an aircraft with multiple lifting surfaces, our engineering requirements limited us to the single wing theme. There were also several factors to consider, the location of the wing on the aircraft, the shape or taper of the wing and the amount of dihedral to be built. The mounting location of the wing could vary relative to the fuselage. The distribution of weight on the wing dictates the aircraft maneuverability meaning a low placed wing mounted under the fuselage would be more maneuverable while a high placed wing mounted on the top of the fuselage would be less maneuverable. Furthermore, the stability of the aircraft also depends on this wing placement. The high wing lends a greater deal of stability than the low wing. This phenomenon is commonly illustrated thorough the analogy of a pendulum which would be obviously more stable with the weight distribution below the pivot or support point as opposed to above. A compromise between these two choices is a mid-wing positioned along the centerline of the fuselage providing a balance of maneuverability and control.

Another factor to consider was the shape or taper of the wing. The most efficient wing would have been shaped so as to match the shape of the lift distribution across it. However, the lift distribution on aircraft wings is elliptical due to vortex flows at the wingtip and such a shape is incredibly difficult to manufacture. A straight wing would be simple to construct, but subject to inefficiencies based on increased induced drag from the wingtip vorticies. A tradeoff between these two considerations was a wing tapered from either the front or rear to approximate the elliptical planform shape but maintain a degree of manufacturability.

Yet another factor to consider was the amount of dihedral built into the wing. Dihedral is angle built into the wing in the positive direction (up) so that the lift vectors of the wings are oriented inboard so as to provide roll stability to the aircraft. Dihedral is difficult to manufacture and can degrade wing efficiency, however it provides invaluable stiffness to the aircraft along the longitudinal axis.

The different permutations of these three factors were then compared using a relative comparison matrix in order to determine the best overall concepts. Reference Table 3.1. The two final concepts developed using this method were a high and a medium wing with a straight edge and a low dihedral. The straight wing was the most successful design due to its ease of manufacture and the inherent strength associated with a solid wing (a tapered wing would have to be built in sections). The benefits of a tapered wing in such a low speed aircraft were considered to be insignificant and the ease of manufacture parameter dictated the straight wing decision.

	Weighting	High Wing, Taper, High Dihedral	High Wing, Taper, Medium Dihedral	High Wing, Taper, Low Dihedral	High Wing, Straight, High Dihedral	High Wing, Straight, Medium Dihedral	High Wing, Straight, Low Dihedral	Medium Wing, Taper, High Dihedral	Medium Wing, Taper, Medium Dihedral	Medium Wing, Taper, Low Dihedral	Medium Wing, Straight, High Dihedral	Medium Wing, Straight, Medium Dihedral	Medium Wing, Straight, Low Dihedral	Low Wing, Taper, High Dihedral	Low Wing, Taper, Medium Dihedral	Low Wing, Taper, Low Dihedral	Low Wing, Straight, High Dihedral	Low Wing, Straight, Medium Dihedral	Low Wing, Straight, Low Dihedral
Low Weight	15	5	5	5	4	4	4	5	5	5	4	4	4	5	5	5	4	4	4
Transportable Wing Span	4	2	3	4	2	3	4	2	3	4	2	3	4	2	3	4	2	3	4
Durability / Survivability	12	3	3	3	4	4	4	3	3	3	4	4	4	3	3	3	4	4	4
Ease of Repair	5	2	2	2	4	4	4	2	2	2	4	4	4	2	2	2	4	4	4
Simplicity of Manufacture	6	1	1	1	4	4	4	2	2	2	5	5	5	1	1	1	4	4	4
Effectiveness Component Integration	10	4	4	4	4	4	4	3	3	3	3	3	3	1	1	1	3	3	3
Maximum Payload	14	2	3	4	3	4	5	2	3	4	3	4	5	2	3	4	3	4	5
Ease of Loading / Unloading	1	2	2	2	2	2	2	3	3	3	3	3	3	4	4	4	4	4	4
Security of Payload During Flight	9	3	3	3	3	3	3	4	4	4	4	4	4	3	3	3	3	3	3
Low Cost	2	3	3	3	4	4	4	3	3	3	4	4	4	3	3	3	4	4	4
Lightweight Powerplant	8																		
Maximum Speed	13	3	4	5	3	4	5	3	4	5	3	4	5	3	4	5	3	4	5
Minimum Takeoff Distance	3	1	2	3	3	4	5	1	2	3	3	4	5	1	2	3	3	4	5
Ease of Manuever	11	1	3	4	1	3	4	1	3	4	1	3	4	1	3	4	1	3	4
Standardized Controls	7	1	2	3	3	4	5	1	2	3	3	4	5	1	2	3	3	4	5
		##	361	413	359	##	474	##	367	419	365	##	##	270	##	385	351	414	##
		16	11	8	12	5	2	15	9	6	10	4	1	17	14	8	13	7	3

Table 3.1: Relative Comparison Matrix for the Wing

3.5.2 Control Surfaces. The control surfaces are another important aspect of aircraft design and in this case included both the main wing control surfaces as well as the stabilizing surface of the aircraft. During the concept phase two basic types of stabilizing surfaces were considered: a canard fashion at the front of the aircraft or conventional tail configuration at the rear of the aircraft. The decision to adopt a canard style design was dropped almost immediately. We wished to maintain the wing as the leading lifting surface so as to eliminate downwash on it and degrade the lifting capabilities of the aircraft. Given a conventional tail, there were three basic designs to be considered: a high crucifix tail, a mid-level crucifix tail, and a T-tail design. Increased stability and the prospect of decreasing the aircraft tip to tail length with dual vertical tails was weighted against the increasing complexity and rated aircraft of such a configuration. For the control surfaces on the main wing, separate flaps and ailerons or one combined "flaperon" covering the entire trailing edge of the wing were considered. The flaperon gave us the advantage of requiring fewer servos and therefore being cheaper in terms of rated aircraft cost and weight. However, it did present us with greater difficulty in maneuvering the plane. Separate flaps and ailerons provided greatly improved control over the aircraft but cost more in terms of rated aircraft cost because of additional servos and weight. The several permutations of these designs were compared using another relative comparison matrix. Refer to Table 3.2.

The three final concepts using this method consisted of a double and single stabilizer design, mid and high tail with all three concepts incorporating large rudder and both flaps and ailerons. A large

rudder, flaps and ailerons were included on all three designs because of the greater maneuverability that they entail. The mid and high tail concepts were retained because of their similarity in design. A closer numeric analysis of these two concepts was needed to determine the best design. The double and single stabilizers were also retained for further consideration.

	Weighting	High Tail, Single Vertical Stabilizer, Large Rudder, Flaperon	High Tail, Single Vertical Stabilizer, Large Rudder, Flaps / Aeilerons	High Tail, Single Vertical Stabilizer, Medium Rudder, Flaperon	High Tail, Single Vertical Stabilizer, Medium Rudder, Flaps / Aeilerons	High Tail, Single Vertical Stabilizer, Small Rudder, Flaperon	High Tail, Single Vertical Stabilizer, Small Rudder, Flaps / Aeilerons	High Tail, Single Vertical Stabilizer, Large Rudder, Flaperon	High Tail, Double Vertical Stabilizer, Large Rudder, Flaps / Aeilerons	High Tail, Double Vertical Stabilizer, Medium Rudder, Flaperon	High Tail, Double Vertical Stabilizer, Medium Rudder, Flaps / Aeilerons	High Tail, Double Vertical Stabilizer, Small Rudder, Flaperon	High Tail, Double Vertical Stabilizer, Small Rudder, Flaps / Aeilerons
Low Weight	15	4	3	1	4	2	4	2	5	5	2	3	6
Transportable Wing Span	4												
Durability / Survivability	12	4	5	2	1	4	4	4	5	4	5	2	1
Ease of Repair	5	3	3	5	4	1	4	4	4	5	3	2	1
Simplicity of Manufacture	6	4	3	2	1	4	5	4	4	1	2	5	4
Effectiveness Component Integration	10	3	3	5	1	4	5	4	4	4	5	5	4
Maximum Payload	14	4	4	5	3	2	4	5	4	3	2	5	4
Ease of Loading / Unloading	1												
Security of Payload During Flight	9												
Low Cost	2	2	5	4	3	2	3	4	4	3	5	2	4
Lightweight Powerplant	8												
Maximum Speed	13	2	4	5	4	3	4	1	4	4	4	3	2
Minimum Takeoff Distance	3												
Ease of Manuever	11	1	5	5	4	2	3	4	4	4	4	4	4
Standardized Controls	7	4	5	3	5	4	2	4	4	2	2	4	5
		302	376	345	287	268	369	325	407	352	315	344	340
			3						1				

	Weighting	Mid Tail, Single Vertical Stabilizer, Large Rudder, Flaperon	Mid Tail, Single Vertical Stabilizer, Large Rudder, Flaps / Aeilerons	Mid Tail, Single Vertical Stabilizer, Medium Rudder, Flaperon	Mid Tail, Single Vertical Stabilizer, Medium Rudder, Flaps / Aeilerons	Mid Tail, Single Vertical Stabilizer, Small Rudder, Flaperon	Mid Tail, Single Vertical Stabilizer, Small Rudder, Flaps / Aeilerons	Mid Tail, Single Vertical Stabilizer, Large Rudder, Flaperon	Mid Tail, Double Vertical Stabilizer, Large Rudder, Flaps / Aeilerons	Mid Tail, Double Vertical Stabilizer, Medium Rudder, Flaperon	Mid Tail, Double Vertical Stabilizer, Medium Rudder, Flaps / Aeilerons	Mid Tail, Double Vertical Stabilizer, Small Rudder, Flaperon	Mid Tail, Double Vertical Stabilizer, Small Rudder, Flaps / Aeilerons
Low Weight	15	4	3	1	4	2	4	2	5	5	2	3	6
Transportable Wing Span	4												
Durability / Survivability	12	4	5	2	1	4	4	4	5	4	5	2	1
Ease of Repair	5	3	3	5	4	1	4	4	4	5	3	2	1
Simplicity of Manufacture	6	4	3	2	1	4	5	4	4	1	2	5	4
Effectiveness Component Integration	10	3	3	5	1	4	5	4	4	4	5	5	4
Maximum Payload	14	4	4	5	3	2	4	5	4	3	2	5	4
Ease of Loading / Unloading	1												
Security of Payload During Flight	9												
Low Cost	2	2	5	4	3	2	3	4	4	3	5	2	4
Lightweight Powerplant	8												
Maximum Speed	13	2	4	5	4	3	4	1	4	4	4	3	2
Minimum Takeoff Distance	3												
Ease of Manuever	11	1	4	4	4	1	2	3	3	3	3	3	3
Standardized Controls	7	4	5	3	5	4	2	4	4	2	2	4	5
		302	365	334	287	257	358	314	396	341	304	333	329
									2				

	Weighting	Low Tail, Single Vertical Stabilizer, Large Rudder, Flaperon	T Tail, Single Vertical Stabilizer, Large Rudder, Flaps / Aeilerons	T Tail, Single Vertical Stabilizer, Medium Rudder, Flaperon	T Tail, Single Vertical Stabilizer, Medium Rudder, Flaps / Aeilerons	T Tail, Single Vertical Stabilizer, Small Rudder, Flaperon	T Tail, Single Vertical Stabilizer, Small Rudder, Flaps / Aeilerons	T Tail, Single Vertical Stabilizer, Large Rudder, Flaperon	T Tail, Double Vertical Stabilizer, Large Rudder, Flaps / Aeilerons	T Tail, Double Vertical Stabilizer, Medium Rudder, Flaperon	T Tail, Double Vertical Stabilizer, Medium Rudder, Flaps / Aeilerons	T Tail, Double Vertical Stabilizer, Small Rudder, Flaperon	T Tail, Double Vertical Stabilizer, Small Rudder, Flaps / Aeilerons
Low Weight	15	4	3	1	4	2	4	2	3	5	2	3	6
Transportable Wing Span	4												
Durability / Survivability	12	4	5	2	1	4	4	4	5	4	5	2	1
Ease of Repair	5	3	3	5	4	1	4	4	4	5	3	2	1
Simplicity of Manufacture	6	4	3	2	1	4	5	4	4	1	2	5	4
Effectiveness Component Integration	10	3	3	5	1	4	5	4	4	4	5	5	4
Maximum Payload	14	4	4	5	3	2	4	5	4	3	2	5	4
Ease of Loading / Unloading	1												
Security of Payload During Flight	9												
Low Cost	2	2	5	4	3	2	3	4	4	3	5	2	4
Lightweight Powerplant	8												
Maximum Speed	13	2	4	5	4	3	4	1	4	4	4	3	2
Minimum Takeoff Distance	3												
Ease of Manuever	11	1	3	3	3	1	2	2	2	2	2	2	2
Standardized Controls	7	4	5	3	5	4	2	4	4	2	2	4	5
		302	354	323	276	257	358	303	355	330	293	322	318

Table 3.2: Relative Comparison Matrix for the Control Surfaces

3.5.3 Fuselage. The different aspects of the airplane all meet in one location, the fuselage. The fuselage consisted of four major design considerations: body shape, tail connection, loading access, and payload capacity. For the concept phase, two basic body shapes were analyzed, a conventional cylinder or blended wing and body. The conventional cylinder would be easier to manufacture and entail less induced and interference drag than the blended body configuration. However, the blended wing and body design offered lower profile drag and potentially lifting characteristics to the fuselage itself. For these two body shapes there existed two types of tail connections: a tail that is blended with the fuselage or a tail that is separate from the fuselage and connected only by a tail boom. The tail boom design would be inherently weaker and demand more complex connections, however, it would reduce the weight of the aircraft for the same tip to tail length. Additionally the tail boom system shifts more weight further forward and increases the aircraft static margin. These considerations were analyzed in yet another decision matrix. Refer to Table 3.3.

Three closely competitive concepts were chosen. All of the concepts incorporated top loading and two utilized maximum payload capacities. Top loading results from the customer requirement for the need to quickly and easily load and off load the payload during the competition. The maximum payload was a customer requirement generally outweighed the customer requirement for low weight associated with a smaller fuselage but medium payload proved to be a competitive concept and was also retained for further analysis. All three selected designs incorporate a tail boom for the reasons stated above. Two of the designs selected were conventional body and one a blended body.

	Weighting	Conventional Body, Continous Fuselage [illegible]	Conventional Body, Continous Fuselage [illegible]	Conventional Body, Continous Fuselage [illegible]	Conventional Body, Continous Fuselage [illegible]	Conventional Body, Continous Fuselage [illegible]	Conventional Body, Continous Fuselage [illegible]	Conventional Body, Tail Boom, Top Loaded, Maximum Payload	Conventional Body, Tail Boom, Top Loaded, Medium Payload	Conventional Body, Tail Boom, Top Loaded, Low Payload	Conventional Body, Tail Boom, Side Loaded, Maximum Payload	Conventional Body, Tail Boom, Side Loaded, Medium Payload	Conventional Body, Tail Boom, Side Loaded, Low Payload	Blended Wing Body, Continous Fuselage [illegible]	Blended Wing Body, Continous Fuselage [illegible]	Blended Wing Body, Continous Fuselage [illegible]	Blended Wing Body, Continous Fuselage [illegible]	Blended Wing Body, Continous Fuselage [illegible]	Blended Wing Body, Continous Fuselage [illegible]	Blended Wing Body, Tail Boom, Top Loaded, Maximum Payload	Blended Wing Body, Tail Boom, Top Loaded, Medium Payload	Blended Wing Body, Tail Boom, Top Loaded, Low Payload	Blended Wing Body, Tail Boom, Side Loaded, Maximum Payload	Blended Wing Body, Tail Boom, Side Loaded, Medium Payload
Low Weight	15	2	3	4	2	3	4	2	3	4	2	3	4	2	3	4	2	3	4	2	3	4	2	3
Transportable Wing Span	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Durability / Survivability	12	4	4	4	4	4	4	5	5	5	5	5	5	4	4	4	4	4	4	5	5	5	5	5
Ease of Repair	5	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4	4
Simplicity of Manufacture	6	5	5	5	4	4	4	5	5	5	4	4	4	4	4	4	3	3	3	5	5	5	4	4
Effectiveness Component Integr	10	2	2	2	2	2	2	4	4	4	4	4	4	2	2	2	2	2	2	4	4	4	4	4
Maximum Payload	14	4	2	2	4	2	2	4	3	1	4	2	2	4	2	2	4	2	2	4	2	1	4	2
Ease of Loading / Unloading	1	3	3	3	2	2	2	3	3	3	2	2	2	3	3	3	2	2	2	3	3	3	2	2
Security of Payload During Flight	9	4	4	4	1	1	1	4	4	4	1	1	1	4	4	4	1	1	1	4	4	4	1	1
Low Cost	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Lightweight Powerplant	8	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Maximum Speed	13	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Minimum Takeoff Distance	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Ease of Manuever	11	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Standardized Controls	7	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
		243	230	245	209	196	211	275	276	263	241	228	243	237	224	239	203	190	205	275	262	263	241	228
		6	10	5	13	17	14	2	1	3	7	11	6	9	12	8	16	18	15	2	4	3	7	11

Table 3.3: Relative Comparison Matrix for the Fuselage

3.5.4 Propulsion. The competition limited engine selection to brushed electric motors from either the Graupner or Astroflite Company. A limit of a single 40-amp fuse was placed on each engine and led us to

include double engine concepts in consideration to account for such a restrictive current draw constraint. The downside of the dual engine system however would be increased weight and rated aircraft cost. Pusher as well as tractor propeller systems were considered. The tractor system would allow the propeller to see a clean, undisturbed airflow, however, would place the wing in its wake. The pusher system, on the other hand, would present a smoother airflow to the wing but subject the propeller its downwash and wake. Finally, the inclusion of gear reduction in the form of a gearbox was considered. This option offered increased engine mechanical output (torque) for the same electrical input, however increased weight and complexity to the design. These concepts were evaluated in Table 3.4.

Three engine concepts were selected for further analysis. All of the chosen concepts incorporated gear reduction and a tractor propeller. The added power benefits far outweighed their cost in weight for the gear reduction. We ultimately concluded that the front mounted or "tractor" propeller concept was easiest to manufacture and was best in terms of the aircraft's thrusting capabilities. Determination of the motor brand was unable to be definitively determined and the Astroflite vs. Graupner debate was reserved for further numerical analysis. Both a single and a twin engine concept were adopted for further analysis. We leaned towards a twin engine based on the decreased current draw, but retained the single engine concept on the chance that we might be able to achieve reasonably powered flight with it after further analysis.

	Weighting	Single Astroflite Engine, Tractor Propellor, Standard	Single, Astroflite Engine, Tractor Propellor, Gear	Single Astroflite Engine, Pusher Propellor, Standard	Single Astroflite Engine, Pusher Propellor, Gear	Single Graupner Engine, Tractor Propellor, Standard	Single Graupner Engine, Tractor Propellor, Gear	Single Graupner Engine, Pusher Propellor, Standard	Single Graupner Engine, Pusher Propellor, Gear	Twin Astroflite Engine, Tractor Propellor, Standard	Twin, Astroflite Engine, Tractor Propellor, Gear	Twin Astroflite Engine, Pusher Propellor, Standard	Twin Astroflite Engine, Pusher Propellor, Gear	Twin Graupner Engine, Tractor Propellor, Standard	Twin, Graupner Engine, Tractor Propellor, Gear	Twin Graupner Engine, Pusher Propellor, Standard	Twin Graupner Engine, Pusher Propellor, Gear
Low Weight	8	5	4	5	4	5	4	5	4	3	2	3	2	3	2	3	2
Transportable Wing Span	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Durability / Survivability	9	3	3	3	3	2	2	2	2	2	2	2	2	1	1	1	1
Ease of Repair	8	4	3	4	3	4	3	4	3	4	3	4	3	4	3	4	3
Simplicity of Manufacture	8	4	3	4	3	4	3	4	3	3	2	3	1	3	2	3	2
Effectiveness Component Integrat	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Maximum Payload	10	1	3	1	2	1	2	1	2	3	5	3	4	3	4	3	4
Ease of Loading / Unloading	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Security of Payload During Flight	9	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Low Cost	8	3	3	3	3	3	3	3	3	2	2	2	2	2	2	2	2
Maximum Speed	10	2	3	1	2	2	3	1	2	3	5	3	4	3	4	3	4
Minimum Takeoff Distance	8	1	2	1	2	1	2	1	2	3	5	3	4	3	4	3	4
Ease of Manuever	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Standardized Controls	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
		193	207	183	187	184	188	174	178	198	230	198	194	189	193	189	193
		5	2	70	8	9	7	12	11	4	1	3	4	6	5	6	5

Table 3.4: Relative Comparison Matrix for the Propulsion System

3.5.6 Landing Gear. The fifth and final aspect of our aircraft design is the most often overlooked portion of the design. Without proper landing gear an aircraft cannot take off or land successfully. Two overall design concepts were judged to be reasonable, a tail dragger and tricycle style landing gear. The tricycle landing gear provides several advantages over a tail dragger design. Tricycle landing gear is more stable because the center of gravity is located in front of the main landing gear, between the two pieces of the landing gear. Tricycle landing gear also has the advantage of bringing the wings to a negative angle of attack after touchdown, keeping the plane on the ground. However, the tail dragger puts the wing at a positive angle of attack during taxi and reduces the takeoff distance. Also considered, was fixed vs. retractable landing gear. Fixed gear initially seemed more advantageous than retractable landing gear because of the complications in manufacture of a retractable system. However, retractable gear offered decreased drag profile for the aircraft when in flight. Finally, a dampened and undamped concept were analyzed. Fluid dampening and spring dampening were examined as means of increasing the survivability of the aircraft. However, their complexity was a tradeoff to be considered. Finally, a steerable nose wheel design vs. a fixed design was analyzed. Reference Table 3.5.

From this analysis we determined the final landing gear concept to be a tricycle system fixed to the fuselage. The spring damped and fixed frontal strut concepts were reserved for further analysis in the preliminary design phase.

	Weighting	Tricycle, Fixed, Undamped (Front Strut)	Tricycle, Fixed, Spring Damped (Frontal Strut)	Tricycle, Fixed, Fluid Damped (Frontal Strut)	Tricycle, Fixed, Spring Damped, Steerable Nose	Tricycle, Retractable, Undamped (Frontal Strut)	Tricycle, Retractable, Spring Damped (Frontal Strut)	Tricycle, Retractable, Fluid Damped (Frontal Strut)	Tail Dragger, Fixed, Undamped (Frontal Strut)	Tail Dragger, Fixed, Spring Damped (Frontal Strut)	Tail Dragger, Fixed, Fluid Damped (Frontal Strut)	Tail Dragger, Retractable, Undamped (Frontal Strut)	Tail Dragger, Retractable, Spring Damped (Frontal Strut)
Low Weight	15	5	3	2	3	3	2	1	3	2	1	2	1
Transportable Wing Span	4												
Durability / Survivability	12	1	3	5	4	1	2	4	2	3	4	1	2
Ease of Repair	5	5	4	2	3	4	3	1	4	3	2	2	1
Simplicity of Manufacture	6	5	3	1	2	3	2	1	3	2	1	2	1
Effectiveness Component Integrat	10	5	4	3	4	3	2	1	4	3	2	3	2
Maximum Payload	14	3	4	5	5	4	3	2	4	3	2	3	2
Ease of Loading / Unloading	1	0	0	0	0	0	0	0	0	0	0	0	0
Security of Payload During Flight	9	0	0	0	0	0	0	0	0	0	0	0	0
Low Cost	2	5	4	1	3	3	2	1	5	4	3	3	2
Lightweight Powerplant	8	0	0	0	0	0	0	0	0	0	0	0	0
Maximum Speed	13	3	4	5	4	2	3	4	1	2	3	4	1
Minimum Takeoff Distance	3	0	0	0	0	0	0	0	0	0	0	0	0
Ease of Manuever	11	3	3	3	5	3	3	3	2	2	2	1	1
Standardized Controls	7	0	0	0	0	0	0	0	0	0	0	0	0
		316	**308**	**306**	**343**	**246**	**219**	**199**	**248**	**221**	**194**	**205**	**126**
		2	**3**	**4**	**1**	**6**	**7**	**10**	**5**	**8**	**11**	**9**	**12**

Table 3.5: Relative Comparison Matrix for the Landing Gear

3.6 Results

The result of the conceptual design phase is several concepts under each of the five major sub-team categories which are viable and will be carried forward into the preliminary design phase. In this

phase they will undergo numerical and graphical analysis in order to select the final and best combination of the competing concepts from this phase of the design. A summary of the selected concepts follows.

Concepts Selcted

Subteam	*Concept*
Wing	High Wing, Straight, Low Dihedral
	Medium Wing, Straight, Low Dihedral
Control Surfaces	High Tail, Single Vertical Stabilizer, Large Rudder, Flaps / Aeilerons
	High Tail, Double Vertical Stabilizer, Large Rudder, Flaps / Aeilerons
	Mid Tail, Double Vertical Stabilizer, Large Rudder, Flaps / Aeilerons
Fuselage	**Conventional Body, Tail Boom, Top Loaded, Maximum Payload**
	Conventional Body, Tail Boom, Top Loaded, Medium Payload
	Blended Wing Body, Tail Boom, Top Loaded, Maximum Payload
Propulsion	**Single, Astroflite Engine, Tractor Propellor,Gear Reduction**
	Twin, Astroflite Engine, Tractor Propellor, Gear Reduction
	Twin, Graupner Engine, Tractor Propellor, Gear Reduction
Landing Gear	**Tricycle, Fixed, Undamped,(Front Strut)**
	Tricycle, Fixed, Spring Damped, Steerable Nose wheel

Table 3.6: Concepts remaining for further analysis after the conceptual design phase

4.1 Introduction

The goal of the preliminary design phase was to first narrow the concepts remaining from the previous design phase to a single idea per component. Analytical methods to include detailed graphical and numerical studies were then used to select component materials, determine final shapes, establish preliminary dimensions, and further specify final configurations.

4.2 Figures of Merit

The first step in conducting the preliminary design analysis was to identify the figures of merit we would use to assess each component and configuration parameter of interest. The figures of merit were chosen based on the most important aircraft mission specifications in order to ensure that the final design concept would yield the best overall aircraft design.

4.2.1 Ease of Manufacture. Perhaps one of the most important considerations in assessing each design possibility is its ability to be constructed given the resources and abilities of the design team members. Obviously more simplistic concepts would lead to designs that could be built with less time and skill and became an important distinguishing criteria in evaluating the remaining design concepts.

4.2.2 Lift to Drag Ratio. The lift to drag ratio of an aircraft constitutes the most basic measure of aircraft efficiency. The lift produced by the aircraft must be equal to its gross weight in order to sustain flight, however, at the same time a minimum amount of drag is desired in order to minimize power consumption and maximize aircraft speed.

4.2.3 Thrust to Weight Ratio. The thrust to weight ratio is measure of the aircraft's propulsive capabilities as compared to its gross weight. The ability of the engines to generate a forward force should be maximized while at the same time minimizing the overall weight that must be propelled. This ratio is an important consideration in a variety of aircraft performance calculations.

4.2.4 Strength to Weight Ratio. Vehicles in flight are subject to a variety of loads and stresses. While specifically quantifying each of these stresses would prove to be near impossible, it is crucial for the design team to feel confident that the overall structure of the aircraft possess adequate strength to withstand their cumulative effects. Additionally, the loads and stresses associated with landing must also be carefully considered in order to ensure safe aircraft retrieval. These factors, however, must be weighed against the desire maintain an overall design that is as light as possible to increase overall performance.

4.2.5 Competition Scoring to Rated Aircraft Cost Ratio. This figure of merit is specific to the contest rules and mission profile of the aircraft. The aircraft's ability to score highly in the competition is desired and a function of its ability to complete the mission profile quickly, safely, and repetitively. However, at the same time, care must be taken to ensure that the rated aircraft cost, a measure of the design's complexity, be minimized to achieve the highest possible overall flight scores.

4.3 Design Parameters Investigated

The design team investigated a variety of parameters in order to arrive at the best possible aircraft configuration. They are organized by their relationship to each sub-project group as can be seen below. Within each project component section, any general concept choices that remained after initial evaluation in the conceptual design section are introduced and further explained. Next, new parameters more specific to each component are introduced and tradeoffs associated with each are discussed. Finally, material and dimensional considerations are introduced. The majority of the residual ideas from the initial conceptual design phase as well as the material selection for each component are analyzed by using a second run of the evaluation matrix and the new figures of merit discussed above. The more specific parameters for each component are more complex and require analytical analyses to further fix the final configurations. Finally, dimensional considerations for each component are studied and specified based on existing constraints. However, dimensional considerations relating to aircraft performance, while introduced in this preliminary design section, are analyzed further in the detail design section.

4.3.1 Wing Parameters. Two wing concepts requiring further analysis were derived from the conceptual analysis section. They were a high wing with a uniform chord and high dihedral and a medium wing with a uniform chord and high dihedral. Therefore, the high wing vs. medium wing must be further studied to narrow our consideration to a final concept. The high wing lends to the aircraft a greater degree of stability, however the medium wing, if passed through the center of the fuselage, would simplify the wing to fuselage connection and could potentially add strength to this particular attachment. Further considerations regarding the wing are described in depth below.

4.3.1(a) Wing Airfoil. The airfoil, or cross sectional shape of the wing is an important characteristic that determines the lift curve slope (lift of the wing as a function of angle of attack), zero lift angle of attack, and stall angle of attack. A wing with an airfoil capable of providing adequate lift at a low angle of attack can reduce aircraft profile drag, however, increased airfoil camber can result in an increase of wing downwash or induced drag.

4.3.1(b) Wing Planform. The wing planform area is a function of its span and mean chord length. The aspect ratio, the wing span squared divided by the planform area is an important characteristic in many aircraft performance considerations. Most notably, higher aspect ratios (longer, more narrow wings) reduce induced drag. Also, a higher wing loading (shorter, more narrow wings) can increase speed and maneuverability but also require more structural strength to withstand increased pressure on the wing surface.

4.3.1(c) Dihedral Angle. The upward sloping angle of the wing lends stability to the aircraft as it orients the lift vectors in a manner that counteracts rolling disturbances along the longitudinal axis. The greater the dihedral angle, the greater the aircraft roll stiffness. However, at the same time, increasing the dihedral angle has a negative effect on the efficiency of the wing as it slightly reduces the free stream velocity the wing's leading edge encounters.

4.3.1(d) Wing Dimensions. The dimensions of the wing will be dictated mostly by the wing planform area which will be optimized using analytical techniques later is this section of the report. However, two parameters dependent on this planform area, the wing span and the wing chord must be defined. A larger span and shorter chord would result in a higher wing aspect ratio which in turn decreases induced drag on the wing. Though this high aspect ratio is a favorable characteristic from a performance standpoint, it comes at the expense of decreased structural integrity.

4.3.1(e) Wing Materials. The wing lifts the entire weight of the aircraft and the material chosen for its structure must be strong enough to support this distributed load across its surface. In addition, the wing must be stiff enough to prevent any flapping, twisting, or bending effects that have huge aerodynamic consequences on its performance. Materials and construction methods considered include an internal balsa structure, a foam core with a carbon fiber coating, an aluminum reinforcement spar, and a carbon fiber reinforcement spar.

4.3.2 Dynamic Control and Stability Parameters. In the conceptual design, it was decided that the final aircraft configuration should contain a conventional tail with a large rudder and a single wing with ailerons and flaps. Considerations that remain after this initial concept evaluation include a high placed tail vs. a middle placed tail and a single vertical stabilizer vs. double vertical stabilizers.

In studying the high vs. the middle placement of the tail, it is noted that the efficiency of the horizontal tail as a positive or negative lifting surface is degraded by wake from the wing. This is the main drawback of having the stabilizing surface in the rear of the aircraft. This phenomenon, however, can be overcome by offsetting the placement of the tail relative to the wing in the vertical plane. However, difficulties in constructing such an offset tail are also important considerations and constitute a tradeoff to a more efficient high placed tail. The dual vertical stabilizers offer increased yaw stability with decreased planform area per vertical surface. This is important since excessive vertical surface area can lead to a condition known as "tail wagging" especially in low speed, R/C aircraft. However, the dual system is more difficult to construct and increases the rated aircraft cost of the final design.

4.3.2(a) Control Surface Dimensions. The size significance associated with the stabilizing aircraft parts is illustrated in the vertical stabilizer discussion above. The same concepts discussed above relate the planform and moment arm of the horizontal stabilizer to pitch stability as well as the size and lateral distance from the CG of the ailerons to roll control. Therefore, size and location of all controlling and stabilizing surfaces are important considerations relating to the aircraft performance.

4.3.2(b) Control Surface Materials. The control surfaces as well as the stabilizing parts are subject to strong wind forces, especially those deployed into the free stream (i.e. elevator, rudders, ailerons). Therefore, material selection of these parts is important to ensure they are strong enough to survive flight conditions and stiff enough to prevent their flutter a condition that can seriously reduce their efficiency. Materials and construction methods considered include a balsa internal structure, a foam structure, and a foam structure with a carbon fiber coating.

4.3.3 Fuselage Parameters. Concepts remaining after the initial fuselage concept evaluation include a lifting body vs. a conventional body, a tail boom configuration vs. a continuos fuselage, and a maximum vs. a medium sized payload capacity. The lifting body fuselage contributes to the lift of the wing but is difficult to manufacture and introduces increased induced drag as opposed to the conventional body. The tail boom configuration forces more of the payload weight to be distributed towards the front of the aircraft and results in a center of gravity further forward and a greater aircraft static margin. However, the tail boom introduces complications in construction and the possibility of increased interference drag. Finally, the primary issues with the payload capacity involve the competition rules since a greater payload results in a higher flight score. The increased weight and fuselage size demands associated with a maximum payload are a trade-off to consider.

4.3.3(a) Fuselage Shape and Dimensions. As with the wing, the cross sectional shape of the fuselage is an important consideration for a lifting body fuselage. If a conventional body fuselage is selected, the cross section becomes less important, however, the desire for a low area profile and smooth contoured shape still exists to reduce boundary layer separation over the structure. The cross section of a potential tail-boom is also important to consider in order to minimize aerodynamic drag over its surface. The dimensions of the fuselage are dependent on the chosen payload capacity and its configuration within the aircraft.

4.3.3(b) Fuselage Materials. The fuselage must be strong enough to support the full weight of the batteries and the payload. It also must interface to several other parts of the aircraft to include the engines, tail, wing, and landing gear and therefore must provide stiff, strong connection parts. It is also important to minimize the weight of the fuselage structure in order to decrease the gross aircraft weight. Materials considered for the fuselage include an internal balsa structure, an aluminum airframe with a single or double metal spine, and a balsa structure with carbon fiber or fiber glass coating.

4.3.4 Propulsion Parameters. Parameters remaining from the initial concept evaluation of the propulsion system include the single vs. twin engine configurations and the Graupner vs. the Astroflite engine systems. The twin engine configuration offers increased thrust for the same amount of current and voltage draw which would translate to higher aircraft endurance. However, the extra weight associated with the twin engine and the complications involved in aligning their respective thrust vectors is a downfall of this configuration. The choice of engine brand is dependent upon the efficiency of each at different power settings. Both of these parameters proved too complex to analyze in a concept evaluation matrix and are therefore studied in detail analytically.

4.3.4(a) Engine Mounting Device Dimensions. The dimensions of the engine mounting device are quite simple to specify. They are directly dependent upon the size of the engines and interface to the rest of the aircraft. Given these two constraints, the mounting devices should be as small as possible to avoid profile drag, however, larger enough to still maintain structural soundness.

4.3.4(b) Engine Mounting Device Material. The device to which the engine or engines are to be mounted is crucial for a tractor propeller system, which was decided upon in the conceptual section. The

entire aircraft is in effect being pulled at these points. This device demands a strong, rigid material, however, as always, minimal weight is desired. Materials considered for this portion of the propulsive system include aluminum, steel, and titanium. Titanium would present the highest strength to weight ratio but the difficulties in welding and working this particular metal could eliminate it from consideration.

4.3.5 Landing Gear Parameters. In the conceptual design section, it was decided that our design would incorporate and tricycle main landing gear system. The issue required further analysis was whether to use a fixed, undamped strut or a spring damped strut for the frontal, nose gear. The fixed strut would be a simpler and less complex system but would translate a great deal of impact force to the fuselage-landing gear interface.

4.3.5(a) Landing Gear Shape. The shape of the main and nose gear is an important feature as it ultimately dictates the total gear weight and the distribution of the landing load. Additionally, a low area cross section is desired to reduce air drag past the system during flight. Cut outs and notches in the gear reduce dead weight to the aircraft but also create stress concentrations at which the system could be damaged during rough landings.

4.3.5(b) Landing Gear Dimensions. The dimensions of the landing gear must be such that they provide adequate clearance between the ground and the propellers which were assumed to extend below any other aircraft component. Additionally, the landing gear was required to provide a wheel base sufficient to prevent the aircraft from rolling or tipping during taxi. However, at the same time, minimal spatial dimensions given these constraints are desired to reduce weight and profile drag on the landing gear.

4.3.5(c) Landing Gear Material. The material for the landing gear is extremely important as it must possess a variety of features. The material must be strong and resilient enough to absorb the impact load of landing without yielding. Additionally, it is important for the main gear to flex slightly in the elastic deformation range so that more impact energy is dissipated rather than being transferred to the fuselage. This flexural distance must be limited, however, to ensure that at all times during the landing sequence propeller clearance with the ground is maintained. A propeller striking the ground could cause the aircraft to tumble or tip and thus result in a catastrophic aircraft failure. Materials considered for the landing gear include Aluminum, Carbon Fiber, and Steel.

4.4 Methods Used to Study Design Parameters. As was previously discussed, the majority of the conceptual decisions remaining from the initial analysis could be determined by directly examining the choices relative to each other using weighted values of our new figures of merit as criteria. Additionally, the material choices for each component could be analyzed using this method. Table 4.1 illustrates this relative comparison in the form of a chart. The remainder of the parameters for each aircraft component were studied using the analytical techniques summarized in subsequent report sections.

			Ease of Manufacture	Lift / Drag Ratio	Thrust / Weight Ratio	Strength / Weight Ratio	Flight Score / RAC Ratio	Total
		Weighting Factor	5	2	2	3	4	
Wing	**Final Concept Selection**	High wing	-2	0	0	-1	0	-13
		Mid wing	1	0	0	1	0	8
	Material Selection	Internal balsa structure	-3	-1	0	0	0	-17
		Foam core, carbon fiber coating	3	1	1	1	-1	18
		Foam core, carbon fiber coating, aluminum reinforcement spar	2	1	2	2	-1	18
		Foam core, carbon fiber coating, carbon fiber reinforcement spar	2	1	3	3	-1	23
Dynamic Control and Stability	**Final Concept Selection**	High tail	1	1	1	0	0	9
		Mid tail	1	-2	-2	0	0	-3
		Single vertical stabilizer	0	1	0	0	1	6
		Double vertical stabilizer	[illegible]	0	0	0	[illegible]	7
	Material Selection	Internal balsa structure	0	0	0	2	0	6
		Foam core	1	0	0	0	0	5
		Foam core, carbon fiber coating	0	0	0	1	0	3
Fuselage	**Final Concept Selection**	Lifting body, tail booms	-2	1	0	0	1	-4
		Lifting body, continous fuselage	-1	2	0	-1	0	-4
		Conventional body, tail booms	1	1	0	0	1	11
		Conventional body, continous fuselage	2	1	0	-1	0	9
	Material Selection	Balsa structure	-2	1	1	-1	0	-9
		Balsa structure, carbon fiber coating	-3	1	1	0	0	-11
		Balsa structure, fiber glass coating	-3	2	2	-1	0	-10
		Aluminum airframe, single spine	3	1	1	1	0	22
		Aluminum airframe, double spine	2	1	1	3	0	23
Propulsion	**Material Selection (engine mount)**	Aluminum	1	0	1	2	0	13
		Steel	1	0	0	2	0	11
		Titanium	-3	0	2	3	0	-2
Landing Gear	**Final Concept Selection**	Fixed frontal strut	2	0	0	1	0	13
		[illegible]	1	0	0	3	0	14
	Material Selection	Aluminum	2	0	1	2	0	18
		Carbon fiber	-3	0	2	3	0	-2
		Steel	2	0	0	0	0	10

Table 4.1: Relative comparison of some remaining possible component configurations and possible material choices. The most positive and therefore preferred choices based on the weighted figures of merit are highlighted in green.

4.4.1 Wing Analytical Methods. In designing for the wing parameters, a iterative routine was utilized. First, an in-depth airfoil selection process was conducted to find the expected wing coefficients of lift and drag. Next, the aircraft weight and cruise velocity were estimated in order to design for the wing planform area. Using this planform, the chord and span of the wing were calculated in the performance analysis section of this report (detail design phase) using an assumed value for the aspect ratio. Finally, the aircraft drag polar and velocity required for maximum efficiency were calculated. The process was then iterated until the wing specifications yielded a cruise velocity equal to the maximum efficiency velocity.

4.4.1(a) Airfoil Analysis. To select the optimum airfoil for our aircraft design, experimental data on low-speed airfoils was collected and studied. Specifically, the airfoils' lift curves (relationship between the angle of attack and coefficient of lift) and drag polar plots (relationship between the coefficient of lift and coefficient of drag) were examined. An airfoil with a high coefficient of lift for a given angle of attack was desired in order to maximize the lift to drag ratio figure of merit. At the same time however, care was taken to ensure that the drag coefficient was relatively low over the aircraft's expected range of angle of attack. Furthermore, airfoils with more negative zero lift angle of attack and higher stall angle's were preferred to ensure the wing would produce lift over a greater range of angles lending to it a greater degree of safety. Finally, the stall characteristics were considered as a "softer" airfoil stall or more gradual drag bucket were preferred in order to eliminate the chances of sudden, radical loss of wing lift. Ten airfoils were initially chosen based on these characteristics. Figures 4.1 and 4.2 are the lift curves and drag polar plots respectively.

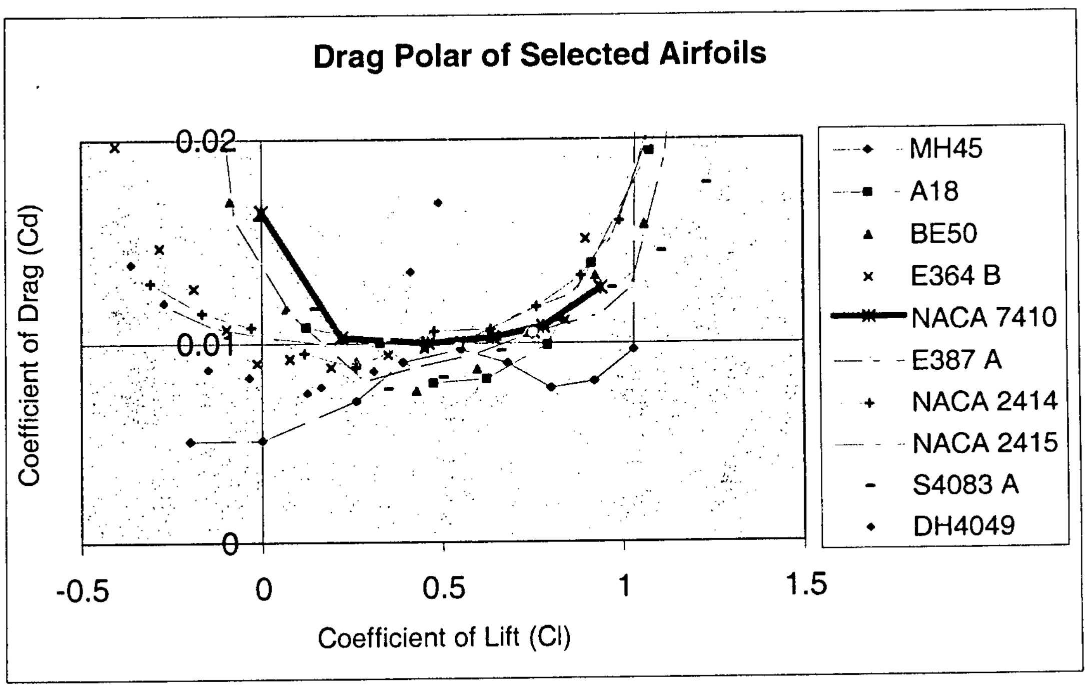


Figure 4.1: Drag polar plots for various airfoils. The NACA 7410, the final airfoil choice for the wing, is depicted by the bold, orange line. Although the drag values are somewhat high, the drag bucket is shallow and at a minimum over the desired values for the lift coefficient.

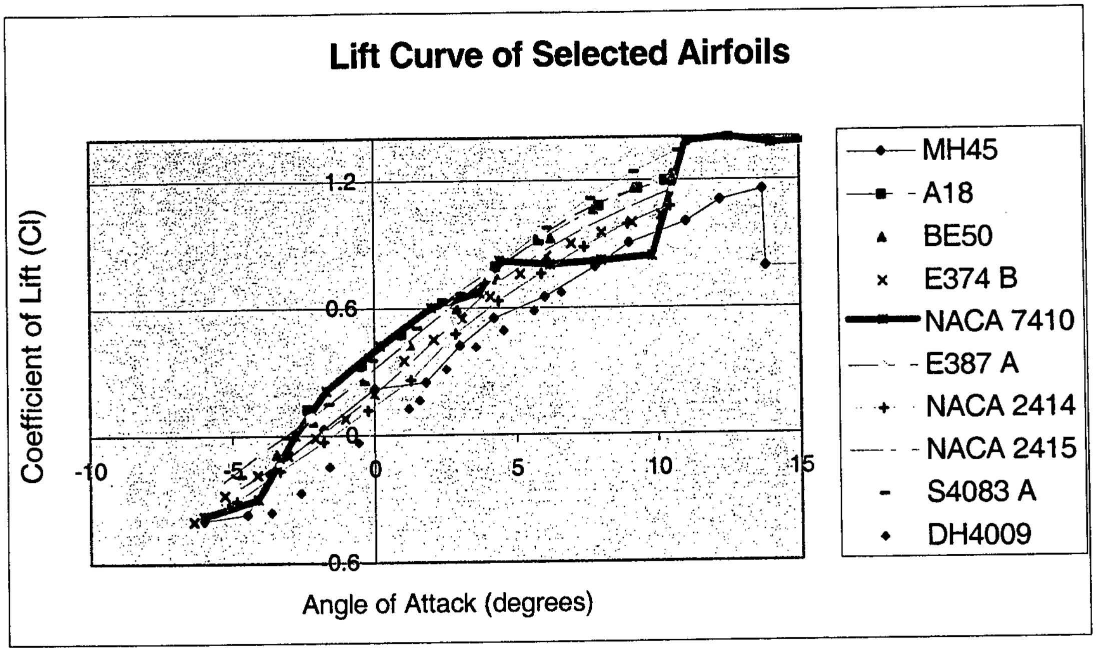


Figure 4.2: Lift curves for various airfoils. Note the high stall angle of attack and the gentle stall characteristics.

Airfoil	Zero Lift Angle of Attack	Stall Angle of Attack	Maximum Coefficient of Lift (CL)
MH45	-2.1	13.7	1.10
A18	-3.5	10.6	1.19
BE50	-2.7	10.7	1.22
E374 B	-2.0	10.2	1.03
NACA 7410	-2.8	11.0	1.38
E387 A	-3.2	10.0	1.18
NACA 2414	-1.5	10.7	1.08
NACA 2415	-2.0	10.7	1.15
S4083 A	-3.0	10.8	1.33
DH4009	-0.4	7.0	0.67

Table 4.2: Summary of relevant airfoil data. The ultimate airfoil choice, the NACA 7410, is highlighted in green.

4.4.1(b) Planform Analysis. The wing planform was calculated by first estimating values for the aircraft gross weight and cruise velocity. Ultimately, the value of the velocity was refined in the aircraft performance section of this report (detail design phase) and the process was iterated until the wing specifications were optimum. Throughout this process, the coefficient of lift for the finite wing was

$$L = \frac{1}{2} \cdot \rho \cdot V^2 \cdot S \cdot C_L$$

$$W = L$$

$$S = \frac{2 \cdot W}{\rho \cdot V^2 \cdot C_L}$$

W: Aircraft Weight
L: Required Lift
ρ: Freestream Density
V: Cruise Velocity
S: Wing Planform Area
C_L: Wing Coefficient of Lift

assumed to be equal to that of the airfoil. Furthermore, standard sea level conditions as well as straight and level flight were assumed. Therefore, the following equation was used to calculate the necessary wing planform area. The optimum result was 10 ft^2.

4.4.1(c) Dihedral Analysis. In determining the numerical value of the dihedral angle, the spiral stability of the aircraft in a turn was considered (this is an appropriate condition to consider as the contest rules call for a 360 degree aircraft turn). Additionally, a parameter known as the equivalent dihedral angle (EDA) of a V-shaped wing was introduced. Since an aircraft wing is usually divided into panels with the dihedral beginning on the outboard sections, a method for equivalent dihedral angle (the dihedral angle of a perfectly V-shaped wing with identical roll characteristics) must be used. The EDA for each panel is the actual dihedral angle of that panel times the difference between the rolling moment fractions at either end of the panel. The rolling moment fraction generated at the dihedral breaks can be found given the wing semi-span station (non-dimensionalized distance from wing root) and Figure 4.3. The total EDA for the wing is the sum of the EDA for each of the panels (summed on only one side assuming the wing is symmetrical).

With the equivalent dihedral angle calculated, the aircraft could then be designed to be neutrally, spirally stable. That is, the dihedral angle was designed so that it along with the effects of roll-yaw coupling would negate the tendency of the aircraft to roll inboard during a bank. This analysis required the assumption of a parameter dealing with the aircraft handling characteristics (distance from the quarter chord point of the mean aerodynamic chord to the vertical stabilizer). This value would later be optimized in the handling characteristics section of the detail design phase. The calculation of prospective dihedral configurations and the final one selected can be viewed in Table 4.3.

$$EDA \cdot \frac{\left(\frac{L_V}{b}\right)}{C_L} = 5.0 \text{ (Approximately)}$$

$$EDA = \frac{5.0 \cdot C_L \cdot b}{L_V}$$

EDA: Equivalent Dihedral Angle
L_V: Distance from 1/4 chord to Vertical Stabilizer
b: Wing Span
C_L: Wing Coefficient of Lift

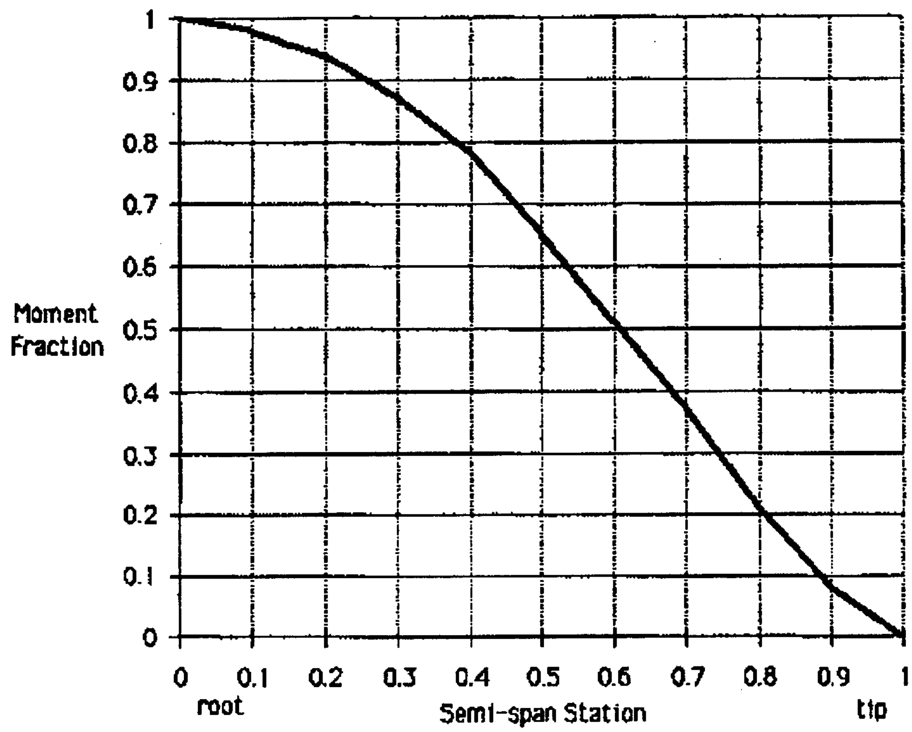


Figure 4.3: Rolling moment fraction for each wing semi-span station. This figure is used to determine the equivalent dihedral angle of a wing. Taken from Blaine Beron-Rawdon's website, *Calculation of an Equivalent Dihedral Angle.*

Coefficient of Lift		0.65		
Distance from 1/4 Chord to Vertical Stabilizer		54 (in)		
Wing Span		120 (in)		
EDA Required	9.444444444			
Length of Inboard Panel (ft)	Semi-Span Statio of Dihedral Break	Rolling Moment Fraction	Outboard Dihedral Angle	EDA
0.50	0.10	0.98	5.0	4.9
1.50	0.30	0.87	5.0	4.35
2.50	0.50	0.65	5.0	3.25
3.50	0.70	0.37	5.0	1.85
0.50	0.10	0.98	7.0	6.86
1.50	0.30	0.87	7.0	6.09
2.50	0.50	0.65	7.0	4.55
3.50	0.70	0.37	7.0	2.59
0.50	0.10	0.98	9.0	8.82
1.50	0.30	0.87	9.0	7.83
2.50	0.50	0.65	9.0	5.85
3.50	0.70	0.37	9.0	3.33
0.50	0.10	0.98	11.0	10.78
[redacted]				
2.50	0.50	0.65	11.0	7.15
3.50	0.70	0.37	11.0	4.07

Table 4.3: Calculation of the equivalent dihedral angle required for neutral spiral stability and equivalent dihedral angles for various dihedral configurations. The best dihedral configuration is highlighted in green.

4.4.2 Fuselage Analytical Methods

4.4.2(a) Payload Analysis. In evaluating the choice between carrying a medium sized and maximum sized payload, we turned to the 2001 AIAA DBF competition results. The figure of merit we sought to maximize was the competition scoring to rated aircraft cost ratio as we based our analysis on the flight scores and payload of the previous year's aircraft. Table 4.4 illustrates the first seventeen finishers in the 2001 competition and their respective flight scores. The 2001 competition allowed a maximum payload of 100 tennis balls. Additionally, aircraft were allowed to complete multiple sorties fully loaded. Therefore, we assumed that any aircraft with a tennis ball total that was a multiple of 100 must have flown their sorties with the maximum payload. As can be clearly seen, the first six out of seven finishers (also the top four finishers) all had aircraft designed to carry the maximum 100 tennis balls. Therefore, assuming that this trend would once again be prevalent in the 2002 competition, we concluded that maximum payload capacity would be necessary to keep pace with the top finishers in the competition.

Entry Name	Paper Score	RAC	Weight (min 5#)	Balls (min 10)	Score	Weight	Balls	Score	Weight	Balls	Score	Flight Score	Total Score
OSU Orange/shamu	90.03	4.70	46.90	200.00	86.90	28.80	300.00	88.80	43.95	200.00	83.95	259.65	4976.86
Steel & Balls	83.25	4.46	26.90	300.00	86.90	26.90	300.00	86.90	26.90	300.00	86.90	260.70	4864.68
OSU Black	91.00	4.75	36.90	200.00	76.90	36.90	200.00	76.90	27.90	200.00	67.90	221.70	4243.73
TLAR II	84.48	5.96	48.90	200.00	88.90	34.20	300.00	94.20	34.20	300.00	94.20	277.30	3927.10
A	92.77	6.27	32.00	160.00	64.00	31.80	160.00	63.80	35.00	160.00	67.00	194.80	2880.90
mU2	81.43	6.81	13.65	194.00	52.45	21.50	200.00	61.50	13.65	194.00	52.45	166.40	1988.80
University of Illinois	83.77	6.61	27.80	100.00	47.80	27.90	100.00	47.90	27.88	100.00	47.88	143.58	1819.22
Scorpion	84.33	5.33	13.50	200.00	53.50	13.50	200.00	53.50			0.00	107.00	1691.31
Carbon Goose	78.70	5.71	11.20	80.00	27.20	5.60	160.00	37.60	9.50	140.00	37.50	102.30	1411.02
Flying Centurions	79.35	7.07	10.00	66.00	23.20	6.00	98.00	25.60	8.95	150.00	38.95	87.75	984.61
Almost Heaven	78.43	6.56	6.15	108.00	27.75	10.00	27.00	15.40	6.50	102.00	26.90	70.05	837.69
Anatolian-Craft	82.90	9.50	8.25	66.00	21.45	13.35	100.00	33.35	11.65	100.00	31.65	86.45	754.07
Aircat 2001 CR2	56.83	8.03	20.70	54.00	31.50	25.80	72.00	40.20	15.55	72.00	29.95	101.65	719.26
Black Knight	79.58	8.35	7.10	70.00	21.10	7.10	100.00	27.10	7.10	90.00	25.10	73.30	698.54
TTF Technologies	74.43	6.21		30.00	6.00	10.00	175.00	45.00			0.00	51.00	611.49
"Georgia Tech" -or-	78.00	5.57	9.80	160.00	41.80			0.00			0.00	41.80	585.77
Knightmare	77.78	4.01		14.00	2.80	7.50	14.00	10.30		14.00	2.80	15.90	308.42
Double Trouble	67.43	6.28		30.00	6.00		54.00	10.80		54.00	10.80	27.60	296.52
Knight Hawk	63.13	7.62		10.00	2.00	5.00	45.00	14.00			0.00	16.00	132.48
Full Monty	54.28	5.79		30.00	6.00			0.00			0.00	6.00	56.29
Lothworks	72.88	6.57		12.00	2.40			0.00			0.00	2.40	26.61
Undecided	54.95	9.38		16.00	3.20			0.00			0.00	3.20	18.75
Absolute Toque	92.05	7.30			0.00			0.00			0.00	0.10	1.26
The GPA killer	76.59	7.04			0.00			0.00			0.00	0.10	1.09
Hokie Bird 5	63.88	6.18			0.00			0.00			0.00	0.10	1.03
NAVY	68.20	7.30			0.00			0.00			0.00	0.10	0.93
The Bosphorus Blue	80.25	100.00			0.00			0.00			0.00	0.10	0.08
No Excuse	76.40	100.00			0.00			0.00			0.00	0.10	0.08
Der Gabelschwanz Teufel	76.20	100.00			0.00			0.00			0.00	0.10	0.08
CityHawk	55.05	100.00			0.00			0.00			0.00	0.10	0.06

Table 4.4: Results of the 2001 AIAA DBF competition. Configurations denoting maximum allowable payload under flight scores are listed in red text.

4.4.3(b) Analysis of Fuselage Shape and Dimensions. Spatial configuration and dimensions of the fuselage can now be specified and are dictated by the payload capacity of the aircraft. Up to this point, we had established that the fuselage would consist of a conventional body with tail booms to move the center of gravity forward thus increasing the aircraft static margin. Furthermore, we had decided on an airframe consisting of dual aluminum spines. Thinking ahead to the engine mount requirement for a strong, rigid interface to the fuselage and realizing the necessity of twin aircraft engines (to be explained later), we decided to incorporate the engine mounts to the fuselage's aluminum spines. This meant mounting the aircraft's propellers on both sides of the fuselage requiring the fuselage to be wide enough to provide clearance between the two spinning blades. Having established the need for 18 inch propellers (to be explained later), we designed to fuselage to be 16 inches in width to give 2 inches of propeller clearance (when the engines were mounted on their sides giving a extra 2 inches of room on either side of the fuselage). Furthermore, this allowed for the loading of 4, 3.5 inch diameter softballs abreast. Requiring an additional 5 rows of 4 softballs to maximize the payload, we determined that the body of the fuselage should be 33 inches long to allow for a total of 6 rows of 3.5 inch diameter softballs plus a middle placed wing with a chord of 12 inches (to be explained later). The total aircraft tip to tail length, which was optimized in the handling characteristics portion of the detail design phase, was determined to be 70 inches. This required 37 inch long tail booms that would angle upwards to account for the 6 inch offset of the high placed tail from the plane of the wing. (Reference 5.8.2(a) in the drawing package).

4.4.3 Propulsion Analytical Methods

4.4.3(a) Analysis of Engine Brands. We were faced with a choice of two different brands of motors to power our aircraft: Astroflite and Graupner. The engineering design process relies heavily on encouraging continuity between projects from year to year. Therefore, in analysis of the engine brands, instead of we based our decision off research conducted by USMA's DBF capstone team from the previous year (2001) and adapted it to account for some additional engine constraints and new competition rules. The engine brand comparison was conducted via a computer program called Ecalc, the industry standard in solving for the motor performance of R/C aircraft. This greatly simplified the analysis of engine efficiency, ultimately a function of the engine power output at any given operating range. For our analyses, the Amperage draw was assumed to be roughly 20 amps in order to yield 7 minutes of flight time (each mission was assumed to consist of approximately 3 minutes on the ground) for a 5 lb battery pack. Maximum achievable aircraft speed was calculated for both motors for purposes of relative comparison only as calculation of our actual aircraft speed will depend on many other factors. We determined that Astroflite was the better engine brand based on its higher efficiency over the engine operating range of power output. Table 4.5 summarizes the characteristics of both brands solved for in Ecalc and illustrates Astroflite as the obvious choice.

Engine Parameters Analyzed	Astroflite	Graupner
Thrust	173 oz.	158 oz.
Run Time (max throttle)	3.7 min	3.6 min
Max Speed	54 mph	52 mph
Watts Created	1364 W	1409 W
Watts Lost	142 W	206 W

Table 4.5: Engine Parameters tested using Ecalc software. From this sample analysis it is clear that Astroflite is the better choice for motor brand. *Adapted form of information taken from USMA 2001 DBF report.*

4.4.3(b) Analysis of Single vs. Double Engine Configuration. The single vs. double engine debate centered around the ability of the aircraft to fly given the new competition motor constraints. By conducting calculations given the battery pack constraint and anticipated engine performance, it was determined that for as much thrust as we required from the system, the current draw would be much too great and the available flight time would be seriously degraded. However, with the dual engine system, we would be able to reduce the current flow for the same amount of thrust. Figure 4.4 illustrates this concept for a general single vs. twin engine system as it shows a twin engine system producing a greater thrust at a lower drag for a constant current draw. These figures were created in Ecalc for Astroflite engines.

Single Engine System **Twin Engine System**

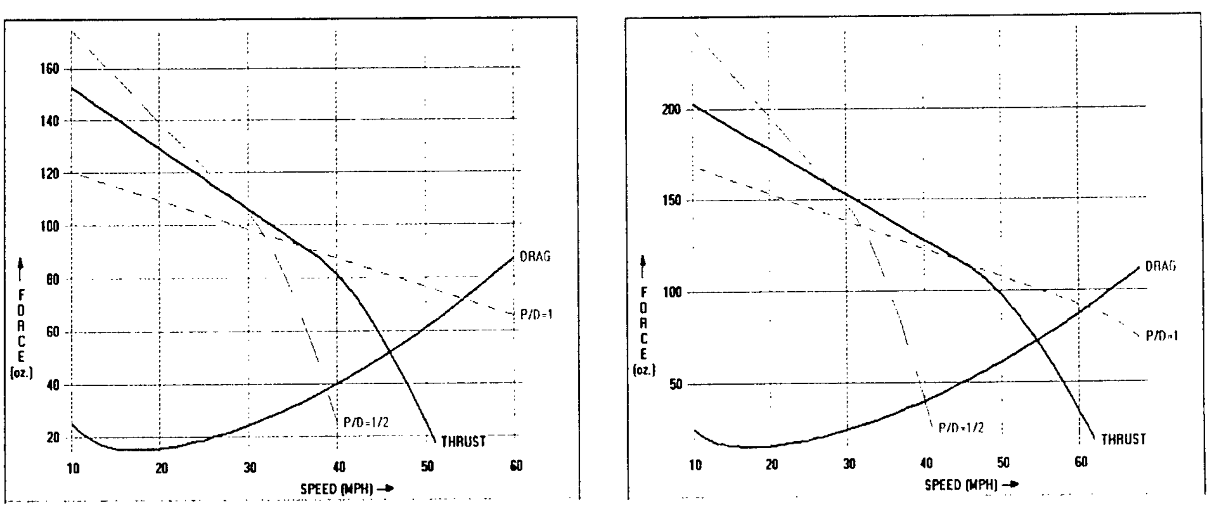


Figure 4.4: Comparison of a sample single engine and twin engine aircraft created in Ecalc. The intersection of the thrust and drag curves is at the cruising speed of each configuration. The twin engine system yields a higher cruising speed and thrust output for the same current draw.

Having chosen the twin engine configuration, we then completed several more power calculations in order to optimize the engine performance by configuring the values of voltage and current draw. The power available for the propellers was assumed to be 58% of the voltage draw times the current draw (see Figure 4.5).

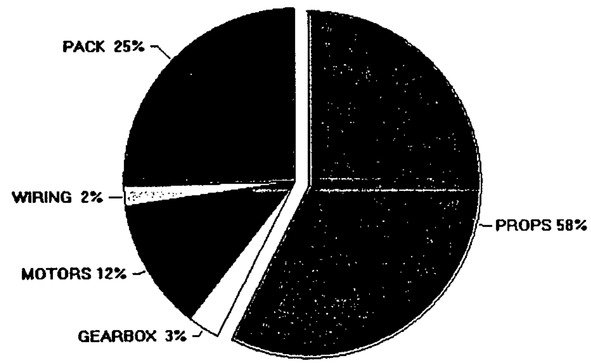


Figure 4.5: Power consumption in the propulsion system. This figure was created in Ecalc and gives an idea of the percentage of engine power available to the propeller.

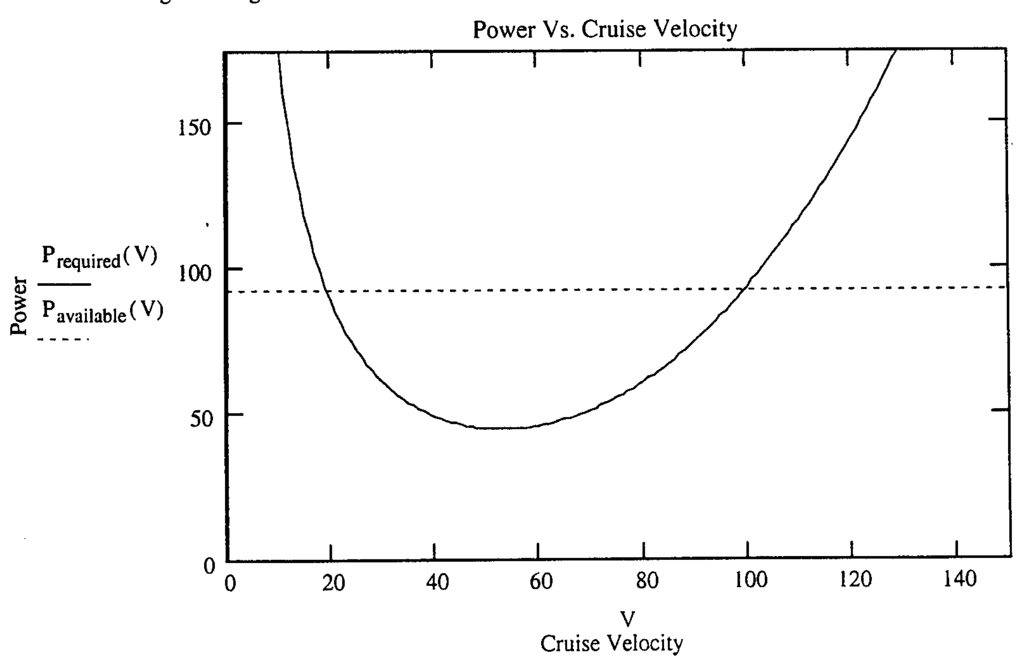
The propeller efficiency was assumed to be at 85% based on the advance ratio and NACA propeller efficiency charts. The total engine efficiency then could be used with the theoretical engine power output (equal to voltage draw times current draw) to solve for expected engine output. The power output was then superimposed on the calculated drag polar plot in terms of required power to overcome drag (developed later in the aircraft performance section of the detail design phase) in order to analyze the engine output and optimize the current and voltage draw parameters. The calculations shown below were created in a mathematical program called Mathcad.

$\eta_{engine} := .58$ $\quad \eta_{propeller} := .85$

$I := 17$ $\quad$ Current draw in amps

$E := 16$ $\quad$ Voltage draw in volts

$P_{available}(V) := \eta_{engine} \cdot \eta_{engine} \cdot I \cdot E$



The intersection of the power required and power available curves (intersection to the right, the left intersection is below the stall velocity and cannot be achieved) is the theoretical cruise velocity. We would expect, however, that our actual cruise velocity would be lower. Later in the performance optimization section, the velocity for maximum lift to drag ratio is found and the throttles are assumed to be set to achieve that airspeed. From there on, velocity for maximum lift to drag ratio is referred to as the aircraft cruise velocity.

4.3.3(c) Propeller Analysis. In analyzing the propeller or thrust producing device, we once again turned to some empirical information taken from the previous year's experimental research. To it, we added results from some of our own experiments. Current draw was set as close to 23 amps as was possible for each test run. The propeller was mounted on a homemade test stand and the thrust it produced was measured using a spring scale. The propeller with the maximum efficiency based on the thrust to current draw ratio (analogous to the thrust to weight figure of merit used in this design phase) was selected. We chose the Bolly Carbon Fiber 18X10 propeller (See Table 4.6).

Propeller	Diameter (in)	Pitch (J)	Current Draw (amps)	Thrust (lbs)	Thrust to Current Draw Ratio
Top Flight 15x8	15	8	23.5	6.5	0.28
Zinger 25x12	25	12	23.9	7.5	0.31
Top Flight 20x8	20	8	21.8	8.5	0.39
Classic 20x6	20	6	21.7	6.0	0.28
Graupner 3-Blade	18	6	23.5	5.9	0.25
Classic 18x10	18	10	23.5	6.5	0.28
Zinger 24X14	24	14	23.0	4.9	0.21
Bolly Carbon Fiber 18X10	18	10	23.0	9.0	0.39

Table 4.6: Experimental Results of propeller testing. The Bolly Carbon Fiber 18X10 propeller proved to be the most efficient. *Portions of this data were taken from the USMA 2001 DBF report.*

4.4.4 Landing Gear Analytical Methods. From earlier analyses, we had already concluded that the landing gear was to consist of a system with the main gear would at the rear of the aircraft and a frontal, spring damped structure at the nose. Analysis of this system required us to predict the loading conditions both structures would undergo during the landing sequence. We then had to set about designing a structure capable of withstanding the initial impact for the rear gear and a system able to quickly dampen oscillations for the nose gear. Additionally, both systems were required to incorporate interfaces to the aircraft wheels and fuselage and support the aircraft when on the ground.

4.4.4(a) Determination of Impact Loading. The loading on the aircraft during landing was modeled as an impact. Before any configuration design of the gear could be started, calculations had to be conducted to determine equivalent forces for these loads. The aircraft was assumed to land at its stall speed (calculated in the performance section of the detail design phase) and undergo a load factor of 1.2 during approach was assumed to calculate the vertical rate of descent. The impact of landing is assumed to take place over a tenth of a second and the aircraft horizontal velocity was assumed to drop

approximately 3 ft/s during this time. The Mathcad worksheet shown below summarizes the impact calculations and determination of equivalent loads.

$Vv := 15 \ \frac{ft}{s}$ — Rate of descent for a 1.2 load factor landing

Vstall = minimum velocity to sustain flight

Vh = Vstall = 39.034 ft/s

Stall velocity is calculated in perfromance analysis section of detail design phase

$Vh1 := 39.034$ right before touchdown

$Vh2 := 36$ right after touchdown

Assume the aircraft lands at its minimum horizontal velocity to sustain flight and drops about 3 ft/s instanteously on touchdown

(continued)

Assume landing load is an impact. Forces are due to impulses acting verticaly and horizontaly on the gear. Therefore, the change in momentum is equal to the equivalent force times the time over which it acts. Assume the time of force action is a tenth of a second.

Verticaly, the change in momentum is the mass times the rate of descent minus zero since the final vertical momentum is zero

Vertical Impulse

Change in Momentum = Impulse

$$m \cdot Vv - 0 = Fv \cdot (.1) \text{ solve}, Fv \rightarrow 7.763975155279503105 \cdot Vv$$

Horizontaly the change in momentum is the following:

$$m \cdot Vh1 - m \cdot Vh2 = Fh \cdot (.1) \text{ solve}, Fh \rightarrow 7.763975155279503105 \cdot Vh1 - 7.763975155279503105 \cdot Vh2$$

Therefore, the loads the gear must sustain are the following:

F vertical = 116.46 lbs

F horizontal = 23.56 lbs

4.4.4(b) Landing Gear Shape Analysis. In designing a general shape of the main landing gear, a computer simulation program called Cosmos Design Star was used. Cosmos used a mesh analysis and finite differencing methods, the computer was then able to yield stress, strain, and deformation plots of the design. Additionally, by specifying material properties to the landing gear, Cosmos was able to calculate the weight of the proposed component. We were then able to make modifications to our initial design of the main gear and test their impacts on the gear weight and structural integrity. Two of our more refined main gear proposals and their loading conditions in Cosmos are shown in Figure 4.6. Based on the stress distributions and weight characteristics of the main gear concepts tested, we ultimately decided on concept b for the final shape of the main landing gear. Further analysis and refinement allowed us to minimize the gear weight while still meeting structural and dimensional requirements.

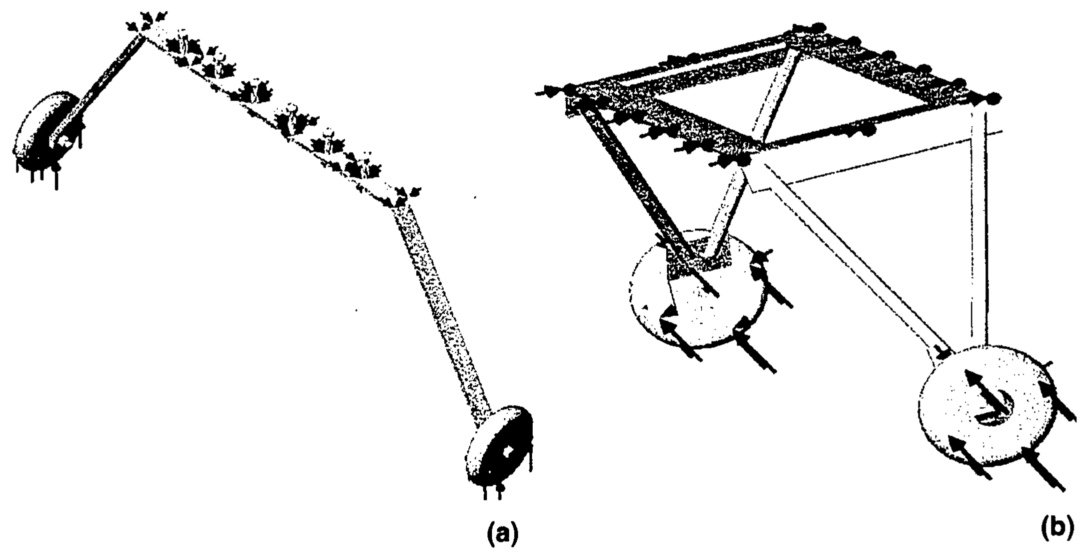


Figure 4.6: Two of the final landing gear shape concepts. A computer simulated finite differencing method was used to test the expected loading on these designs. The green arrows represent locations at which a fixed condition exists, and the red arrows represent the impact load.

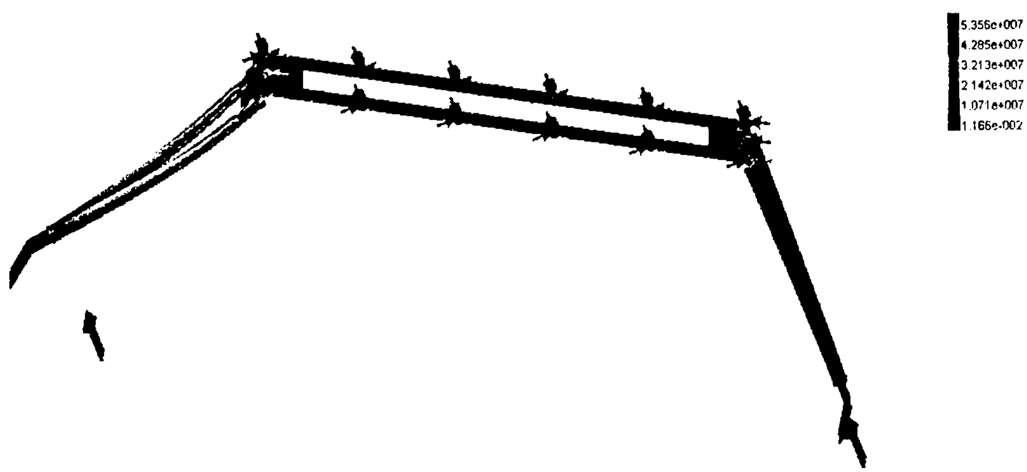


Figure 4.7: Stress gradient and deformation plot of the main gear for an off center landing. The strut deformation is well within the elastic region of the material and the stresses are below the yield values.

4.4.4(b) Nose Gear Design. In designing the spring for the nose landing gear system several considerations were taken into account. The oscillations in the spring had to be critically damped in order for the system to return to constant equilibrium in the least time possible. Furthermore, the total displacement of the nose gear had to be limited. A maximum displacement value of 1.2 inches was chosen. The system was assumed to be modeled by a simple, 1 dimensional harmonic response and the differential equation of motion, solution, and response plot are shown in the following Mathcad worksheet for values of the spring constant and damper that realizes the above parameter requirements.

$K := 50 \quad \frac{lb}{ft}$

$m := \frac{.5}{32} \quad slugs$

$v_i := 15 \quad \frac{ft}{s}$

$c := 113.14 \ \frac{slug}{s}$

K = Spring constant of the nose landing gear spring
m = Mass of nose landing gear
c = Dampening constant of nose landing gear
vi = Rate of aircraft descent
y = Displacent of nose landing gear
ωn = Natural frequency of the nose gear's oscillations
ωd = Damped frequency of the nose gear's oscillations
ξ = Dampening Factor of nose landing gear
t = Time after impact

$$m\cdot\frac{d^2}{dt^2}y + c\cdot\frac{d}{dt}y + K\cdot y = 0$$

Differential equation of motion for an unforced spring system

Let $\omega_n := \sqrt{\frac{K}{m}} \qquad \xi := \frac{c}{2\cdot\omega_n} \qquad \omega_d := \omega_n\cdot\sqrt{1-\xi^2}$

$$y = e^{-\xi\cdot\omega n\cdot t}\cdot\left(c1\cdot\cos\left(\omega_d\cdot t\right) + c_2\cdot\sin\left(\omega_d\cdot t\right)\right)$$

$$\frac{d}{dt}y = -e^{-\xi\cdot\omega n\cdot t}\cdot\left[c_1\cdot\left(\omega_d\right)\cdot\sin\left(\omega_d\cdot t\right) + c_2\cdot\left(\omega_d\right)\cdot\cos\left(\omega_d\cdot t\right)\right] - \xi\cdot\omega n\cdot e^{-\xi\cdot\omega n\cdot t}\cdot\left(c1\cdot\cos\left(\omega_d\cdot t\right) + c_2\cdot\sin\left(\omega_d\cdot t\right)\right)$$

$c_1 := 0$

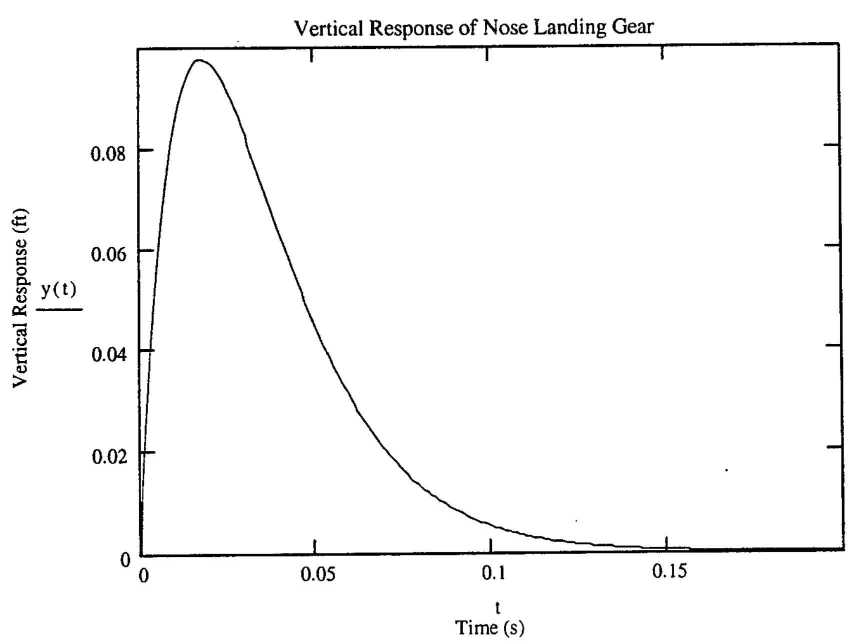
$c_2 := \frac{v_i}{\left(\omega_d\right)}$

Assuming zero initial displacement and initial velocity (downward) of 15 ft/s (1.2 g landing), solve for initial conditions

$t := 0, .001 .. 10$

$y(t) := e^{-\xi\cdot\omega_n\cdot t}\cdot\left(c_1\cdot\cos\left(\omega_d\cdot t\right) + c_2\cdot\sin\left(\omega_d\cdot t\right)\right)$

Response of nose gear after impact



Having designed for the spring constant and spring damper value in order to achieve the desired front gear oscillations, it is now necessary to design the physical spring for the nose landing gear. The spring is assumed to be fashioned out of aluminum (modulus of rigidity = 10.4 Mpsi) with a ½ inch thickness. The spring is also assumed to have plain ends meaning the number of active coils equals the number of total coils. The properties of the spring can then be solved for.

$d := .0417$

$G := 10400000$

$D := .25$

$N := 5.0$

K = Spring constant
G = Material Modulus of rigidity (10.4 Mpsi)
d = Spring thickness (assumed to be .0417 ft)
D = Spring diameter
N = Number of coils

$$K := \frac{d^4 \cdot G}{8 \cdot D^3 \cdot N}$$

$$K = 50.315 \quad \frac{lb}{ft}$$

4.5 Preliminary Design Results

Wing	Materials	Foam, Carbon Fiber
	Wing Placement	Middle
	Airfoil	NACA 7410
	Dihedral Angle	11 degrees (with flat inboard panel 1.5 ft long)
	Planform Area	10 ft^2
Control Surfaces	Materials	Foam, Carbon Fiber
	Tail Placement	High
Fuselage	Materials	Aluminum, Balsa
	Type	Conventional with tail booms
	Length	10 inches
	Width	8 inches
	Height	8 inches
Propulsion	Engine Brand	Astroflite
	Number of Engines	2
	Current Draw	17 V
	Voltage Draw	16 amps
	Propeller	Bolly Carbon Fiber 18X10
Landing Gear	Materials	Aluminum
	Main Gear Height	7.10 inches
	Main Gear Wheel Base	30 inches
	Nose Gear Spring Constant	50 lb/ft
	Nose Gear Dampening Constant	113.14 slug/s
	Nose Gear Spring Diameter	3 inches
	Nose Gear Number of Spring Coils	5

Table 4.7: Summary of significant conclusions from the preliminary design phase.

5.0 Detail Design

5.1 Performance Data

To conduct the analysis of the aircraft performance a mathematical worksheet within Mathcad was constructed. Using the worksheet format, the eventual performance characteristics could be optimized by simply changing the assumed values of the parameters at the top of the sheet. For many portions of the previously described design phases, this performance analysis was done in tandem with the various analytical techniques in order to arrive at a components optimum in terms of our design figures of merit as well as ultimate aircraft performance.

5.1.1 Aircraft Drag Polar. The drag polar of the aircraft is a relationship between its coefficient of lift and coefficient of drag. For the entire aircraft, it can be represented graphically, as was done for the airfoil choices in the preliminary design phase, or can be represented analytically. To derive the total aircraft drag polar we broke the aircraft coefficient of drag down into its component due to lift and component due to parasite drag and then rearranged the expression to relate the coefficient of lift to the coefficient of drag. Shown below is the commented Mathcad worksheet used to accomplish this derivation and the resulting plots created internal to the worksheets.

Definition of Symbols

V	= Velocity
K	= Drag Similarity Parameter
e	= Spanwise Efficiency Factor
AR	= Wing Aspect Ratio
S	= Wing Planform Area
W	= Estimated Aircraft Weight
ρ	= Freestream Fluid Density
D	= Drag Force
CL	= Aircraft Coefficient of Lift
CD	= Aircraft Coefficient of Drag
CDo	= Aircraft Parasite Drag Coefficient
$P_{required}$	= Required Power

Parameters

$W := 25$	Aircraft Weight
$S := 10$	Planform Area = $b \cdot c$
$AR := 10$	Aspect Ratio = $\frac{b^2}{S}$
$e := .95$	Efficiency Factor: from published plots
$\rho := .002378$	Freestream Density: assume standard sea-level conditions

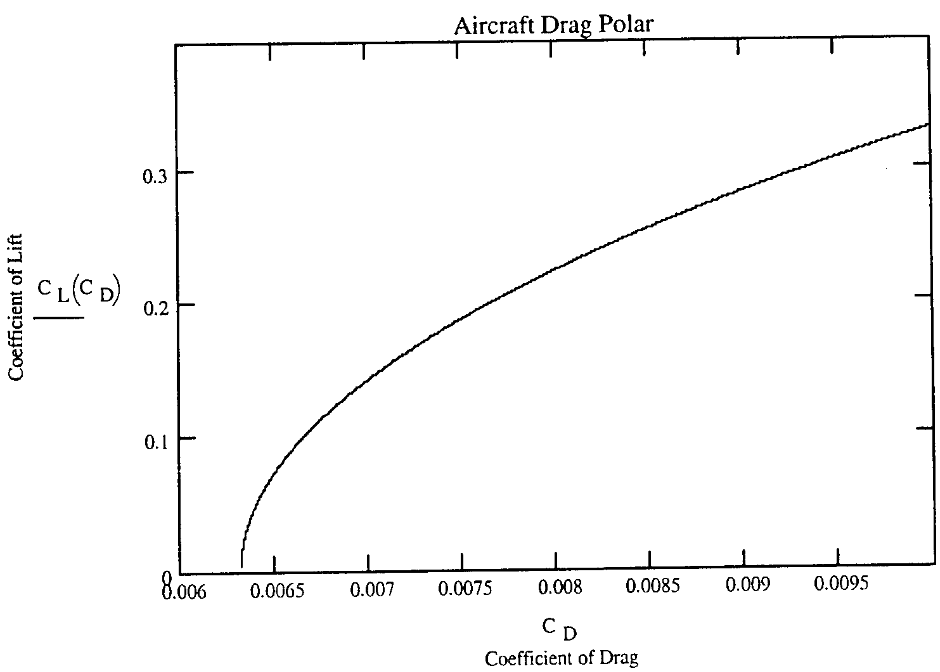
Drag Calculations

$K := \dfrac{1}{\pi \cdot e \cdot AR}$	Similarity Parameter
$V := 75$	Estimation of Cruise Velocity
$C_L := \dfrac{2 \cdot W}{\rho \cdot V^2 \cdot S}$	Determination of Aircraft Coefficient of Lift: assume straight and level flight
$C_L = 0.374$	
$C_D := .011$	From NACA Drag Polar: assume aircraft coefficient of drag equal to that of wing
$C_D = C_{Do} + K \cdot C_L^{\,2}$	General Form of the Drag Polar Equation
$C_{Do} := C_D - K \cdot C_L^{\,2}$	Solve for the Parasite Drag Coefficient
$C_{Do} = 6.318 \cdot 10^{-3}$	Parasite Drag Coefficient

Drag Polar

$C_D := 0, .000001 .. .05$

$C_L(C_D) := \sqrt{\dfrac{C_D - C_{Do}}{K}}$	Coefficient of Lift as a Function of Coefficient of Drag



5.1.2 Power Analysis. Having established the aircraft drag polar, we were then able to relate the power necessary to overcome drag to the free-stream velocity and depict the results in the form of a plot. Basically the drag polar plot in a different form, this power vs. velocity relationship is used often in aircraft performance analyses.

Power Calculations

$$V := 0..\,200$$

$$C_L = \frac{2\cdot W}{\rho\cdot V^2\cdot S}$$

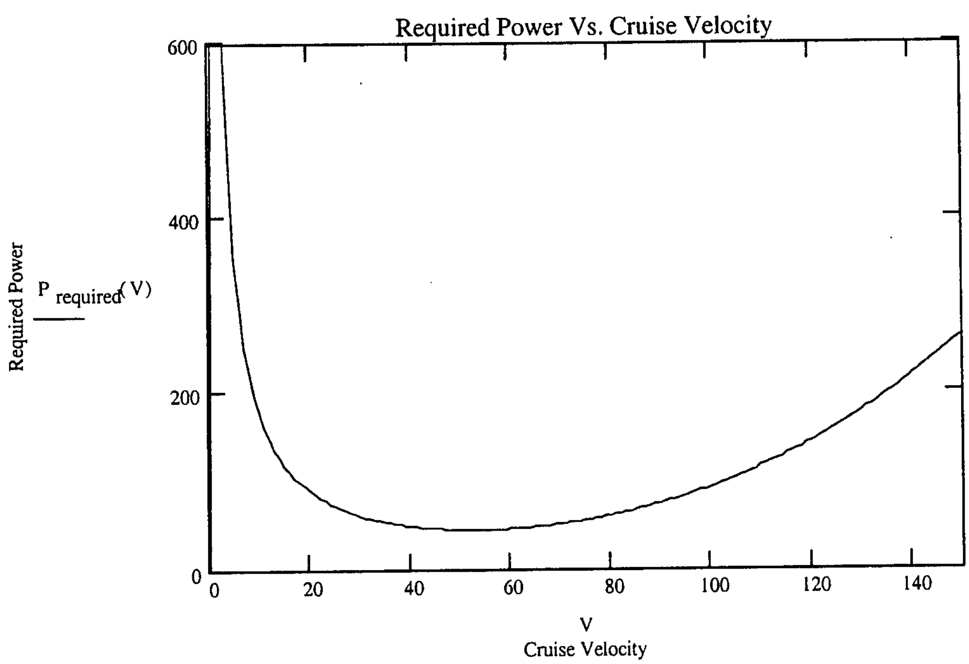
$$C_D = C_{Do} + K\cdot C_L^{\,2}$$

$$D = \frac{1}{2}\cdot\rho\cdot V^2\cdot S\cdot C_D$$ Determination of Drag Force

$$P_{required} = D\cdot V$$ Power Required to Overcome Drag Force

Power Required vs. Cruise Velocity Plot

$$P_{required}(V) := \left[\frac{1}{2}\cdot\rho\cdot V^2\cdot S\cdot\left[C_{Do} + K\cdot\left(\frac{2\cdot W}{\rho\cdot V^2\cdot S}\right)^2\right]\right]\cdot V$$



5.1.3 Aircraft Performance Calculations. Using the worksheet shells shown above and the parameters presented thus far, virtually every aspect of aircraft performance can be solved for.

Aircraft Performance

Propeller Specifications

$$J = \frac{V}{N \cdot D}$$

$\eta_{propeller} := .85$ Derived from advance ratio and propeller efficiency curves

Maximum Efficiency, Velocity of Max Efficiency, and Velocity of Max Endurance

$$Efficiency_{max} := \frac{1}{\sqrt{4 \cdot K \cdot C_{Do}}}$$

$Efficiency_{max} = 19.324$

Maximum Efficiency

$$V_{maxEfficiency} := \left[\frac{2}{\rho} \cdot \sqrt{\frac{K}{C_{Do}}} \cdot \left(\frac{W}{S}\right)\right]^{\frac{1}{2}}$$

$V_{maxEfficiency} = 52.181 \ \frac{ft}{s}$

Velocity of Maximum Efficiency

$$\frac{d}{dV}\left[\frac{1}{2} \cdot \rho \cdot V^2 \cdot S \cdot \left[C_{Do} + K \cdot \left(\frac{2 \cdot W}{\rho \cdot V^2 \cdot S}\right)^2\right]\right] \cdot V$$

$V := 40$

Given

$$\frac{3}{2} \cdot \rho \cdot V^2 \cdot S \cdot \left[C_{Do} + 4 \cdot K \cdot \frac{W^2}{\left(\rho^2 \cdot V^4 \cdot S^2\right)}\right] - \frac{8}{\left(\rho \cdot V^2 \cdot S\right)} \cdot K \cdot W^2 = 0$$

$V_{maxEndurance} := Find(V)$

$V_{maxEndurance} = 39.649 \ \frac{ft}{s}$

Velocity of Maximum Endurance

Cruise Properties

Assume Cruise is at $V_{maxEfficiency}$

$$W = \frac{1}{2} \cdot \rho \cdot V_{cruise}^{2} \cdot S \cdot C_L$$

$$C_{L.cruise} := \frac{2 \cdot W}{\rho \cdot V_{cruise}^{2} \cdot S}$$

$C_{L.cruise} = 0.434$

Coefficient of Lift to sustain cruise flight

$$C_{D.cruise} := C_{Do} + K \cdot C_{L.cruise}^{2}$$

$C_{D.cruise} = 0.013$

Coefficient of Drag in Cruise Flight

$\alpha := 1.2 \quad deg$

Cruise Angle of Attack

Minimum Velocity for Sustained Flight and Maximum Available Thrust

$C_{Lmax} := .7$

$$V_{minimum} := \sqrt{\frac{2 \cdot W}{\rho \cdot C_{Lmax} \cdot S}}$$

$$T_{available} := \frac{\eta_{propeller} \cdot P_{rated}}{\left(V_{minimum}\right)}$$

$$\frac{T_{available}}{W} = 0.298$$

Maximum Thrust to Weight Ratio

Take Off and Landing Performance

$C_{Lmax} := 1.38$

$$V_{stall} := \sqrt{\frac{2}{\rho} \cdot \left(\frac{W}{S}\right) \cdot \frac{1}{C_{Lmax}}}$$

$V_{stall} = 39.034 \quad \frac{ft}{s}$

Apporximate landing speed

$$S_{ground} := \frac{1.21 \cdot \left(\frac{W}{S}\right)}{\left(32.2 \cdot \rho \cdot C_{Lmax}\right) \cdot \left(\frac{T_{available}}{W}\right)}$$

$S_{ground} = 96.137 \quad ft$

Ground Roll / Takeoff distance (assuming no obstacles)

Turning Performance

$$n_{max} := \left[\frac{\frac{1}{2} \cdot \rho \cdot V_{cruise}^2}{K \cdot \left(\frac{W}{S}\right)} \cdot \left[\left(\frac{T_{available}}{W}\right) - \frac{\left(\frac{1}{2} \cdot \rho \cdot V_{cruise}^2\right) \cdot C_{Do}}{\left(\frac{W}{S}\right)} \right] \right]^{\frac{1}{2}}$$

$n_{max} = 4.257$ g's

Maximum Load Factor

$$R := \frac{V_{cruise}^2}{32.2 \cdot \sqrt{n_{max}^2 - 1}}$$

$R = 36.342$ ft

Minimum turing radius

$$\omega := \frac{32.2 \cdot \sqrt{n_{max}^2 - 1}}{V_{cruise}^2} \cdot 2 \cdot \pi$$

$\omega = 0.173 \ \frac{rad}{s}$

Maximum turning rate

$$Time := \frac{2 \cdot \pi}{\omega}$$

$Time = 36.342$ s

Minimum time to complete a full circle turn

Gliding Flight Performance

$$\theta_{minSink} := atan\left(\frac{1}{Efficiency_{max}}\right)$$

$$\theta_{minSinkdeg} := \theta_{minSink} \cdot \frac{360}{2 \cdot \pi}$$

$\theta_{minSinkdeg} = 1.667$ deg

Minimum sink angle

$$SinkRate := \sin\left(\theta_{minSink} \cdot deg\right) \cdot \sqrt{\frac{2 \cdot W \cdot \cos\left(\theta_{minSink}\right)}{\rho \cdot S \cdot C_{L.cruise}}}$$

$SinkRate = 0.035 \ \frac{ft}{s}$

Sink rate in glide at minimum sink angle

Over the course of the previous calculations, the cruise velocity was assumed to be set so as to achieve the maximum lift to drag ratio. For the sake of clarification, from this point on, cruise velocity will refer to this maximum efficiency velocity.

Performance	
Efficiency (max L/D)	19.324 ft/s
Wing Loading (W/S)	2.5 lb/ft^2
Maximum Thrust to Weight Ratio (T/W)	0.278
Cruise Characteristics	
Cruise Velocity	52.18 ft/s
Coefficient of Lift @ Cruise	0.434
Coefficient of Drag @ Cruise	0.013
Angle of Attack	1.2 deg
Turning Performance	
Turn Radius	34.62 ft
Turn Rate	.173 rad/s
Glide Performance	
Minimum Sink Angle	1.667 deg
Minimum Sink Rate	0.035 ft/s
Landing Performance	
Ground Roll	102.9 ft
Approach Angle	14.3 deg
Approach Distance	155.2 ft

Table 5.1: Summary of aircraft performance data.

5.2 Estimated Mission Performance

The parameters displayed in the Mathcad worksheets are the result of an iterative study of the aircraft performance parameters and their impact on the mission profile. The values that we have presented yield aircraft performance characteristics that best meet the demands of the mission the competition calls for. Our final design is expected to cruise at an airspeed of 69.58 ft/s. Neglecting the turns, the mission calls for the aircraft to fly a pattern with an equivalent distance of 2000 ft. This means that at its cruise velocity, the aircraft will require 28.75 s to fly the straight portions of the pattern. The 360 degree turn required by the competition rules can be accomplished with our design at a radius of 36.34 ft and can be completed in approximately 36 seconds. We predict that landing, braking, retrieving the aircraft, and loading or unloading the payload can be accomplished in 90 seconds. Therefore, a conservative estimate (with 30 seconds added for taxi, turns along the course, and other unforeseen circumstances) for the time to complete a single flight sortie is approximately 185 seconds or a little over 3 minutes. If this is accomplished 3 sorties should be able to be completed with maximum payload for the loaded lap in the 10 minute time limit. We feel this would make our aircraft extremely competitive from studying the results of last year's competition.

5.3 Weight and Balance. The specification of the center of gravity and the determination of forces needed to balance the aircraft are critical considerations in finalizing the design of the aircraft. To find the position of the aircraft center of gravity relative to the wing quarter chord point (assumed to be the aerodynamic center of the aircraft), the total moments forward and aft of the quarter chord must be

calculated and summed. The net total moment is then divided by the weights of all the components contributing to it. For our particular aircraft configuration, the weight of the payload is evenly balanced on either side of the wing as is the structure of the fuselage. To simplify the calculations, it is assumed that this weight distribution is symmetrical about the quarter chord point of the wing rather than the half chord point. The distance between these points is practically negligible and such an assumption allowed us to conclude that the payload and fuselage were balanced evenly about the center of pressure. Other component weights requiring consideration, at various distances forward and aft of the quarter chord point, include the main and nose landing gear, the tail system, the engines, and the battery pack. Figure 5.1, the aircraft weight and balance sheet, illustrates these weights and their respective moment arms. Table 5.1 depicts the calculation of the center of gravity point. The aircraft dimensions were designed to ensure a center of gravity forward of the center of pressure and the calculations confirm this as they yield a center of gravity location 2.58 inches forward of the wing center of pressure.

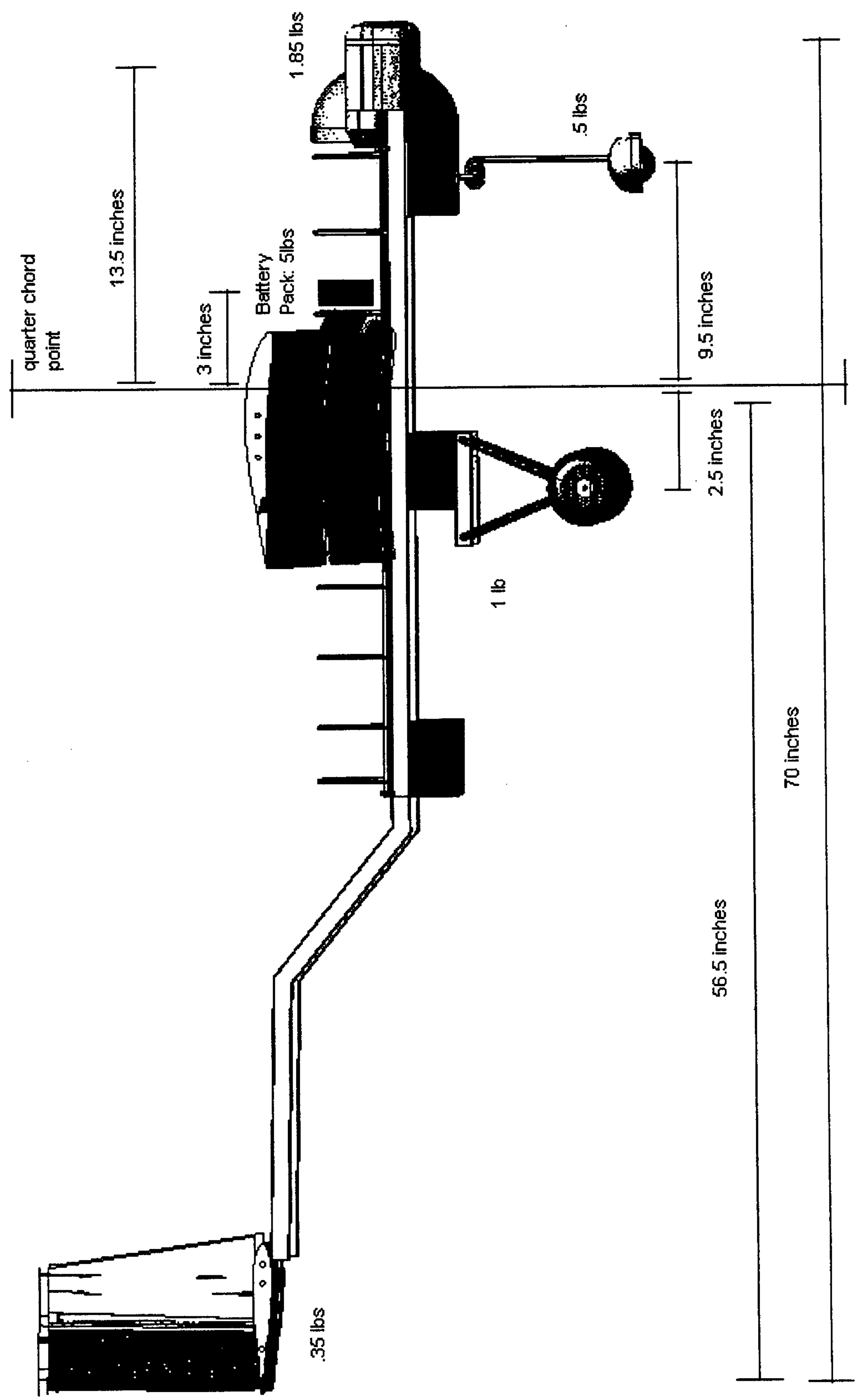


Figure 5.1: Weight and Balance Sheet. All significant, unbalanced component weights are depicted.

Forward of Quarter Chord			Aft of Quarter Chord		
Component	**Weight**	**Moment Arm**	**Component**	**Weight**	**Moment Arm**
Engines	1.85	13.5	Rear Landing Gear	1.0	2.5
Nose Landing Gear	0.5	9.5	Tail Assembly	0.4	56.5
Battery Pack	5.0	3.0			

Center of Gravity Location:	2.58045977 inches forward of 1/4 chord

Table 5.1: Significant weights forward and aft of the quarter chord point and the calculation of the center of gravity location

5.4 Aircraft Handling Qualities

The handling quality refers to the stability and maneuverability of the aircraft in the lateral, longitudinal, and transverse planes. Dynamic stability and maneuverability are inversely related to one another. That is, as the aircraft maneuverability increases, it becomes more responsive to the controls of the pilot and it is less likely to return to some equilibrium position given some perturbation. On the other hand, as dynamic stability (roll, pitch, or yaw stiffness) increases, the aircraft requires increasing input controls to deviate from a trimmed condition and is more likely to return to equilibrium flight if disturbed. For the purposes of this project, we were much more concerned with safety and stability than with maneuverability. Therefore in designing the overall aircraft, care was taken to ensure positive stiffness about the rolling, yawing, and pitching axes. The calculations used to determine these stability criteria are presented below.

5.4.1 Longitudinal Stability. Longitudinal stability or pitch stiffness refers to the aircraft's tendency to return to trim after a pitching disturbance. A parameter governing the degree of longitudinal stability is the aircraft static margin or the difference between the aircraft neutral point location and the aircraft center of gravity location. A positive static margin, and hence positive longitudinal stability, is therefore associated with a condition in which the neutral point is located aft of the center of gravity. In performing this analysis, it was first necessary to determine the neutral point of the aircraft as a whole assuming that the neutral point of the wing body was located at the quarter chord location. The following equations applied.

$$h_n = h_{n_{wb}} + \frac{a_t}{a}\cdot V_H\cdot\left(1 - \frac{d}{d\alpha}\varepsilon\right) - \frac{1}{a}\frac{d}{d\alpha}C_{mo} + \frac{d}{d\alpha}C_p$$

h_n = Aircraft Neutral Point (wing chord lengths)
h_{nwb} = Wing Body Neutral Point (wing chord lengths)
a_t = Tail Lift Curve Slope
a = Wing Lift Curve Slope
V_H = Horizontal Tail Volume
C_{mo} = Pitching Moment at 0 Angle of Attack
C_p = Pitching Moment Due to Propulsion System
α = Angle of Attack

$\frac{d}{d\alpha}C_p = 0$ $\frac{d}{d\alpha}\varepsilon = 0$

$h_{nwb} := .25$ $a := 2\pi$

$a_t := 2\cdot\pi$ $C_{mo\alpha} := -.01$ $V_H := .27115$

$$h_n := h_{nwb} + \frac{a_t}{a}\cdot V_H - \frac{1}{a}\cdot C_{mo\alpha}$$

$h_n = 0.523$ Aircraft Neutral Point

The engines were assumed to be mounted in a plane parallel to that of the center of gravity thus eliminating the pitching moment due to the propulsion system and the tail downwash was assumed to be negligible given its high placement relative to the wing. With the aircraft neutral point (in terms of wing chord lengths) and the center of gravity, which was calculated in the weight and balance section, the static margin can then be determined.

$h := 2.58$ $h_n := .523$

$c := 12$

$h := \frac{h}{c}$

K_n = Aircraft Static Margin
h_n = Aircraft Neutral Point (wing chord lengths)
h = Aircraft Center of Gravity (wing chord lengths
c = Wing Chord Length (inches)

$K_n := (h_n - h)$

$K_n = 0.306$ Static Margin

5.4.2 Weathercock Stability. The equation for yaw stiffness or "weathercock stability" is shown below. The coefficient $C_{n,\beta}$ quantifies the tendency to return to equilibrium after a yaw disturbance. The resulting $C_{n,\beta}$ was .0002416 and the aircraft was therefore assumed to be stable in this plane.

$$\frac{C_{n\beta}}{C_L^{\ 2}} = \frac{1}{57.3}\left[\frac{1}{4\pi A} - \frac{\tan\Lambda_{c/4}}{\pi A(A + 4\cos\Lambda_{c/4})}\left(\cos\Lambda_{c/4} - \frac{A}{2} - \frac{A^2}{8\cos\Lambda_{c/4}} + 6(h_{n_w} - h)\frac{\sin\Lambda_{c/4}}{A}\right)\right]$$

5.4.3 Roll Stiffness. The equation for roll stiffness is shown below. Numerical analysis of this equation revealed that α_x = .00272 meaning the aircraft would remain stable in this plane.

$$\frac{\partial C_l}{\partial\phi} = C_{l_\beta}\alpha_x$$

5.5 COMPONENT SELECTION

5.5.1 Servos. The servo motors that we chose are off the shelf from Futaba RX Company. There are a total of 4 servo motors in our airplane. One for both the left and right ailerons, one for the tail rudder, and one for the elevator. The servo motor connected to our tail rudder will also control our front landing gear. These servos are the electrical to mechanical interface for the rudder, wheel, elevator, and both ailerons. Our receiver and transmitter are also from Futaba. The reason we chose Futaba is because they have the largest selection of radio controlled components at a reasonable cost. Futaba was also recommended to us by groups from past competitions.

The two servo motors for the ailerons will help the pilot control the steering of the aircraft. These servo motors are small because they do not have a lot of force to push against to steer our aircraft. These servo motors will be located in the middle of each wing and will be connected with a short cable directly to the aileron.

The servo motors for the tail rudder and elevator are larger because they require more force to operate than the servo motors for the ailerons. These servo motors will be centrally located in the middle of the fuselage and will be connected to the control surfaces using longer control cables. The cable that controls our twin tail rudders will be spliced into a "Y" and then connected to each tail rudder. There will also be a separate cable connected to the same servo that will control the steering of the front landing gear.

5.5.2 Batteries. The batteries packs were built by SR Battery Company. We selected SR Battery Company to build our battery packs because they were recommended to us by past groups and they have a good history of reliability. Our battery system is made up of two separate battery packs with 17 cells each. The cells are 2400 Series High Rate Max Cells. Each cell is rated at 2400 mAH and 1.2 Volts. Each battery pack is rated at 20.4 Volts and weighs 2.33 pounds. Together the battery packs weigh 4.66 pounds, which is below the requirement of 5 pounds total for the battery packs.

The reason we chose to use two separate battery packs instead of a single large one is due to the 40 amp fuse constraint per battery pack. Because we are using two engines in our design that are rated at close to 40 amps with the propellers that we have chosen, we felt that having a separate battery pack for each engine would supply each engine with more power and increase our overall thrust and speed.

5.6 Systems Architecture

Once the detail portion of our design was complete the major components were drawn in Pro-Design, a CAD software program. The connections between the different subsystems were drawn and tested in Pro-Design through the creation of an assembly of the entire aircraft. This allowed us to see the entire aircraft prior to its being built and look for any major connections or overlooked interface difficulties. Using this we looked closer for proper clearance and balance. Using this assembled plane as a model the team sat down and developed a plan on how to integrate each major part during the construction

design phase. This overall design made it clear in each team members mind what the form of the final design was to look like in three dimensions.

5.7 Rated Aircraft Score

The rated aircraft score is a measure of the complexity and level of technology associated with the design. For the purposes of the competition, maximum scoring is achieved with a minimum possible rated aircraft cost. The RAC was optimized along with the performance and figure of merit parameters to come to the best overall design. The final rated aircraft cost worksheet is shown below in table 5.2.

Rated Aircraft Cost Worksheet

Coefficient	Description	Value
A	Manufacturers Empty Weight Multiplier	**100**
B	Rated Engine Power Multiplier	**1500**
C	Manufacturers Cost Multiplier	**20**
MEW	Manufacturers Empty Weight	**17.85**
Wing	3.5	
Propulsion	6.85	
Landing Gear	1.5	
Fuselage	6	
REP	Rated Engine Power	**2.25**
# engines	2	
Battery Weight (lbs)	5	
MFHR	Manufacturer Man Hours	**241.97**
Wing Span	11.42	91.36
Max Chord	0.91	7.28
Control Surface	5	15
Tip to Tail Length	5.833	58.33
Vertical Surface No active control	0	0
Vertical Surface Active control	2	20
Horizontal Surface	1	10
Servo or Motor control	4	20
Engine	2	10
Propeller	2	10

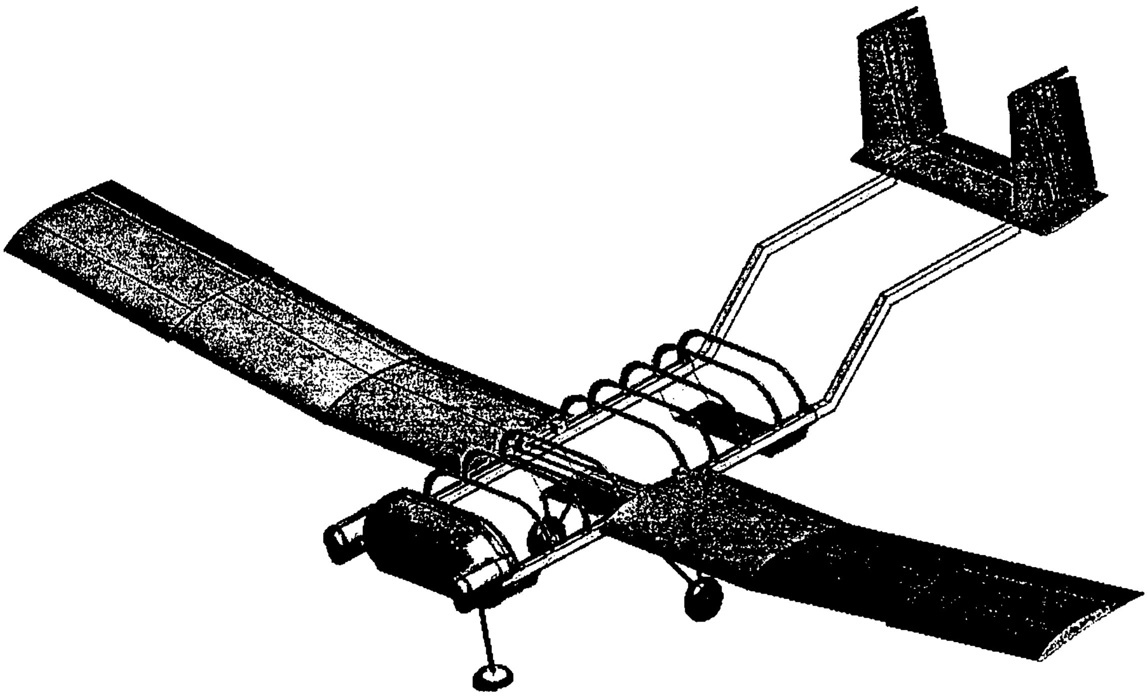
Rated Aircraft Cost, $ (Thousands) = (A*MEW + B*REP + C*MFHR)/1000

RAC	**9.9994**

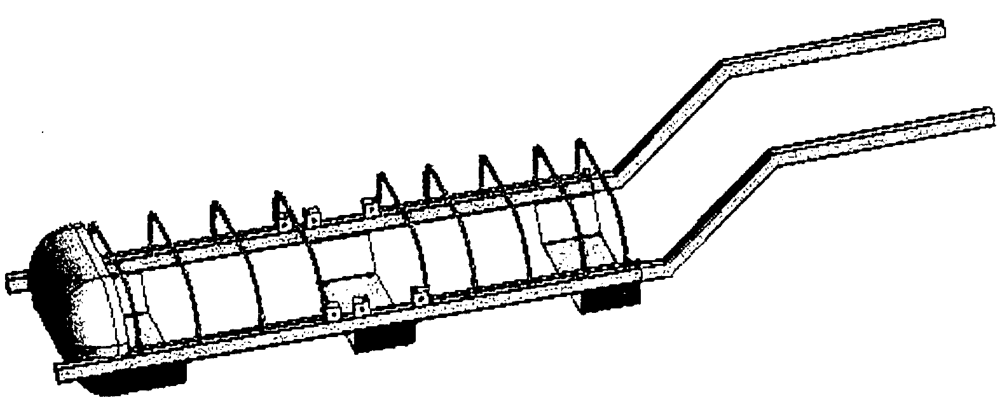
Figure 5.2: Rated Aircraft Cost worksheet. The final RAC value is highlighted in yellow at the bottom.

5.8 Drawing Package

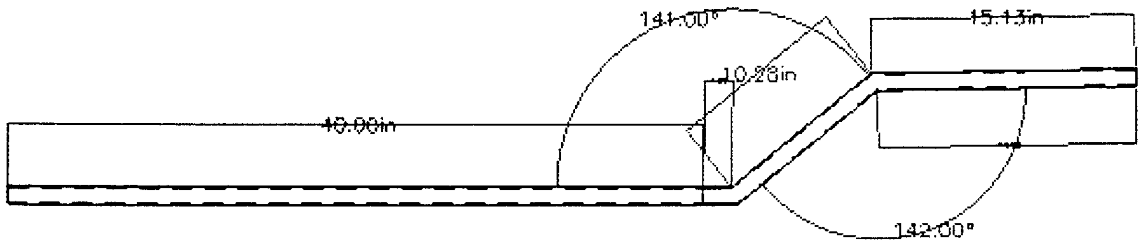
5.8.1 Aircraft Assembly



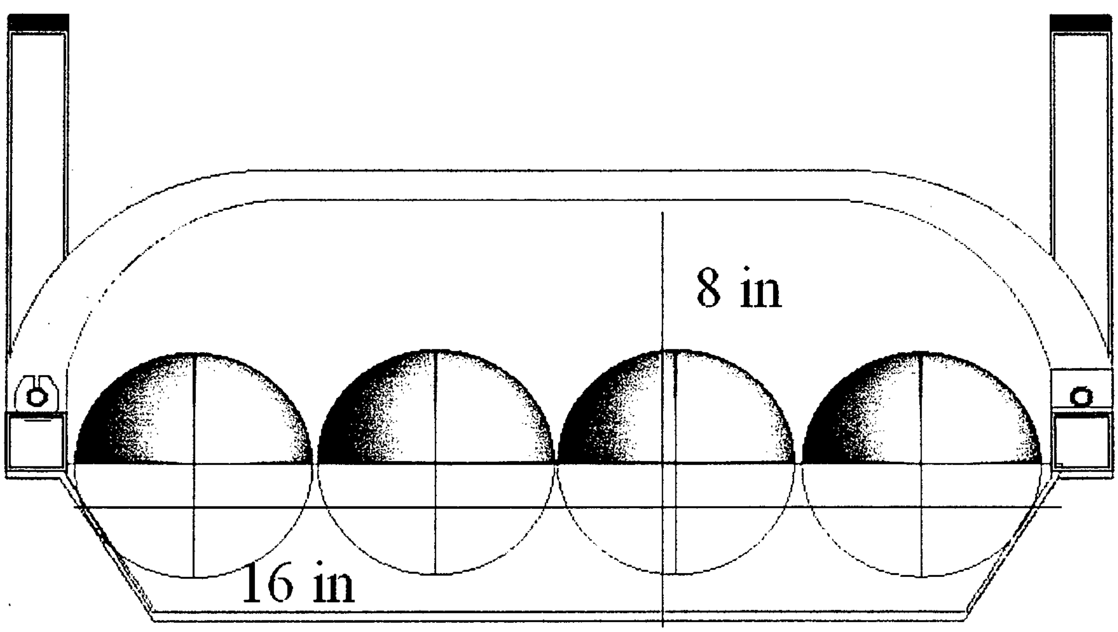
5.8.2 Fuselage



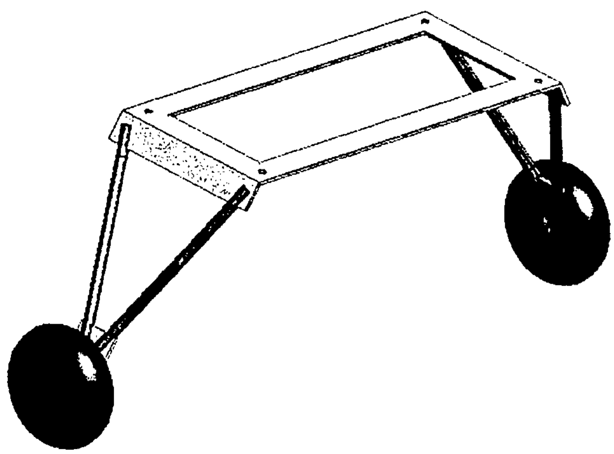
5.8.2(a) Fuselage Main Spar 2-D Drawing

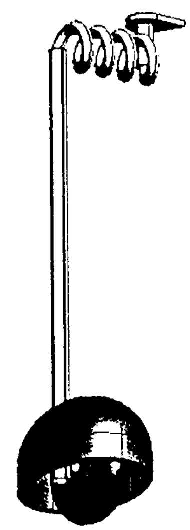


5.8.2(b) Payload Placement

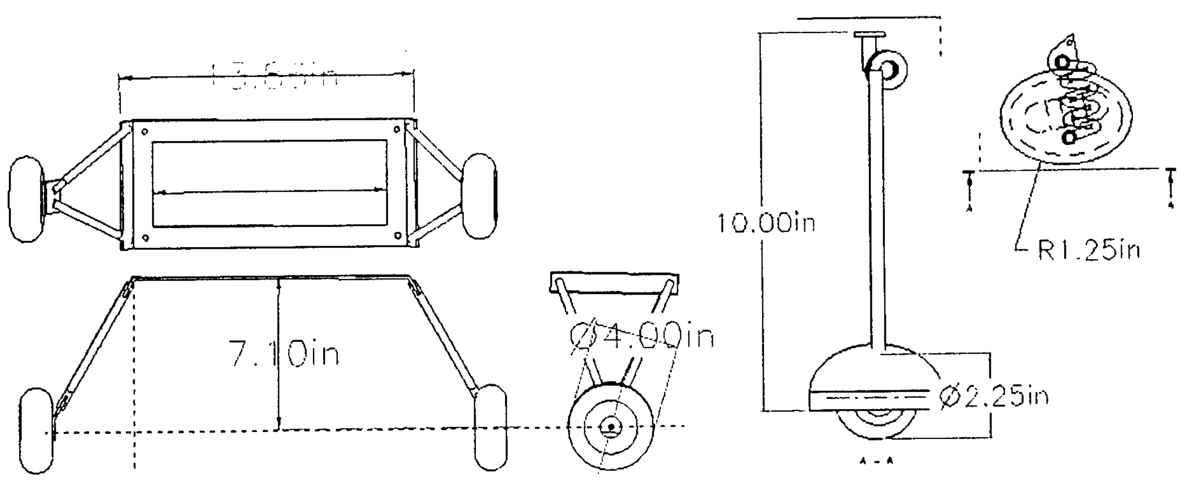


5.8.3 Landing Gear Main/Nose

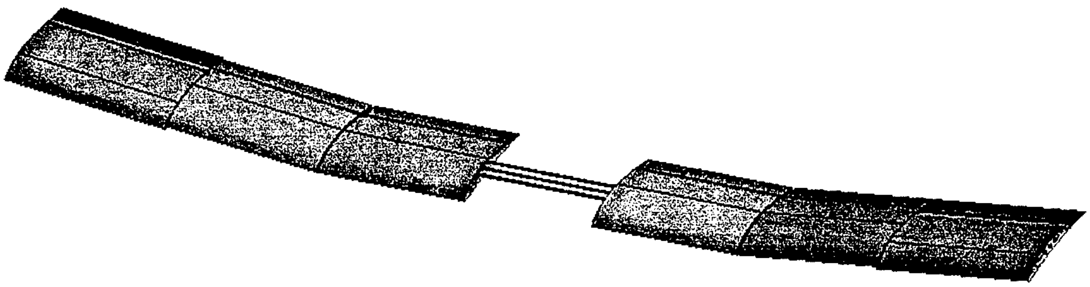




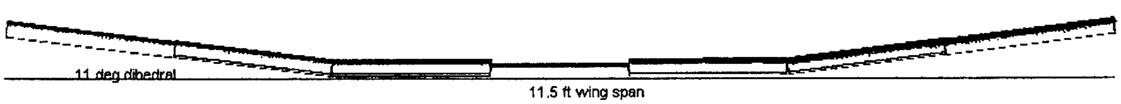
5.8.3(b) Main Landing Gear 2-D Drawing and Nose Landing Gear 2-D Drawing

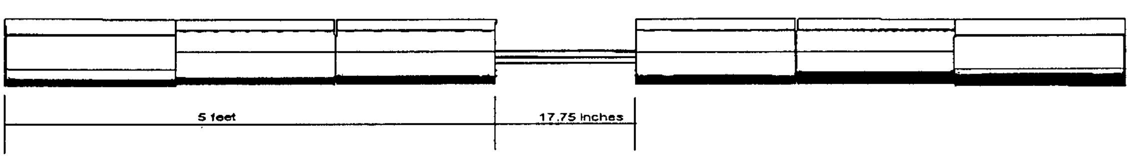


5.8.4 Wing

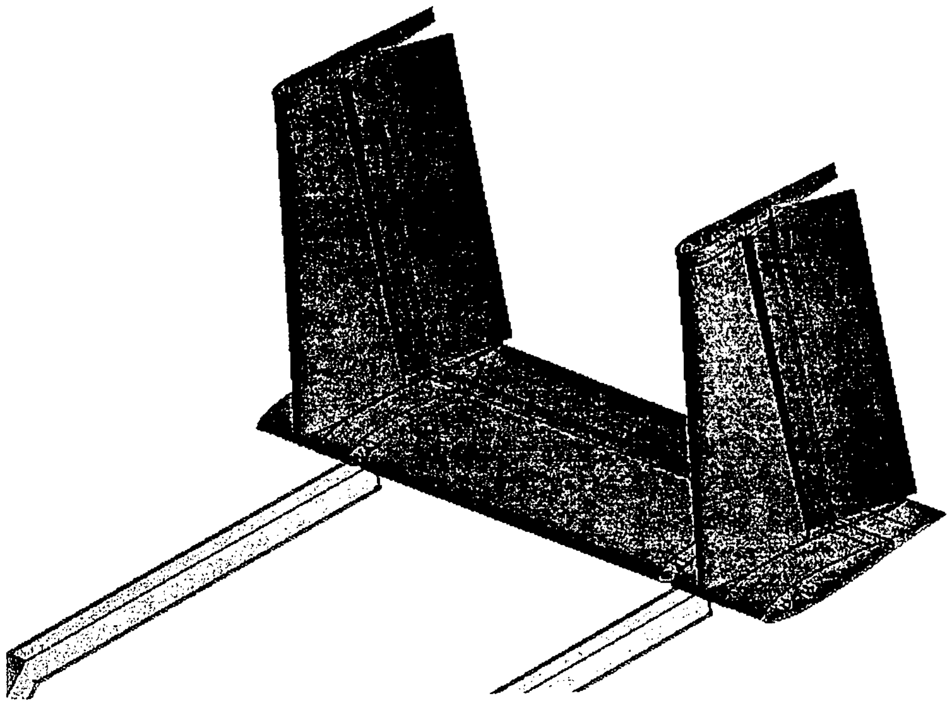


5.8.4(a) Wing Drawings

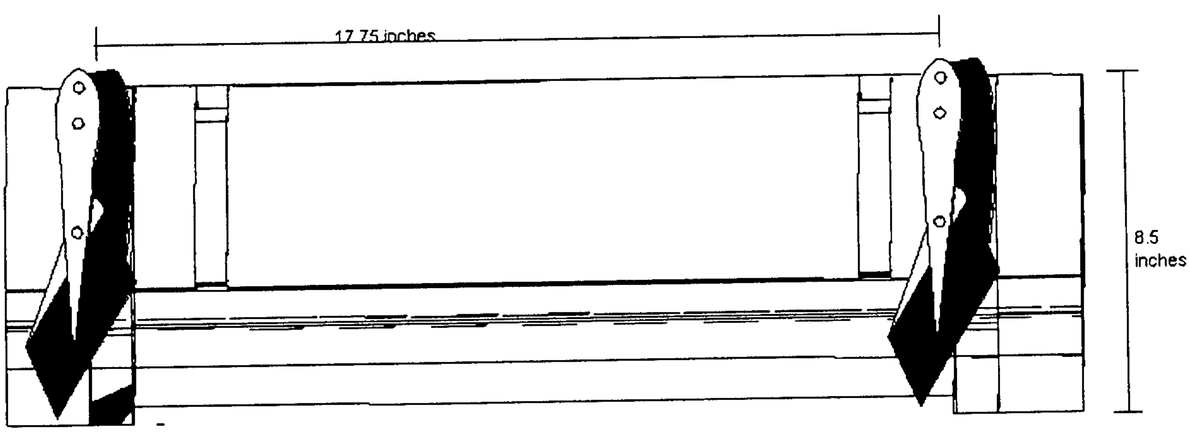




5.8.5 Tail



5.8.5(a) Tail Top View



6.0 Manufacturing Plan

6.1 Manufacturing Process

Choosing a manufacturing process and developing a coherent plan is a necessary part of the design process. Having a coherent plan lowers the cost and time required to produce or assemble a part for manufacture. With a shared budget limited by grant money there was little room to manufacture parts multiple times if they did not meet specifications. For this phase of the design process the basic manufacturing process used was that proposed by Mr. Clive Dym and Mr. Patrick Little in their book Engineering Design: A Project-Based Introduction. This six-step process was:

- Estimate the costs of manufacture for each design alternative
- Reduce the cost of components (ex. Carbon Fiber or Fiber Glass?)
- Reduce the cost of assembly (ex. Metal bolts provide strength but cost weight)
- Reduce the cost of supporting production
- Consider effect of designing for manufacture on other objectives (ex. Lower performance for easier manufacture
- Review and Revise the design

The final method for building each of the aircraft's major components is as follows.

6.1.1. Wing. For the wing, a foam core is used for shape. Over the core, carbon fiber is applied which gives the necessary strength to lift our airplane. To interface with the fuselage, three carbon fiber rods run through the inner portion of the wing and connect through three brackets located on the spar of the fuselage. The control surfaces consist of two ailerons, which are also carbon fiber-covered. The wing has monokote over the outside to ensure a smooth surface.

6.1.4 Fuselage. The plan for manufacturing the fuselage was based on simplicity and weight reduction. The two aluminum spares joined by 90-degree angled aluminum ribs for the bottom and a standard balsa rib top proved to be the best way to reduce the weight and decrease any thrust differentials created by having two engines. We used ¾ inch 90-degree angled aluminum for the bottom ribs and bend them to an elliptical shape. The aluminum spares and crossbars give us enough strength to withstand any landings. The balsa ribs on top are be cut in a similar shape to the bottom and connected by a ½ inch thick balsa spare running through the middle of the ribs. The softballs are held in place with common foam board, in which we made cutouts for the softballs to lay nicely in the fuselage. The best way to fasten the balsa ribs to the aluminum frame was to have them hinge on an aluminum rod on one side and free on the other side, fastened with velcro. This provides us with easy access to the softballs. Our fuselage will provide tremendous strength and easy connections that make the design simple and lightweight.

6.1.2. Tail / Control Surfaces. Construction began with cutting foam core for the vertical and horizontal stabilizer and then forming that foam around the main spars of the aircraft which made up the fuselage. Carbon fiber was then added to the tail in order to provide the strength necessary to support the control surfaces. The rudder and elevator were constructed from a balsa frame and then attached to the tail.

6.1.3 Landing Gear. To reduce the weight of our landing gear to a bare minimum, we used aircraft-grade aluminum alloy 6061-T6 for the main components of the rear landing gear. Everything but the wheels were machined to get the exact dimensions we wanted rather than opting for a pre-made set. The individual parts were welded together to maximize strength while minimizing weight. We used high-quality, pneumatic tires with low-friction bearings to minimize friction and allow us to customize the amount of stiffness in the tires.

For the front gear, we again used a pneumatic tire to reduce friction and absorb the shock from landing, coupled with a standard spring-assembly to allow strength in both the compression and shear forces imposed on the gear during landing.

6.2 Manufacture Timeline

In order to organize the building process, a timeline was developed to which all manufacturing teams were held accountable. Care was taken in development of the plan to ensure that components required to be used as reference for others were constructed first. The planned and actual building timeline is shown in Figure 6.1 on the following page.

6.3 Skill Assessment

Skill level for manufacture was an important consideration throughout our design process. One of our figures of merit for consideration was simplicity of manufacture. This had an impact on not selecting some more efficient designs because of the amount of skill needed and the propensity for mistakes that comes with complicated construction. Often a task needing to be accomplished was far beyond the skills of any member of the team. For these tasks we turned to department technical advisors for assistance in metal working, welding and complicated wood working. For our skill analysis we conducted a simple analysis of the task, skill level involved, whether it was necessary and whether assistance would be needed. This analysis is illustrated by table 6.1.

Task	Skill Level	Necessary	Action
Foam Cutting	Medium	Y	None
Welding	High	Y	Need Assistance
Wiring	High	Y	None
Wood Cutting	Medium	Y	None
Metal Shaping	High	Y	Need Assistance
Wing Attachments	Medium	Y	Need Assistance
Coating	High	Y	Need Assistance
Spars	Medium	Y	Need Assistance
Movable Motor Mounts	High	N	Simplify
Battery	High	Y	None
Steerable Nose Gear	High	N	Need Assistance
Separate Flaps Aileron	Medium	N	Eliminate

Table 6.1 Skill Matrix showing the relative skill necessary for each building technique utilized in the aircraft manufacture and actions required to simplify if necessary.

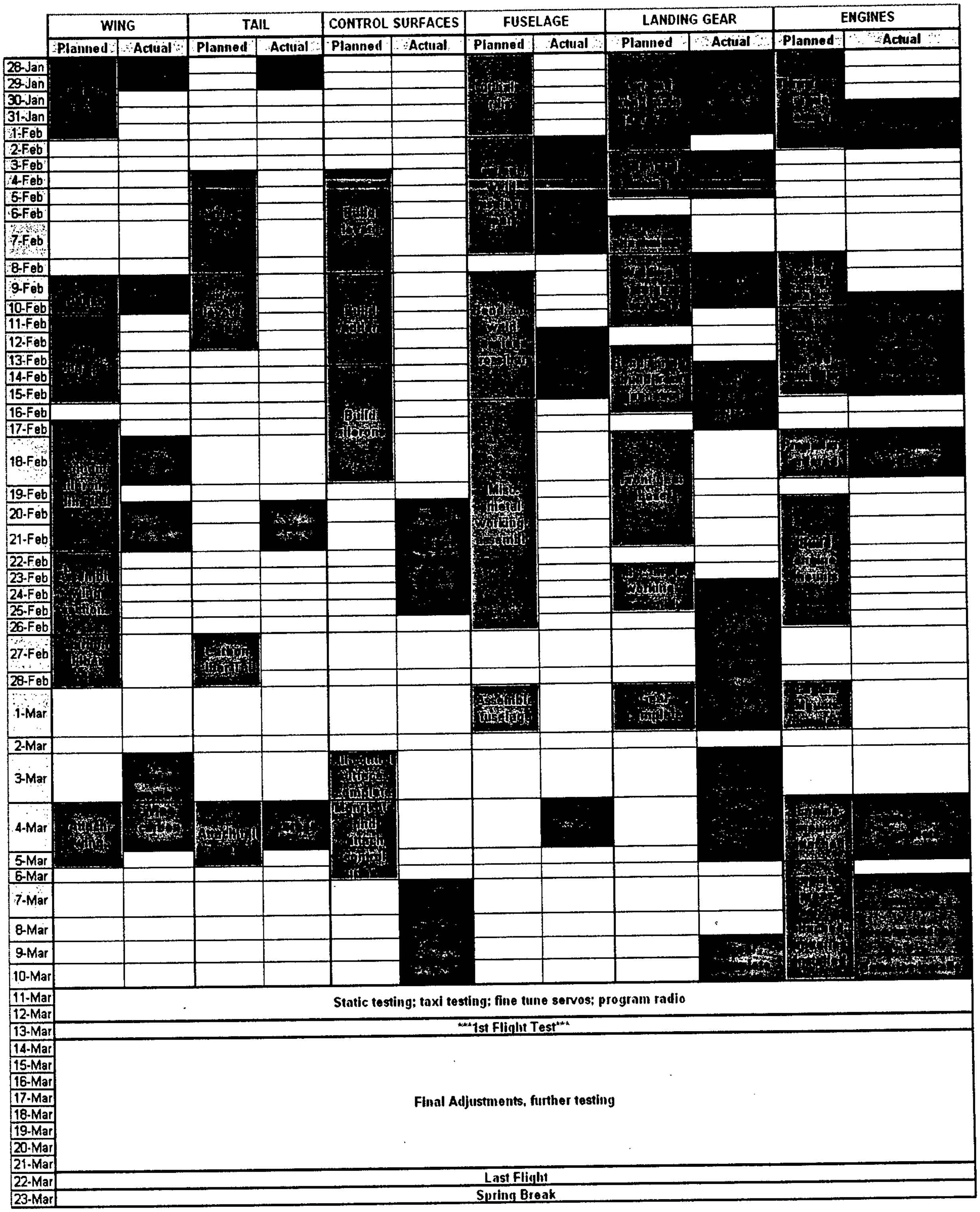


Figure 6.1: Building timeline, an in-depth summary of the manufacturing schedule. The originally planned events are shown in blue and the actual events are shown in red. The plan for aircraft testing and refinement is shown in orange.

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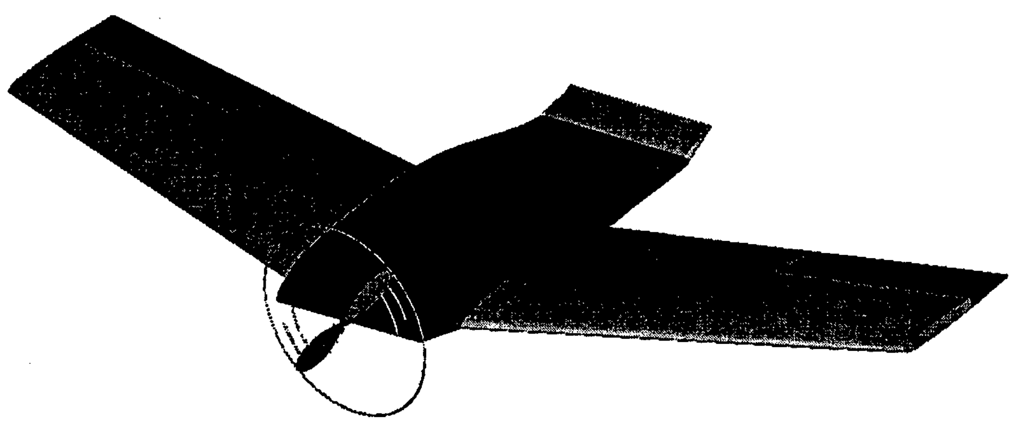
UNIVERSITY OF CALIFORNIA, SAN DIEGO

Proposal Phase Design Report

Submitted March 12, 2002



TLAR 3.5

aka "Brick"

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1 Executive Summary

Flying wings are theoretically extremely efficient aircraft. All an aircraft needs is lift, provided by the wing, and thrust provided by an engine mounted to the wing. All other components that are typically associated with an aircraft, such as a fuselage, tail and nacelles, add weight and drag. Flying wings are lighter, stronger and less expensive airplanes. Less structural interfaces exist, considerably less material and labor is needed to construct them and, with proper analysis, a very strong structure is produced. However, this ideal aircraft precedes itself. Careful design and rigorous analysis must be performed to design a flight worthy flying wing. Stability issues dominate the design of a flying wing, which does not have the luxury of long moment arms to typical or "conventional" pitch and yaw control surfaces. The payload requirements for this competition require that TLAR 3.5, not be a true flying wing, but rather a blended wing-body.

In order to meet these requirements, the team first established a set of design goals that the final aircraft configuration should meet. To maximize the score, it was decided that the design of the aircraft be light and fast, maximizing RAC and flight scores through time optimization. Ideally, the aircraft would complete mission very quickly, while meeting the functional requirements of the design as specified by the rules stated above.

Experience gained from the previous year's entries, "TLAR I" and "TLAR II", were beneficial in designing the new aircraft. This year, the team began the project with a better overall understanding of the design process, materials, design trade-offs and time constraints. This experience and knowledge, coupled with the Figures of Merit (FOMs) and design parameters, enabled the team to work more efficiently and effectively.

1.1 Conceptual Design Phase

The conceptual design phase is intended to generate basic properties of the aircraft based upon mission requirements. The team analyzed various design parameters such as wing shape and fuselage type to eliminate those that did not offer reasonable chances of high flight scores. The basic design parameters were then combined into a total of 12 working configurations. The component interactions of these configurations were then investigated to determine the best possible arrangement for further study in the Preliminary and Detailed Design Phases.

1.1.1 Alternative Designs

The design parameters investigated for the initial set of concepts included all those that could potentially result in a winning flight score. The wing variations included rectangular, swept, and flying wing. The fuselage concepts were torpedo, lifting body, and flying wing. Empennage concepts were limited to t-tail and no tail. The concepts were rated by predicting weight, RAC, and three general FOMs: Manufacturability, Transportation, and Handling. The FOM's were designed to eliminate those concepts

that were clearly problematic without having to perform a full aerodynamic analysis on each of the concepts. Handling was predicted using general flight characteristics of similarly built general aviation craft and previous competition aircraft. Manufacturability was determined from previous team experience with TLAR I and TLAR II as was the transportation FOM. When each concept was rated, the final general configuration was chosen. It was determined that a blended wing aircraft would yield the lowest RAC and highest flight scores while still being within the team's ability to construct.

1.1.2 Design Tools

The design tools used were mainly spreadsheets and codes left from previous years in competition. They were modified to account for the new rules and the new types of aircraft configurations being considered. Hand calculations based on general aviation performance equations were also used to estimate parameter interactions.

1.2 Preliminary Design Phase

The goal of the Preliminary Design Phase was general sizing of the best concept generated from the Conceptual Design Phase. Specific aerodynamic properties of various components were introduced as variables in the evolution of the aircraft. More accurate design tools were used to create a better analytical model of the final design concept.

1.2.1 Design Variations

The team examined several variations to each major aircraft component in order to select the best possible component specific configuration. The major components included the wings and the fuselage since it was determined in the Conceptual Design Phase that there would be no empennage. Payload size was also determined in the Preliminary Design Phase.

The payload specifications had to be determined so the rest of the aircraft could be tailored to it. The two choices for payload configuration were light with 12 balls and heavy with 24 balls. The heavy payload depended on softball count for high scores whereas the light payload depended on low flight times and lower RAC for high overall scores. The light payload was chosen as the best possible configuration.

Fuselage variations were influenced by the payload specifications. The lifting body of the fuselage had to be an airfoil shape, and thus the size of the fuselage was dictated by the thickness of the airfoil. Pitch was also considered due to lack of an empennage. It was determined that a reflexed airfoil would provide the necessary positive pitch. The airfoils selected for analysis were the MH60, Roncz/Marske7, S5010, and the E186. The Roncz/Marske7 was selected due to its thickness and low drag. The size of the fuselage was determined to be 16" to accommodate the payload.

The wing variations included span/chord lengths, geometric/aerodynamic twist, and sweep angle. Span was iterated from 80 to 120 inches and chord from 10 to 25 inches for root and tip airfoils. The pitch of the wing was determined to be negative to balance the fuselage moment. Therefore

aerodynamic twist was investigated by researching various high lift airfoils with negative moment coefficients. The tip airfoil was selected to be symmetrical or nearly symmetrical to provide good control. The final results indicated a span of 96" (including 16" fuselage), a root chord of 20" (E211), and tip chord of 14" (Ames-A02).

Landing gear options included composite and steel piano wire struts. The steel struts were selected due to their ease of construction and low price. In previous years they have been shown to be effective in absorbing landing loads.

The motor was determined to be the Graupner Ultra 3300/7 from TLAR II. The motor has ample power for TLAR 3.5, which was estimated to weigh less than that of TLAR II. Propeller specifications were varied from 16"x12" to 20"x20" to account for top speed and static thrust. An 18"x12" propeller was estimated to have the best performance.

1.3 Detailed Design Phase

The Detail Design Phase is intended to establish the final sizing of all aircraft components and to calculate the predicted aircraft performance. The final analysis includes aerodynamic calculations as well as structural calculations. All components were balanced to achieve maximum stability for the maximum flight scores.

1.3.1 Design Tools

The Wing Analysis program by Hanley Innovations was used to iterate the final sizings from the Preliminary Design Phase. It generates aerodynamic coefficients for wings and the lifting body fuselage.

Structural calculations were done by hand and using the NASTRAN finite element program.

1.3.2 Final Results

The aircraft components were iterated through the Wing Analysis program and hand calculations until the optimum configuration had been reached. The fuselage consisted of the Roncz airfoil with fairing between the side panels and the wings to increase overall lift/drag efficiency. The wings incorporated twist and dihedral to prevent stall and side-slip effects.

The final MEW was estimated to be 9.1 lbs. The estimated RAC is 7.995. With a top speed of approximately 60mph and a 12 softball payload, the projected final flight score is 3.717. Assuming a paper score of 86, the total final score is 120.

2 Management Summary

The 2002 UCSD Design/Build/Fly Project team consists of eleven undergraduate students from various engineering disciplines. At the first project meeting it was decided that the most efficient way to design and build the plane would be to group members together with assigned areas to work on. Each member would work in a group on one aspect of the plane, when complete move on to the next task. At times it was necessary to go back and change designs. The project manager, Josh Adams, assigned each member into an area of the project of which the member found interesting. The following focuses are listed in Table 1. The amount that each participated is represented by a number from 1 to 5, 5 being a high level of participation. A blank means no involvement.

	Area of the Project that Team Member Participated In									
Team Member	Wings	Fuselage	Landing Gear	Propulsion System	Power Management	Systems Architecture	Drawing Package	Finite Element Analysis	Manufacturing	Fundraising
Josh Adams	5	4	2	4	5	2		5	4	2
Brooke Mosley		2			2	4			5	
Steve Wong		1					5		3	
Aaron Pebley		1		2					2	
Albert Lin		1				4			2	
Guy Watanabe	5	5	5	5		2		3	5	3
Matthew Napoli		2	3				3		5	
Brian Berg		2	3						5	
Jillian Allan		1	1			4			5	5
Ceazar Javallana		1							2	3
Hamarz Argafar		2	2	2	5				5	

Table 1 - Project Participation by each member

Weekly meetings were set where the groups discussed their design ideas, chose the most optimal designs, then reported it to the entire team. The team then reviewed the design and either incorporated it or discussed improvements. Often there were problems with component interactions, where it was necessary for multiple groups to work cooperatively. The end of each meeting was designated as the time to combine each of the group's ideas into one practical plane.

The specific team member profiles are shown in the table below. This table lists some of the important skills that each member possesses; however, this table does not display the full talents and

knowledge that the team possesses because aircraft design and manufacturing spans so many disciplines.

	Major	Year	Pro-Engineer	AutoCAD	SolidWorks	FEMAP/Nastran	Technical Writing	Machining
Josh Adams	ME	SR	X	X		X	X	X
Brooke Mosley	AE	FR		X				X
Steve Wong	ME	SR	X	X	X			
Aaron Pebley	AE	SR			X			
Albert Lin	AE	JR		X				X
Guy Watanabe	AE	SR	X	X		X	X	X
Matthew Napoli	AE	FR					X	
Brian Berg	AE	FR		X			X	
Jillian Allan	BE	JR			X			X
Ceazar Javallana	AE	JR	X	X				X
Hamarz Argafar	AE	FR		X			X	X

Table 2 - Highlights of Member Skills

2.1 Task Scheduling

In early October, the group decided upon a schedule of completion dates. Each subgroup was expected to complete tasks by a certain deadline. The chart below (Figure 3) depicts the planned and actual dates of completion (DOC) of each major event. Problems that were encountered completing these tasks were quickly resolved through teamwork and subgroup collaboration. The subgroup dependencies were as follows

Assembly of Design Team: The returning DBF members from 2001 met two weeks before the start of school to decide how many new members the project would need to compete in this years competition. A meeting was held the first week of classes, and the team began to form. The team eventually solidified with 11 members.

Notice of Intent to Compete: This notice was sent by the project manager Josh Adams, on October 19, 2001.

Obtain Funding: While our search for project funding has been continuous over the past 11 months, our saving grace has come in the form of grants from General Atomics (Aeronautical Systems) and Jacobs School of Engineering (UCSD). Generous component and material donations were obtained from Hi-Tech RCD, Diversity Model Aircraft, Corland Co. and San Diego Silent Electric Flyers.

Conceptual and Preliminary Design FOMs: Conceptual FOMs were applied in the conceptual design phases and preliminary FOMs designs in order to either eliminate or accept designs.

Conceptual Design: Having found designs that met all of our goals, a final conceptual design could be assembled.

Preliminary Design: The conceptual design having been finalized, the design could be scrutinized and improved, and initial sizing could be made.

Detailed Design: The preliminary design was optimized to ensure the best flight characteristics and mission scores possible given the design parameters.

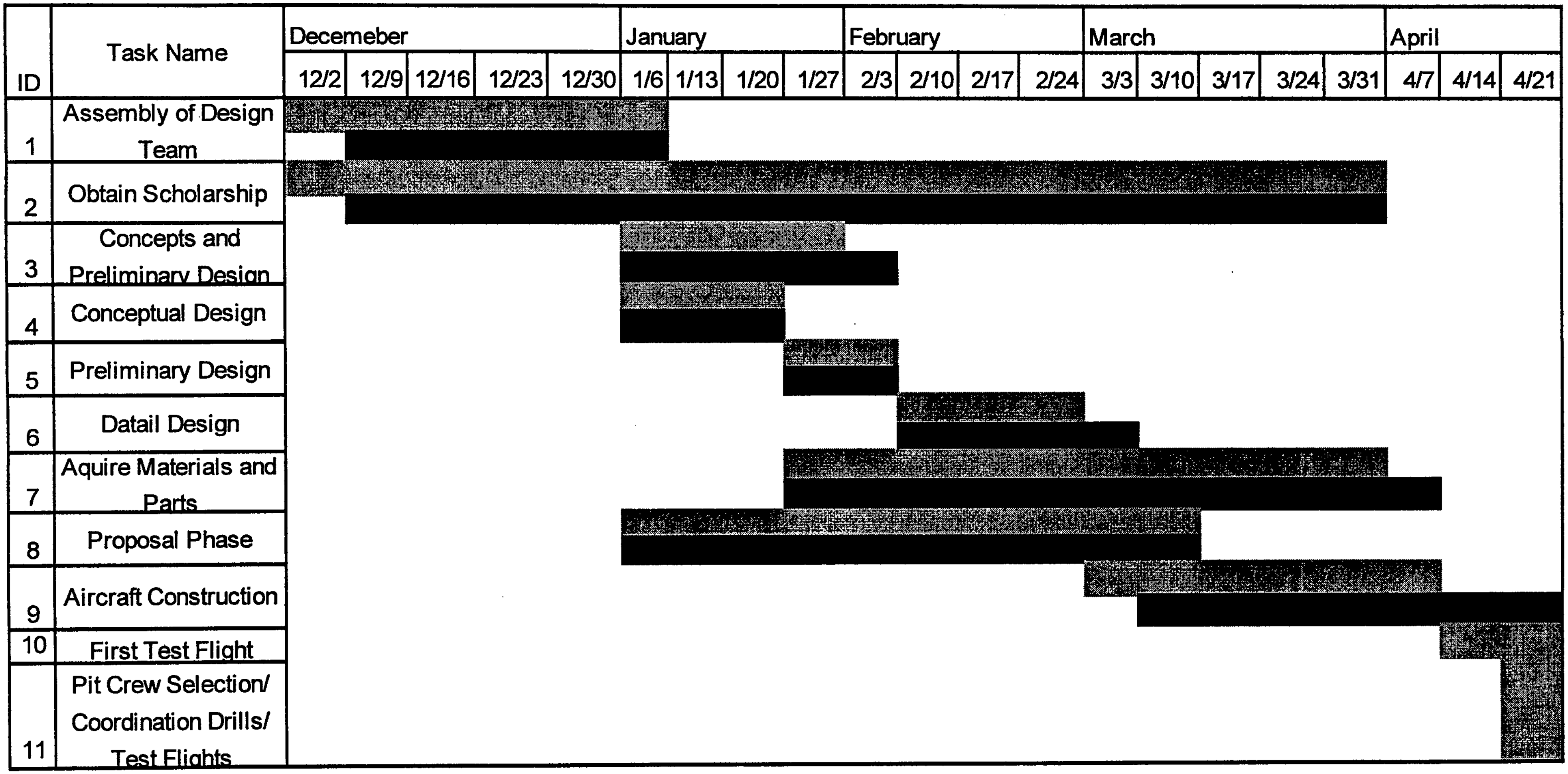
Acquire Materials and Parts: This was an ongoing task. Materials were purchased and replaced as needed. As structural designs and system architecture were finalized, appropriate parts, such as servos, were ordered.

Proposal Phase: Having a large part of the manufacturing complete and all of the performance and mission data calculated, the report was able to be written for submission to competition judges.

Aircraft Construction: The longest and most important process of the entire project, having started in December, will commence with the test flights of our plane in late March.

Test Flight(s): Having completed construction, test flights are vital to ensure structural integrity, good component interaction, performance, and most importantly serve as practice for the pilot.

Pit Crew Selection and Coordination Drills: Since the competition will be timed, having a good "Pit Crew" that is coordinated in thoughts and procedures in different scenarios to unload our cargo will be imperative to scoring well and winning the contest.



*The gray area represents the desired schedule, while the blue represents the actual.

Table 3 - Project Milestone Chart

3 Conceptual Design Phase

In the Conceptual Design Phase the team brainstormed the most basic aircraft design parameters such as wing planform and fuselage shape. The goal of this phase was to combine the best parameters into several design configurations to be analyzed in more detail in the Preliminary Design Phase. The tools used to evaluate various components were developed in spreadsheets and pencil/paper calculations, as well as from the team's previous experiences with the competition.

3.1 DESIGN PARAMETERS

3.1.1 Wing Planform

The wing planform is a critical design parameter due to the fact that it has a significant effect on the performance of the aircraft in flight. The team desired a planform that performed well at low speeds and gave the optimal control. Towards those goals, it was determined that the primary concerns in the conceptual stage for the wing planform were good low-speed lift/drag characteristics and construction feasibility. The options considered were elliptical, rectangular, swept rectangular, and delta wings.

It was found through research of the various wing types that the delta wing would only reach full potential at speeds greatly exceeding those of the competition. Therefore it was not considered further.

The elliptical wing has better theoretical performance than a rectangular wing due to the fact it creates an elliptical lift distribution. This decreases the end effects of the wing and allows for better lift/drag ratios. However, as in previous years, the problems associated with constructing a precise elliptical shape outweighed the benefits of performance and it was eliminated as a possibility.

Rectangular wings do not perform as well as elliptical wings because of the vortices created at the wing tips. This results in higher drag and reduced efficiency. However, unlike elliptical wings, rectangular wings are relatively easy to construct with precision. Rectangular wings were therefore advanced for further analysis.

3.1.2 Wing Configuration

The choice of configuration has perhaps the largest influence on the development of the aircraft than any other component. The remainder of the aircraft is tailored to the flight characteristics that the wing defines. The goals set in this part were good handling properties, lift/drag efficiency, and structural feasibility. The configurations analyzed in the initial stages were monowing and bi-wing. Winglets were also considered for the monowing.

A bi-wing aircraft can be beneficial to structural integrity and transportation. Shorter wings reduce the stress seen by the spar(s) and are much easier to transport than a longer mono-wing. However, a bi-wing aircraft is less efficient in terms of lift per unit span. Interactions between the top and bottom

sections interfere with the lift of the bottom wing, thereby reducing lift across the entire span. A bi-wing would require more wing area to lift the same load as a monowing.

A monowing with winglets has a better lift distribution than a monowing without winglets. This is due to the endplate effect preventing vortices at the wing tips due to pressure differences. The disadvantage to winglets is that they increase drag and require structural reinforcement to the wing tip and the spar. Neither concept had a clear advantage so both were advanced for further study.

3.1.3 Fuselage Configuration

The fuselage is critical part of the aircraft structure as it functions to connect the wings, control surfaces, and landing gear as well as to absorb/transfer the loads created by those components. The fuselage also houses the motor, payload, avionics, and all other important aircraft components. The contest rules dictating that the payload be arranged in the horizontal plane served to reduce the number of practical design possibilities. The concepts selected for analysis were the wide lifting body, flying wing, and narrow "torpedo".

The "torpedo" concept had the payload in two or three rows extending the majority of the fuselage. This would allow a very narrow streamlined body with relatively low drag. However, this year a front/rear loading fuselage would hamper pit stop crews since each softball must be hand loaded. While the torpedo design does not have any overwhelming advantages, it does not have any great disadvantages. It is incorporated into further concepts due to the experience of the team with the design and the success of previous years with it.

The wide lifting body takes advantage of the softball arrangement rules by using the large area required by the payload to create lift. The airfoil shape of the fuselage creates the lift and the payload can be stored in the thickest section. This fuselage allows for a slightly smaller wing area and low drag. Fuselage size is a concern as the airfoil shape must be sufficiently thick to hold the payload without creating an excessively long body. Access to the payload is also an issue but the benefits to performance make the concept worth looking into.

The flying wing body incorporates the fuselage and the wings into one continuous structure. The greatest benefit to this would be the higher efficiency of the entire aircraft. Since the entire body acts as a wing the lift per unit area is increased and the drag normally induced by a fuselage is eliminated. However, such a structure requires very large wings to connect to the fuselage. This is problematic both for construction and transportation reasons.

3.1.4 Empennage

The empennage, with elevators and rudders, allows yaw and pitch control of the aircraft. It is also a stall safeguard either by stalling after the wing to maintain control, or stalling before the wing to prevent

the wing from stalling. A number of variations of empennage exist and can provide differing levels of control and efficiency. The options considered were T-Tail, conventional tail, canard, and no empennage (flying wing).

The conventional tail has the horizontal stabilizer under the vertical stabilizer. This design is easy to construct as the stabilizers may be anchored to a boom or on the end of the fuselage without much reinforcement since each only supports its own weight and lift forces. There are no major disadvantages to this design except that the empennage must be sufficiently far way from the fuselage to prevent the fuselage wake from interfering with the elevator. Despite being a viable design, the conventional tail design was discarded in favor of the T-tail.

The T-tail design places the horizontal stabilizer on top of the vertical stabilizer. This not only eliminates possible interference from the fuselage, but also creates an end plate effect for the vertical stabilizer. This increases the overall efficiency of the empennage as tip vortices are eliminated. The disadvantages are that the vertical stabilizer must be reinforced to support the horizontal stabilizer. In past years, this has been accomplished to great success and this concept was chosen as superior to the conventional tail.

A canard places the vertical stabilizers ahead of the main wings and the aircraft center of gravity. Typically the canard has a higher aspect ratio and is designed to stall before the main wings. This makes it much more difficult to stall the main wings as the aircraft will not be able to achieve dangerous angles of attack. The disadvantages include lack of a vertical stabilizer and a short moment arm. Without a vertical stabilizer, there is no yaw control. In order to obtain sufficient pitch control, the canard must either have a large moment arm, or be increased in size over a more conventional tail. The canard still represents a feasible design concept as it could lower the RAC and allow for a smaller structure than other designs.

The flying wing has no empennage. It uses reflexed airfoils and wing twist/sweep for pitch control and either end plate rudders for yaw control or no direct yaw control at all. The benefit to structural weight and RAC is significant. Weight is saved by excluding the boom and control surfaces, allowing for fewer batteries and greater speed. With no control surfaces, the MFHR of the RAC is noticeably reduced. This design provided a clear advantage and was advanced to the next stage.

3.1.5 Propulsion

The propulsion system is the component that allows the aircraft to sustain powered flight. It is required to achieve the 200 ft take off limit and propel the aircraft around the course at maximum speed. The characteristics of the propulsion system of most interest to the project are static thrust, top speed, and battery power management. These aspects must be tuned in such a way as to allow for maximum flight score and lowest RAC. The most practical concepts discussed were dual motors, single motor, and single motor geared along with several battery configurations. The motors used were the Graupner Ultra 3300/7 used last year's competition.

The battery weight restriction of 5 lbs. limited the number of battery cells to 38, which amounted to approximately 40 volts. For standard radio controlled propulsion systems, 40 volts exceeded the rated power of many motor possibilities. While a motor's power is related to the input voltage, they can normally withstand higher voltages. Problems experienced in last year's competition showed that this was not a desirable arrangement. The team used the previous years 38 cells as a working estimate to be refined in later design stages.

Concerns for power as well as those regarding additional weight of this year's design did necessitate that a dual-motor configuration also be proposed. The dual-motor configuration would place the engines on the wings, supported by the spar. Last year's competition demonstrated that such a configuration provides excellent static thrust and high top speeds. Dual motors also allow for smaller propellers, which in turn aid in shorter landing gear. However, this year the RAC cost is extremely high. It was decided that the high performance was not adequate to justify the increased RAC.

The greatest benefit of a single motor is the minimization of the RAC. In order to generate the required thrust, however, a single non-geared motor requires a larger propeller than does the dual-motor configuration. In the previous year, 14 in. propellers were used to generate a total of 9 lbs of thrust. Using this figure as a baseline, the team estimated that the new propeller would fall into the 17-19 in range to achieve similar results. The larger diameter was still viewed as practical and this concept was advanced.

The use of a gearbox can greatly enhance propeller speeds. It is possible to gear a motor up to achieve higher rpm with a larger propeller to reach higher top speeds at the expense of static thrust. This concept would make it more difficult to make the 200 ft take off distance but would potentially shorten lap times considerably. In previous years the power plants easily managed the take off distance limit by more than 50 ft. Therefore the team considered it possible to advance the geared motor.

3.1.6 Landing Gear

The landing gear determines the ground handling characteristics of the aircraft and adds drag while in flight. A preferable landing gear design has good ground stability and low drag in flight. The concepts included for analysis were tail-dragger, tricycle, and quadracycle.

The tail-dragger design places the main wheels slightly ahead of the CG and the rear wheel farther back near the empennage. The concept was used in previous years and has proven reliable under light to moderate conditions. However, under high wind conditions, ground handling is poor due to the wind moving the empennage, and thus the rear wheel. As relatively high winds are expected at the competition site, it was decided that the tail-dragger would not be practical.

The tricycle landing gear has the two main wheels slightly behind the CG and the third wheel ahead of the CG near the nose. This arrangement has improved handling characteristics over the tail-dragger. The difficulty lies in the fact that if the propeller is in the nose of the aircraft, the nose landing

gear must be long enough to allow clearance. With larger propellers, the gear must be longer and stronger. Aerodynamically, the longer the landing gear the more drag is created. This design was advanced due to the very desirable ground handling.

A quadracycle was initially proposed due to the wide rectangular fuselage concepts. There are four wheels placed around the CG. This design does not have any clear advantages over the tricycle landing gear. It does in fact create more drag and requires more structural reinforcement while still facing the same problems as the other designs and was discarded.

3.1.7 Aircraft Configurations

The design concepts remaining to this point in the conceptual design phase were incorporated into 6 configurations with the option of adding winglets, for a total of 12 variations.

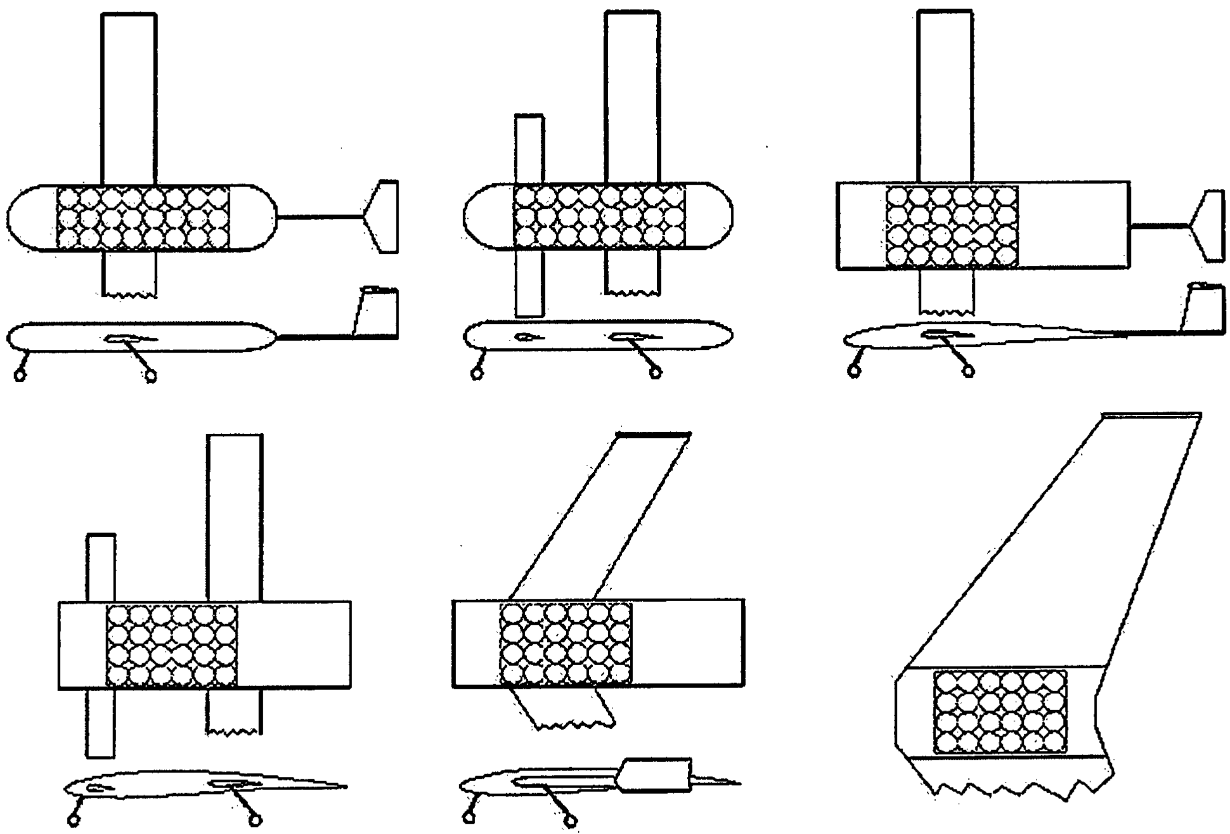


Figure 1 - Design Possibilities (Winglet designs not shown)

Design #	Planform	Winglets	Fuselage	Empennage	Projected Empty Weight (lbs) (X3)	# Control Surfaces (X1)	Handling (X1)	Manufacturability (X2)	Ease of Transportation (X1)	Projected RAC (X3)	Advanced
1	Rectangular	N	Torpedo	T-Tail	15.0	4	3	2	1	9.6	N
2	Rectangular	Y	Torpedo	T-Tail	15.5	4	3	2	1	9.8	N
3	Rectangular	N	Torpedo	Canard	13.0	4	1	1	3	9.4	N
4	Rectangular	Y	Torpedo	Canard	13.5	4	1	1	3	9.6	N
5	Rectangular	N	Lifting Body	T-Tail	15.0	4	3	2	1	9.2	N
6	Rectangular	Y	Lifting Body	T-Tail	15.5	4	3	2	1	9.4	N
7	Rectangular	N	Lifting Body	Canard	13.5	4	1	1	3	9.1	N
8	Rectangular	Y	Lifting Body	Canard	14.0	4	1	1	3	9.3	N
9	**Swept**	**N**	**Lifting Body**	**none**	**12.0**	**2**	**1**	**2**	**3**	**8.6**	**Y**
10	Swept	Y	Lifting Body	none	12.5	2	1	2	3	8.8	N
11	Flying wing	N	Flying wing	none	11.5	2	1	0	1	8.8	N
12	Flying wing	Y	Flying wing	none	12.0	2	1	0	1	9.0	N

Table 4 - CDP Possible Designs and Figures of Merit

3.2 Figures of Merit

The figures of merit quantify the most important design aspects of the aircraft. They are used to optimize the aircraft design and achieve the highest possible flight score. The FOM's used by the team to rate each concept were weight, handling characteristics, ease of transportation, and ease of construction.

3.2.1 Projected Empty Weight

Weight calculations were based on average component densities estimated from previous competitions. As weight directly impacts speed and power consumption, it was considered to be one of the primary FOM's used. Since all concepts generated in the initial stage of the Conceptual Design Phase utilize the same power plant and batteries, it was possible for the team to compare power usage using only weight and drag estimates (i.e. it was not necessary to calculate a flight plan for concept elimination purposes).

3.2.2 Handling Characteristics

This FOM rates basic stability pitch/yaw control, turn radius, and ground handling. The flight performance of the aircraft is determined by the ability of the pilot to control the aircraft. With good pitch/yaw control and tight turn radii, the pilot can lower the flight times by several valuable seconds. With

the emphasis on sortie time in the competition, it was necessary to closely examine the general aerodynamics of each configuration.

3.2.3 Ease of Transportation

A well-designed aircraft is useless unless the team can get it to the competition intact. Therefore, transportation was a major issue as the team operated on a limited budget of time and money. In previous years, the aircraft was designed in part with shipment in mind and no less emphasis was placed on it this year.

3.2.4 Ease of Construction

The final FOM rates the ability of the team to construct the aircraft from the design on paper. It is important to incorporate this FOM in the design process so that the final product matches the calculations. Sacrifices must often be made on performance due to constraints on tools and materials.

3.3 Conceptual Summary

In the Conceptual Design Phase the team reviewed the most basic design parameters. Planform, fuselage and wing configuration, empennage, propulsion, and landing gear concepts were combined into a set of theoretically viable aircraft configurations. Each of these configurations was rated according to the RAC and FOM's geared towards maximizing flight scores. After all aspects had been rated, the design estimated to be the most efficient and most practical was chosen.

Concepts 1 & 2: The first two concepts were generated from TLAR II in the 2001 competition. The monocoque fuselage provides good torsional rigidity and relatively low drag when streamlined. The T-tail empennage allows for excellent pitch and yaw control, resulting in good flight handling. This concept has proven to yield top flight scores in the past.

With a weight estimate of 15 lbs (15.5 with winglets), the initial two design concepts ranked among the heaviest of the group. This would significantly increase lap times and therefore lower flight scores. It was predicted that any benefits gained from the good handling characteristics would have been eliminated by the relatively high RAC. The designs were therefore eliminated as possibilities.

Concepts 3 & 4: The torpedo-canard configuration has the benefit of good stall characteristics. Weight reduction on the order of 2 lbs. accounting for the lack of empennage support allows for faster lap times over concepts 1 & 2. The shorter projected dimensions would make transportation easier than other concepts.

The estimated RAC was not as low as originally predicted. This was due to the lengthening of the fuselage to limit the canard size. The team predicted lower handing capabilities than the more conventional designs due to the lack of direct yaw control and sensitivity of the canard to stall.

Concepts 5 & 6: The lifting body concept was conceived to make maximum use of the area housing the payload. The lift produced by the fuselage allows for smaller wings, and therefore a lower RAC. The payload is limited to the thickest parts of the airfoil, making it more accessible to ground crews.

The lifting body-T-tail designs share much in common with the torpedo-T-tail configuration. The RAC and weight savings were not as significant as originally expected. The handling characteristics are virtually identical and there was no obvious advantage in terms of flight scores.

Concepts 7 & 8: The lifting body-canard configurations are similar to the torpedo counterparts on paper. The handling and RAC are based on the same component assumptions.

While the RAC and weight values are similar to the torpedo concepts, the structural concerns are not. The torpedo designs may be lengthened or shortened as the situation dictates. The lifting body, however, is constrained by the airfoil shape. It was discovered that in order to place the canard at the prerequisite distance to allow maximum efficiency, the fuselage must be enlarged significantly. The sheer size of the fuselage renders these concepts unfeasible.

Concept 10: The blended wing design resulted in the lowest RAC and the second lowest predicted weight. The efficiency of this structure allows for quick lap times at a low power usage.

The disadvantage to this structure is that it is not able to maneuver as well as the other configurations. The lack of direct yaw control raises similar problems to the canard designs. However, the considerable weight savings and the low RAC make this design the best overall configuration.

The winglet design improves the lift distribution by reducing tip vortices. This allows the wingspan to be shortened slightly. However the winglets also increase drag and require significant structural support. These factors led the winglet-blended wing to be eliminated.

Concepts 11 & 12: The flying wing design is theoretically the most efficient in terms of structural weight. The team estimated that the flying wing would have the shortest takeoff distance and one of the lowest RAC.

The difficulty in the flying wing lay in construction. The size of the wings made it difficult to make out of foam for a composite lay-up. A rib structure was ruled out due to the precision required in airfoil shape and wing twist/sweep angles. The large root chord and wingspan also increased the RAC and resulted in the design being disqualified.

Final Design Configuration: The blended wing was the final design choice based on RAC calculations and flight score predictions. With a predicted RAC of 8.6, the blended wing design had as much as a 10% advantage over other possibilities. The weight advantage is also significant. The lighter aircraft would require much less power throughout the entire sortie and could potentially fly much faster.

It was also feasible for both manufacturing and transportation. The structural components for the blended wing were very compact. This is attributed to the fact that it is not required to support an empennage. Thus not only is the empennage eliminated, but all the supporting structure is eliminated as well. Initial scale drawings predict that no structure behind the payload tray will be required.

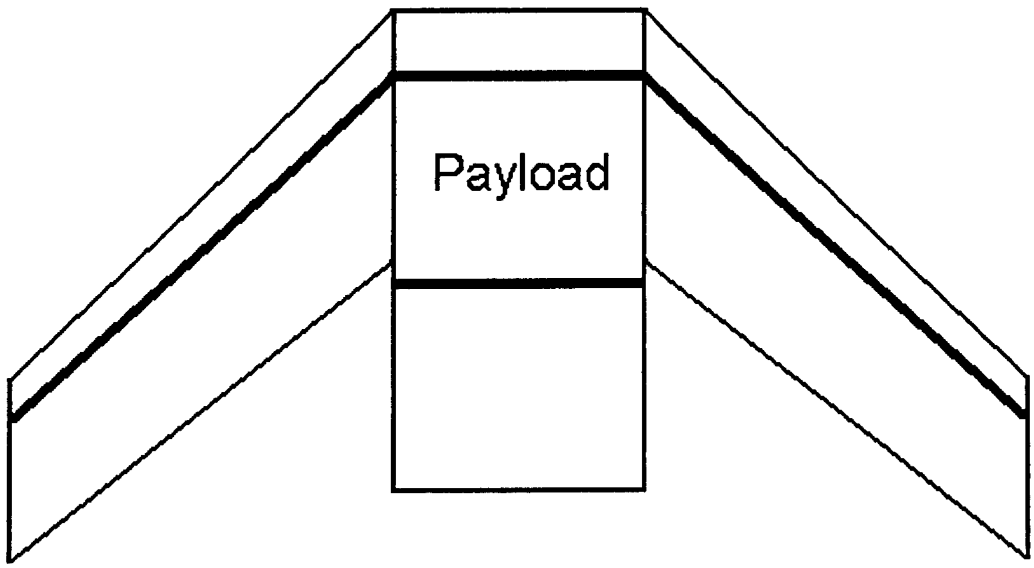


Figure 2 - Final Design Concept: Blended Wing

4 Preliminary Design Phase

As the design of the aircraft progresses, estimates for actual design specifications need to be made. The PDP of the design focuses on completing the initial calculation and estimation of general sizing, performance, and configuration with regards to the final aircraft design created in the second stage CDP.

During this process, techniques developed for the design of propeller driven general aviation aircraft were used to a large extent. Standard aerodynamic theories, such as Prandtl's lifting line method, were also implemented in team written computer algorithms as well as the Wing Analysis Program (Hanley Innovations) to analyze the aerodynamic properties of the aircraft. While these techniques are not expected to be exact, they were implemented with confidence as their accuracy was validated by TLAR I (2000) and TLAR II (2001). Overall, these proven methods provided excellent analysis results while directly exposing the design team to the fundamental principles of aircraft design.

As the CDP was highly focused on determining design parameters and FOMs to apply to the design process, so was the PDP. Here, however, the considerations that need to be made are more related to the optimization of flight performance and aircraft component interaction, than with the broader concerns examined in the previous sections. For each area of the preliminary phase, design parameters and FOMs will be applied as suit that particular area, and will be described locally.

4.1 Analytical Tools

The primary tool used in size and score estimations were spreadsheets developed over the last two years of competition. They incorporate the basic aerodynamic equations and RAC/flight score calculations. TLAR I (2000) and TLAR II (2001) were straight wing aircraft with t-tail empennages so the spreadsheets had to be adapted to the blended wing equations. These equations took into account many factors that were not present in the previous aircraft such as sweep, pitching moment, and stability margins.

A program called Wing Analysis (Hanley Innovations) was used in the iteration of various component sizes and airfoil selection once the basics had been established. It is attached to the Hanley Innovations program Visual Foil which contains the entire UIUC airfoil database and NACA specifications. The code runs a numerical analysis on the defined wing to give the theoretical performance. The program inputs included airfoils, planform dimensions, speed, and angle of incidence. The outputs included Reynolds number, drag, lift, moments, bending forces, and power usage. The team found this to be particularly useful since it allowed a greater number of variables to be introduced. It also permitted the calculation of such things as end effects that would otherwise be very time and labor intensive.

4.2 Design Parameters

4.2.1 Payload Size:

The flight score of the 2002 competition depends as much on sortie time as payload capacity. Unlike previous years, carrying less payload does not necessarily equal a lower score. The team evaluated the various possibilities and balancing between capacity and speed.

A large payload achieves a high score by having the numerator of the flight score equation dominate. With the maximum payload of 24 softballs, the numerator equals 30. The minimum payload would have a numerator of 16. Thus a small payload aircraft would have to cut the flight time approximately in half to achieve the same score. This is difficult to achieve, and the large payload aircraft has the advantage in raw flight scores.

The advantage of a small payload is the shorter payload transfer times and a lighter aircraft. The reduced weight of the aircraft impacts several areas of the overall score. A lighter aircraft can achieve higher speeds and can therefore lower flight times. A lower weight also decreases the number of batteries necessary to complete the sortie. Battery weight plays heavily into the RAC. So while the smaller payload may not be able to cut the flight time in half to match the large payload, the difference in flight score is offset by the lower RAC.

It was not possible for the team to calculate which setup had the numerical advantage due to the abundance of variables for which there was very little concrete data. It was decided based qualitative estimation that the light payload was preferable due to the many benefits. As the rules dictated that the softballs must be inline and in the horizontal plane, a 12 ball payload was selected. The 12 ball payload had the largest number of symmetrical configurations.

4.2.2 Fuselage:

Optimization of the fuselage included determining a viable support structure and the necessary dimensions for maximum payload capacity. The fuselage muse also be designed for the motor and all supporting devices. In the case of the blended wing design, aerodynamics also became an important issue with the selection of the airfoil shape. The Reynolds numbers used were approximated from the fuselage of TLAR II to be $2x10^5$.

<u>Airfoil:</u> The airfoil shape of the fuselage for a blended wing must not only provide lift, but must also be factored in pitch control. The team decided that the fuselage would be built as a reflexed airfoil to provide more stability rather than a standard airfoil to provide more lift. This was due to the fact that the fuselage would have a very short span and thus the maximum lift would not be significant compared to the wings. The fuselage could, however, still provide a large amount of pitch with the limited span.

Another primary feature evaluated in the airfoil selection process was the maximum thickness. A thick airfoil was necessary to ensure a short fuselage and a lower RAC. This criterion was applied to the

reflexed airfoils in the UIUC and NASG airfoil databases and four possibilities resulted: E186, Roncz/Marske7, S5010, and MH60. The evaluation of these airfoils is provided in the final sizing section.

Payload: The payload configuration affects the size of the fuselage and can indirectly influence pit stop times. Access is also a concern as time is critical to the overall flight score.

The consensus was that the most efficient payload configuration is the most compact arrangement possible. This would reduce the amount of support structure required and make it easier to load/unload. The best configuration to reduce fuselage length and structure was judged to be 3 rows (spanwise) and 4 columns based upon the 12 softball payload.

The options for the payload access were front loading, rear loading, and top loading. The front/rear loading concepts were used in last year's competition and worked well with speed loaders. However this year, with each ball required to be hand loaded, a front/rear loader would restrict access to the payload area and would slow the loading process considerably. A top loading configuration would make the entire payload area accessible and multiple pit crew members would be able to participate in loading. Therefore the top loader was chosen as the best configuration.

Structure: The structure of the fuselage must be stiff enough to resist aeroelasticity effects and survive the loading conditions during high G turns and landing. The top loader configuration established in the payload section complicated matters due to a disruption of shear flow around the fuselage. The possibilities investigated were semi-monocoque, box, and keelson structures.

The first design concept considered was the keelson, which is essentially a panel that runs along the length of the bottom of the fuselage. This panel would have to be quite strong and large in order to transfer and absorb loads effectively, which raised weight concerns. The shape of the fuselage also presented problems with the keelson in that it would have to be curved. Thus the keelson design concept was dismissed.

The semi-monocoque structure uses a cylindrical fuselage reinforced by bulkheads to take both torsion and bending stresses. However, it the semi-monocoque requires the skin of the fuselage to be continuous for shear flow. With the top loading configuration would only work if the loading hatch could be reintegrated with the rest of the structure to transfer loads. Reinforcing the hatch connectors would be difficult to design such that it would not interfere with the pit crews. The semi-monocoque structure was disqualified for this reason.

The box structure consists of side panels connected by several bulkheads to absorb bending stresses. The torsional strength is provided by the payload tray. When the payload tray is elevated several inches from the bottom of the fuselage, it creates a closed loop with the side panels and the skin under the tray. This structure is both compact and efficient in that the payload tray serves multiple purposes. This configuration was deemed to be the best choice overall.

Construction Technique: The most practical possibilities for constructing the payload tray and supporting structure were plywood and foam core carbon composite. It is easier to machine the correct specifications into plywood than it is to form the composite lay-up mold. The composite approach

however is lighter and stronger. It is also possible to customize the carbon lay-up to achieve the maximum strength where it is needed. Therefore the composite approach was chosen.

For the skin of the airplane, a similar composite approach was proposed. The skin would be a fiberglass or carbon lay-up over a fuselage shaped mold. The foam would be dissolved away and leave the skin to bond to the bulkheads and side panels.

4.2.3 Wings:

The wings design parameter deals with the general sizing of the wing and associated components as well as construction technique. Wing geometry was chiefly governed by necessary flight characteristics and optimized by RAC concerns while construction methods were based on available tools and skills.

Geometry: The swept planform was selected in the CDP as part of the efficient blended wing configuration. The angle of sweep is important in determining the pitch control of the aircraft as well as the longitudinal stability. Other factors such as taper and twist (aerodynamic/geometric) also contribute to the stability of the aircraft and must be balanced to achieve maximum performance. The final factor to be considered was the lift and pitch characteristics of the fuselage.

The fuselage was determined to be the reflexed Roncz/Marske7 airfoil. The Roncz/Marske7 has a relatively high positive pitch coefficient. Normally the wings of a blended-wing or flying-wing configuration are reflexed for control purposes. The team investigated the possibility of having a more conventional negative pitch wing to balance for the fuselage positive pitch. The advantage to a negative pitch wing is that airfoils with higher lift coefficients may be used, resulting in a smaller wing area. It was not certain if the fuselage would be able to balance out the wing pitch, but the benefits to RAC and construction made this configuration the most promising for the team.

The sweep, taper and twist specifications were iterated using the Wing Analysis program by Hanley Innovations and are covered in the final sizing section.

Lateral Stability: Concerns over side-slipping effects on the aircraft during banked turns were raised due experiences during the previous competition. Dihedral is used to provide stability in these situations. The wings are angled slightly up at the root. When in a banked turn, the dihedral causes the inner wing to be at less of an angle than the outer wing. The inner wing generates more lift, creating a tendency for the aircraft to roll upright. The exact angle required to provide maximum performance is difficult to calculate due to the complexities of the flow during turns. Due to increased stability concerns due to lack of an empennage, dihedral was included in the design.

Another measure taken to ensure stable flight was implementation of both geometric and aerodynamic twist in the wing to prevent stall at the ailerons. The geometric twist serves to reduce the effective angle of attack at the tip, whereas the aerodynamic twist changes the airfoil at the tip to one with better stall characteristics.

Longitudinal Stability: The longitudinal stability is normally controlled by the empennage. In a flying wing or blended wing configuration, the pitch is controlled by the sweep of the wing and the location of the control surfaces on the wing. The team could not accurately determine the dynamic performance characteristics of the design with only 2 aileron control surfaces. It was proposed that a third control surface be included on the fuselage. An elevator on the fuselage could adjust the reflex of the airfoil and thus increase or decrease the pitch without relying on the ailerons. It was decided that this approach would result in more predictable flight characteristics and that the increase in RAC was negligible.

Control Surfaces: The type and location of control surfaces were dependent, in part, upon the rest of the primary aircraft structures. The available options included flaps, ailerons, or a hybrid control surface. Achieving the necessary level of control was the principal concern, followed by the impact to the RAC and building complexity.

Flaps were considered to increase lift during takeoff in order to minimize the takeoff distance. However, initial calculations on the lifting capacity of the wing suggested that the aircraft would have no trouble lifting off in 200 ft. Additionally, the inclusion of flaps would add significantly to the RAC (+6 hrs for 2 flaps +10 hrs for 2 servos).

Spoilerons were chosen in place of ailerons to ensure control of the rolling/turning of the aircraft at low airspeeds. They are hybrid structures combining spoiler effects with aileron control surfaces. They help the flow stay attached to the wing during slow, high AOA flight, ensuring that the ailerons do not stall before the main part of the wing. They can also be used after touch down to reduce the lift of the main wing and ensure that the aircraft stays on the ground.

Construction Technique: The construction techniques determine the internal structure of the wing. The possible methods included a rib structure and a foam core composite structure. The goal was to maximize the strength to weight ratio, while also considering the implications to time and skill management.

Aircraft grade plywood is relatively cheap and lightweight. Manufacturing a wing with this material, however, is a complicated task. The wing would have to be made out of a series of ribs, each of which had to be individually assembled and shaped. The difficulty of this fails to satisfy the ease of construction FOM.

A foam core composite wing holds several advantages over the plywood wing. It was less labor-intensive and was considerably stronger while being comparable in weight. The structure and procedure were established during the development of TLAR II in the 2001 competition. Despite the fact that composite wings are somewhat more expensive than plywood and require specialized tools, the high strength to weight ratio resulted in the decision to use them in the final wing.

4.3 Final Sizing for PDP:

4.3.1 Fuselage:

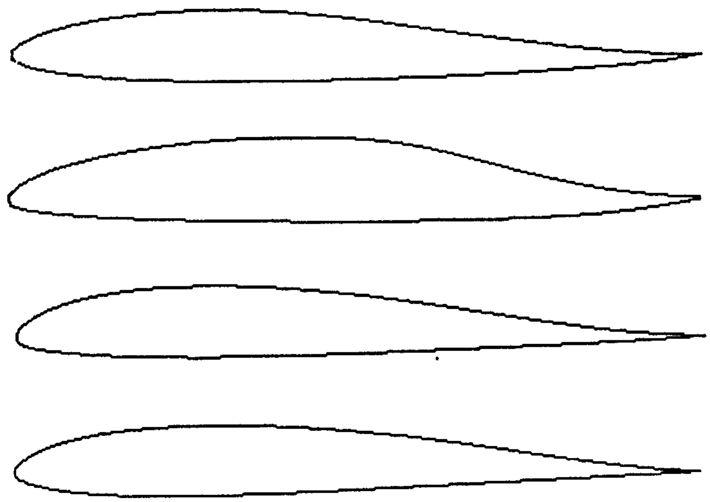


Figure 3 - Best Possible Airfoil Choices

Airfoil	Max Thickness	Max Cl	Max Cd	Cm(c/4)	Stall α	α Cl=0	Length*
E186	0.1027	0.817	0.0127	0.044	10	1.25	50"
Roncz/Marske7	**0.1205**	**0.679**	**0.0087**	**0.005**	**6**	**-0.14**	**40"**
MH60	0.1008	0.996	0.0134	0.026	10	-0.35	48"
S5010	0.098	0.843	0.0069	0.008	9	-0.52	50"

*Length was estimated as the best length required to fit 3 rows of softballs in a horizontal plane

Table 5 - Airfoil Data Calculated at Re=2,000,000

The Roncz/Marske7 was the thickest of the four choices at 12.05%. The fact that the airfoil maintains this approximate thickness along the majority of the chord length allowed for the shortest projected length. The drag was also lower by as much as 54%. The marske7 does not provide nearly as much lift as the other airfoils, however, and it stalls much earlier at 6°, giving the aircraft limited maneuverability (as compared to the 10° stall angles for the E186 and the MH60). In weighing these factors, the team chose to weight thickness and drag over the others. The low lift could easily be compensated for by the wings and the stall angle was judged to be secondary to higher speeds as the flight plan called for low time, not extreme maneuvers.

The final size of the fuselage was determined to be 40" in length, and 16" in width (see table 5). This was based on the 3x4 payload configuration in the thickest part of the Roncz/Marske7 airfoil. The CG of the payload was placed at approximately 14" from the leading edge.

With a chord length of 40" the maximum thickness of the airfoil is 4.82". A softball was measured to be approximately 3.9" in diameter. Based upon these calculations, the payload tray was placed at ¼"

above the chord line. With this configuration, the payload tray would have holes cut where the balls would sit. This allowed for more distance from the bottom of the fuselage and thus a larger torsional rigidity.

4.3.2 Wings:

Based upon the estimated dimensions of the fuselage, the moment generated around the quarter chord was calculated using the Wing Analysis program to be 130 in-lb. The wing would have to balance this moment for trim flight. They would also have to provide enough lift to balance the full weight of the plane (minus the lift of the fuselage).

The negative pitch was required to balance the fuselage, which meant that the wings could use conventional high lift airfoils to reduce size. The E214 had worked well in previous years and was analyzed along with the E212 and E211 for the root airfoil. The tip was desired to be a symmetric airfoil for control purposes. The AMES02 was selected as an arbitrary initial airfoil based upon VisualFoil data. Designs were iterated trough the Wing Analysis program varying wing span, taper ratio, root chord, sweep angle, airfoils, and twist. The final design statistics are summarized in Table 6.

	Fuselage	Wing
Speed (mph)	65	65
AOA	3	2
Span (Inches)	**16**	**80**
Sweep (Deg.)/Loc.	0 (c/4)	25 (c/4)
Tip (Inches)	**40**	**14**
Root (Inches)	**40**	**20**
MAC (Inches)	39.9794	17.1778
Tip Airfoil	**RONCZ LOW DRAG**	**NASA/AMES A-02**
Root Airfoil	**RONCZ LOW DRAG**	**E211 (10.96%)**
Planform Area (Sq.Inches)	**640**	**1360.0001**
Taper Ratio	1	0.7
Twist Angle (Degree)	0	0
Tip Reynolds Number	2,021,367	707,478
Root Reynolds Number	2,021,367	1,010,684
MAC Reynolds Number	2,020,328	868,066
Lift Coefficient	0.0176	0.158
Induced Cd	0.0002	0.0033
Profile Cd	0.0044	0.0058
Total Drag Coefficient	0.0047	0.0091
Moment Coefficient	0.069	-0.0712
Lift/Drag	3.7641	17.3445
Total Lift (Pounds)	**0.8439**	**16.1248**
Total Drag (Pounds)	**0.2242**	**0.9297**
Mom. (c/4) (Pounds-Inches)	**132.5**	**-124.9**

Table 6 - Fuselage and Wing Data From Wing Analysis Program

The data provided by the Wing Analysis program allowed preliminary sizing of the fuselage and wings to be balanced through iteration. The areas of focus were the pitching moment, total lift, and lift/drag ratio. In order to balance all the values, an angle of incidence of 1° was created between the wing and the fuselage. Due to the high Cm of the E211 airfoil, it was difficult to lower the pitching moment of the wing to match that of the fuselage. The lift was very favorable at almost exactly what the aircraft was predicted to weigh fully loaded.

In order to place the wing on the fuselage to balance the pitch, a stability margin of .1 was assumed. The exact distance of the neutral point from the C.G. was calculated by multiplying the stability margin by the mean chord of the wing. This gave a distance of 1.7". In order to position the wing so satisfy this criterion, the spar crosses through the fuselage, just ahead of the payload tray. Scale drawings have the back of the spar flush with the front bulkhead supporting the payload tray.

In order to reduce torsion stresses, a pin was designed into the root of the wing approximately 8" behind the spar. This pin would project into the fuselage and transfer the torsion into the side panels of the box structure. This design had been proven in previous competitions.

TLAR II used a 3° dihedral to provide lateral stability. This figure was reached by empirical means. This value was also used for TLAR 3.5.

4.3.3 Control Surfaces:

Typical sizing of a general aviation aileron structure is approximately 25% of the span and 25% of the chord, extending to the outer edge of the wing. This assumes a straight wing configuration with no aerodynamic or geometric twist. The swept wing design would complicate the calculations due to the inclusion of pitch control. However, with the decision to include an elevator on the fuselage, the pitch control problem was eliminated. Therefore the team decided to use the 25% approximations. The resulting ailerons were 24" spanwise and 4.25" long (calculated using mean aerodynamic chord length).

Based upon analysis done on the wings and the fuselage, a third control surface was added to the fuselage to allow for greater pitch control. Unlike the ailerons, the fuselage elevator size was not arrived at by general sizing rules. The VisualFoil component of the Wing Analysis program was used to determine the effect of flap size and deflection on the airfoil aerodynamics. The optimum configuration based upon the effects on stall, pitch and lift was 6"x18". A larger flap would result in a large negative lift, increasing wing loading and structural stresses. A smaller flap would not generate a significant amount of pitch.

4.3.4 Propulsion and Power System:

The propulsion choices that were made in the conceptual phase of the paper were based on general power management, motor life, and RAC concerns. In the PDP, there was a lot more known about the aircraft configuration and more precise propulsion system design could be conducted.

In limiting the motor type and manufacturer, the contest organizers made the determination of motor type a fairly narrow search. From efficiency data of Astro and Graupner motors, the Astro motors were eliminated immediately. Upon analysis of the available Graupner motors, the only class that were seen as applicable due to thrust requirements and power available were the Ultra 3300 series. These motors are designed to run off of 20 volts and turn a direct drive propeller.

With the class of motor selected, manufacturer supplied data relating motor current draw to the efficiency, output wattage and RPM was used to determine exactly which 3300 series model would be used. Assuming a maximum current draw of 35 Amps, safely below the 40 Amp limit, the 7 wind series was compared to the 8 wind. At this current level, the efficiency, output wattage and RMP were found to be 81.4%, 565 Watts, and 7000 RPM, respectively, for the 8 wind motor. By contrast, at the same current level, the 7 wind motor gave an efficiency, output wattage and RPM of 83.5%, 650 Watts, and 8300 RPM, respectively. Given the added efficiency and RPM output of the 7 wind model, it was chosen as the motor for the final design.

The propeller choice was made from a decision for a cruise speed of 60 mph, at a current setting below maximum. The Graupner Ultra 3300/7 motor operates most efficiently at a current draw between 23 and 29 Amps. At this setting and RPM, an 18" diameter x 12" pitch propeller gives a speed of approximately 70 mph. This factor of safety will be verified during flight testing to determine if an alternate propeller will be needed. For efficiency during glides and descents, experience gained during last year's competition dictated the use of a thin carbon fiber folding propeller, which is durable, light and has less drag in flight when not powered.

The final step in determining the power system was the choice of batteries. The weight, voltage and current limit was dictated in the rules, so the batteries were chosen based on durability, availability and cost. The Sanyo 2400 batteries that were used in last year's competition performed very well under raised current levels and fast discharging, so their reliability had been proven. This fact, combined with their increased availability and reduced cost this year, led to them being selected.

The final configuration for the power plant of one Graupner 3300/7 motor powered by 20 Sanyo 2400 batteries and turning carbon fiber thin folding props of 18" diameter and 12" pitch.

4.3.5 Landing Gear:

At this stage, more emphasis was placed on general sizing to limit minimize weight and drag. It was necessary to optimize the landing gear configuration as well as specific components associated with the landing gear including gear struts, wheels, and gear blocks.

Landing Configuration: The tricycle landing gear places the two main gear behind the C.G. In previous years, the struts were supported by hard points on the spar. However, this year the spar location was 6.5" ahead of the C.G. at the root of the wing. The landing gear would have to be somewhere near the middle of the wing where the sweep brought the spar back behind the C.G. This

would create a large moment arm and unnecessary stresses in the wing structure. The decision was made to locate the main gear at the torsion pin reinforced sections of the fuselage.

Gear Struts: A number of materials were considered for the gear strut, which was designed to interface with the wing spar, including steel, aluminum, and carbon. Carbon and steel would provide the most strength, while aluminum and carbon would be the lightest. Due to the difficulty of manufacturing and lack of impact strength, carbon struts were eliminated as a feasible option. Aluminum, despite its reduced weight, would not guarantee the performance requirements needed and was also eliminated. Due to the proven performance of the previous year's struts, a steel piano wire torsional configuration was chosen for the design of the struts. With this design, the majority of the force of the impact of landing would be absorbed by the torsional loading of the piano wire, and would not be directly transferred to the gear block and ultimately the fuselage.

Wheels: For the construction of the wheels, rubber and aluminum were considered. Due to the weight of the rubber and the fact that traction was not a major concern, the front wheels were constructed from aluminum. Due to the fact that the expansion joints in the runway could cause takeoff problems if the front wheels were too small, the wheels were designed to be 4 in. in diameter and ¼ in. in thickness. The front wheel, however, would only absorb approximately 10% of the impact of the landing, so its size could be considerably smaller. For this reason, and to simplify construction, a standard rubber model aircraft wheel of 3" diameter was chosen.

5 Detailed Design Phase

In the detailed design phase (DDP), the aircraft configuration resulting from the Preliminary Design Phase is was analyzed in more detail to optimize the final product. The detailed analysis consisted of final sizing, general performance, and mission performance estimates. Final sizing of the aircraft components was performed to ensure structural integrity and aerodynamic efficiency. General aerodynamic performance was calculated to ensure 200 ft take-off distances and maneuverability. Finally the mission performance was calculated to provide a final score estimate. These calculations were based on estimates of weight and balance, aerodynamics and engine power. The following table outlines the final aircraft dimensions, configuration and properties.

5.1 Final Sizing

The Preliminary Design configuration analysis resulted in general sizes of wings, fuselage, and all other major components. The detailed design phase refined these measurements taking into account the aerodynamic properties of all parts and overall efficiency.

5.1.1 Fuselage

The main goals for the detailed phase fuselage analysis were a reduction of drag and in increase in overall efficiency.

The Preliminary Design fuselage specifications called for a 16" span and a 40" length of the Roncz/Marske7 airfoil. This resulted in a block shape with adverse pressure and vortices at the side panels. The easiest way to eliminate these effects was to create fairing from the fuselage side panel to the wing root. The inner 16" of the fuselage could not be modified due to space restrictions imposed by the payload. Therefore the plane was lengthened spanwise to include room for the fairing. The variables ideally would include fairing size and airfoil interpolation. However the Wing Analysis program would not allow variation of the airfoil at selected positions along the body. The final results are summarized in the table below.

	Fuselage(20in)	Fuselage(16in)
Speed (mph)	65	65
AOA	**2**	**2**
Span (Inches)	20	16
Sweep (Deg.)/Loc.	0 (c/4)	0 (c/4)
Tip (Inches)	40	40
Root (Inches)	40	40
MAC (Inches)	40.0	40.0
Tip Airfoil	RONCZ/Marske7	RONCZ/Marske7
Root Airfoil	RONCZ/Marske7	RONCZ/Marske7
Planform Area (Sq.Inches)	800	640
Taper Ratio	1	1
Twist Angle (Degree)	0	0
Tip Reynolds Number	2,011,714	2,011,714
Root Reynolds Number	2,011,714	2,011,714
MAC Reynolds Number	2,010,680	2,010,680
Lift Coefficient	0.0058	0.0047
Induced Cd	0	0
Profile Cd	0.0044	0.0044
Total Drag Coefficient	0.0044	0.0044
Moment Coefficient	0.069	0.070
Lift/Drag	**1.30**	**1.08**
Total Lift (Pounds)	**0.34**	**0.23**
Total Drag (Pounds)	0.26	0.21
Mom. (c/4) (Pounds-Inches)	**165.6**	**133.1**

Table 7 - Wing Analysis Summary for Fuselage

A 2" fairing on either side of the fuselage was estimated to be optimum. The lift/drag ratio was increased by 20%. A linear interpolation between the Roncz/Marske7 and the E211 was the most feasible option in terms of construction. Despite the fact that the program could not directly analyze this case it was predicted that there would not be a significant change in results.

5.1.2 Wings

The wing calculations were reanalyzed based upon the detailed fuselage parameters. The wings are required to balance the pitch of the fuselage in order to maintain trim flight. The efficiency of the wings was also analyzed and modifications made to allow for the lowest power consumption and the fastest possible speeds. Stall was also introduced as a variable in the calculations.

In the Preliminary Design Phase the wings incorporated sweep and aerodynamic twist to provide sufficient control of the aircraft. The new fuselage specifications dictated that the wing parameters be adjusted to insure stability. Tip stall was also a concern as the chord of the wing was 70% of the root

chord, causing a lower Reynolds number and a lower stall angle. The design parameters were input into the Wing Analysis program and the final results are summarized below.

	Prelim Wings	Final Wings
Speed (mph)	65	65
AOA	**2**	**2**
Span (Inches)	80	80
Sweep (Deg.)/Loc.	25 (c/4)	25 (c/4)
Tip (Inches)	14	14
Root (Inches)	20	20
MAC (Inches)	17.1778	17.2
Tip Airfoil	NASA/AMES A-02	NASA/AMES A-02
Root Airfoil	E211 (10.96%)	E211 (10.96%)
Planform Area (Sq.Inches)	1360.0001	1360
Taper Ratio	0.7	0.7
Twist Angle (Degree)	0	-2
Tip Reynolds Number	707,478	704,100
Root Reynolds Number	1,010,684	1,005,857
MAC Reynolds Number	868,066	863,921
Lift Coefficient	0.158	0.1569
Induced Cd	0.0033	0.0033
Profile Cd	0.0058	0.0064
Total Drag Coefficient	0.0091	0.0097
Moment Coefficient	-0.0712	-0.089
Lift/Drag	**17.3445**	**16.22**
Total Lift (Pounds)	**16.1248**	**15.92**
Total Drag (Pounds)	0.9297	0.98
Mom. (c/4) (Pounds-Inches)	**-124.9**	**-155.3**

Table 8 - Wing Analysis data for Wings

In order to delay tip stall, a geometric twist angle of -2° was introduced. The pitching moment was increased to keep pace with the fuselage. These figures were used in the final performance analysis and for the final configuration of the aircraft.

5.1.3 Final Configuration Aerodynamic Data

The wings and fuselage were both iterated in the Wing Analysis program to create static equilibrium. The next stage was an angle of attack analysis for the final configuration. Thus far the majority of the calculations had been restricted to trim flight. Varying the angle of attack allows for more accurate mission performance analysis.

The primary concern of the team throughout the initial design processes has been pitch control. The moment diagrams were plotted by the Wing Analysis program and are summarized in the figures below.

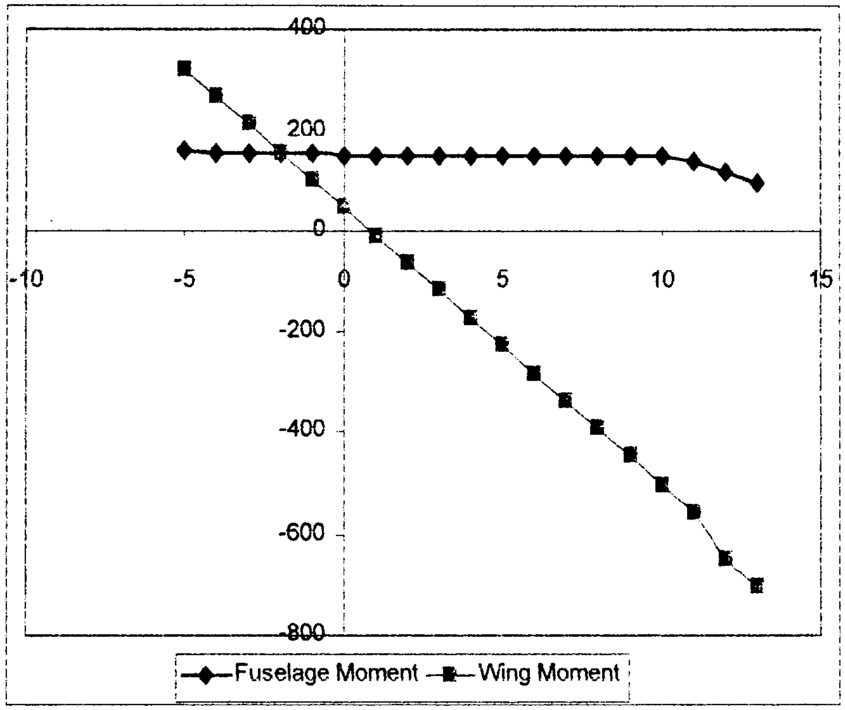


Figure 4 - Wing and Fuselage Moments (in-lb) vs. AOA

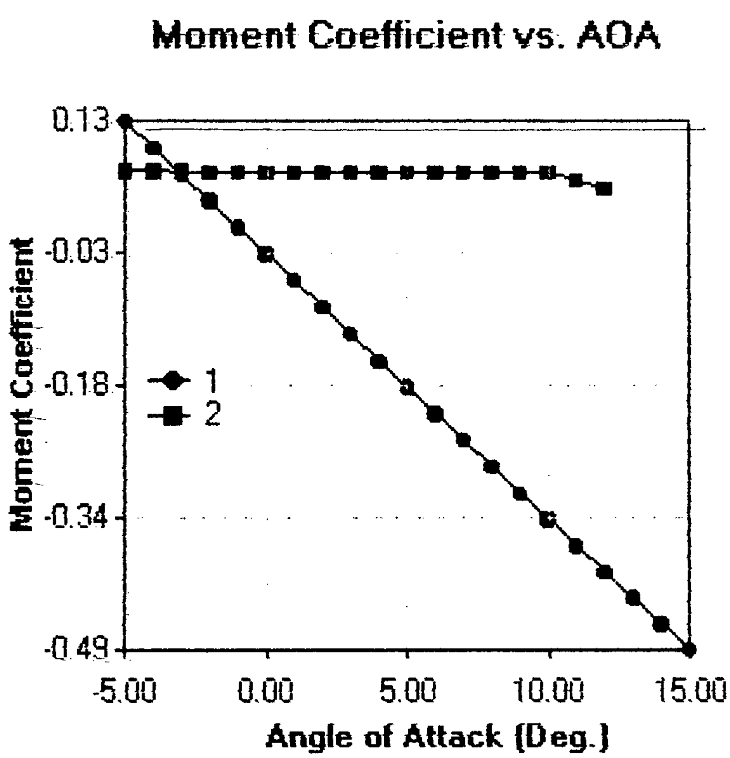


Figure 5 – Fuselage and Wing Moment Coefficients vs. AOA

The plots show clearly that the aircraft has a tendency to remain within a narrow angle of attack range. As the aircraft pitches downward, the wingtips experience a negative lift due to their near symmetrical nature. Therefore the wing structure has an overall positive moment. As the aircraft pitches upward, the wings increase their negative moment, tending to return to trim flight. While this effect was desired in the design process, it potentially limits maneuverability.

Stall characteristic analysis was also performed on the final configuration. The characteristics were derived from lift and induced angle vs. AOA plots.

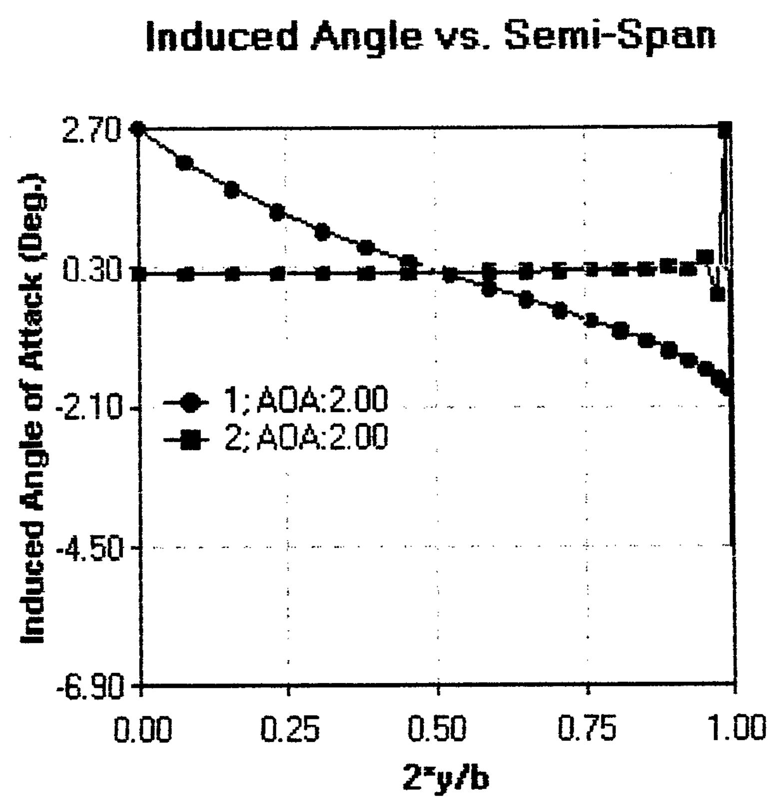


Figure 6 - Induced AOA (Wings-1, Fuselage-2)

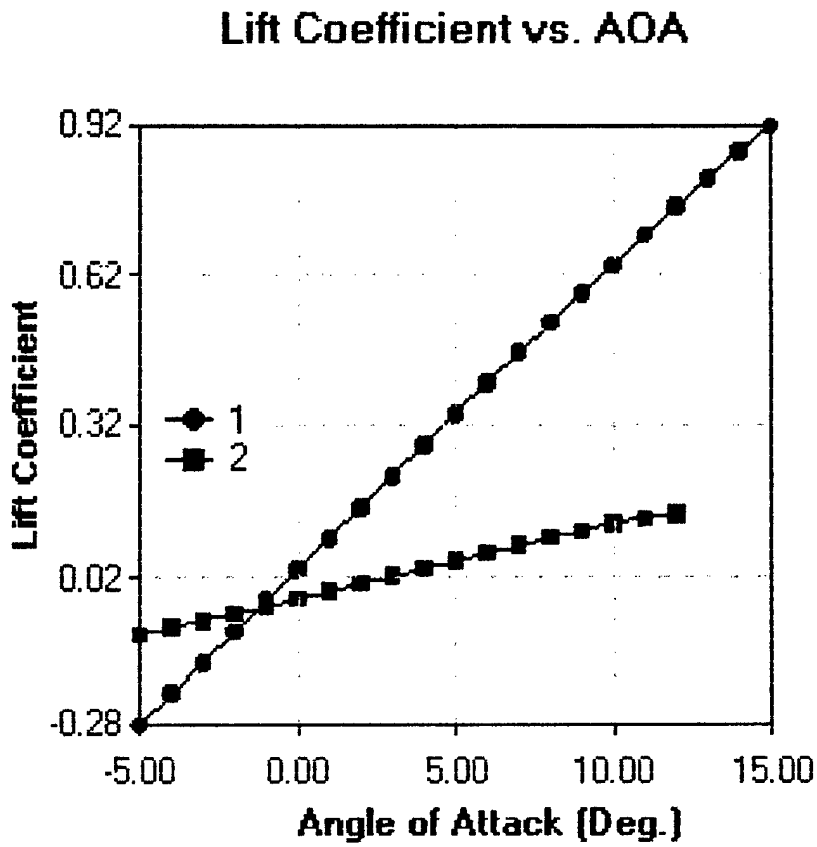


Figure 7 - Cl vs. AOA(Wings-1, Fuselage-2)

C_L versus AOA can be seen in figure 7 and shows that the fuselage clearly stalls before the Wings. From figure 6 the wing tips have a slight downward force in level flight, and the lift can easily be changed to provide roll or pitch control.

The final plots were used to determine the optimum level flight angle of attack. Minimum drag occurs at approximately 3°, which also closely agrees with the trim flight calculations in the final sizing section.

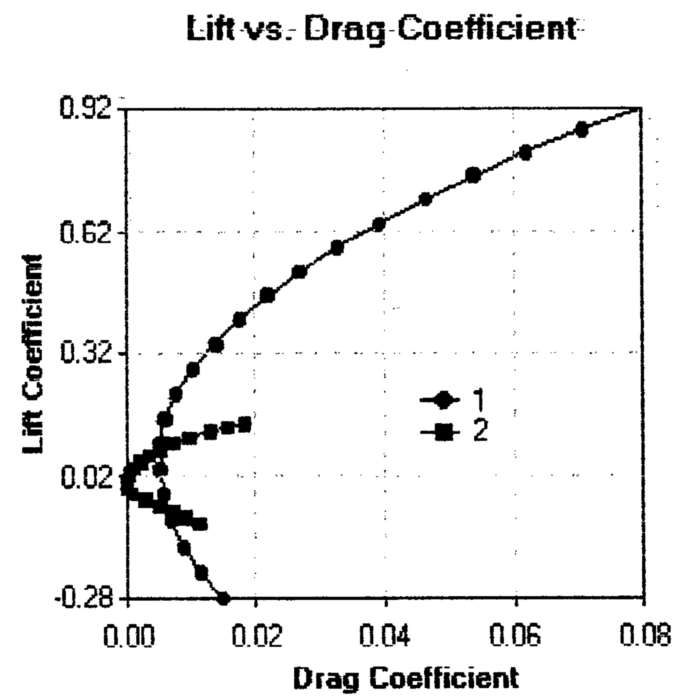


Figure 8 - CL vs CD (Wings-1, Fuselage-2)

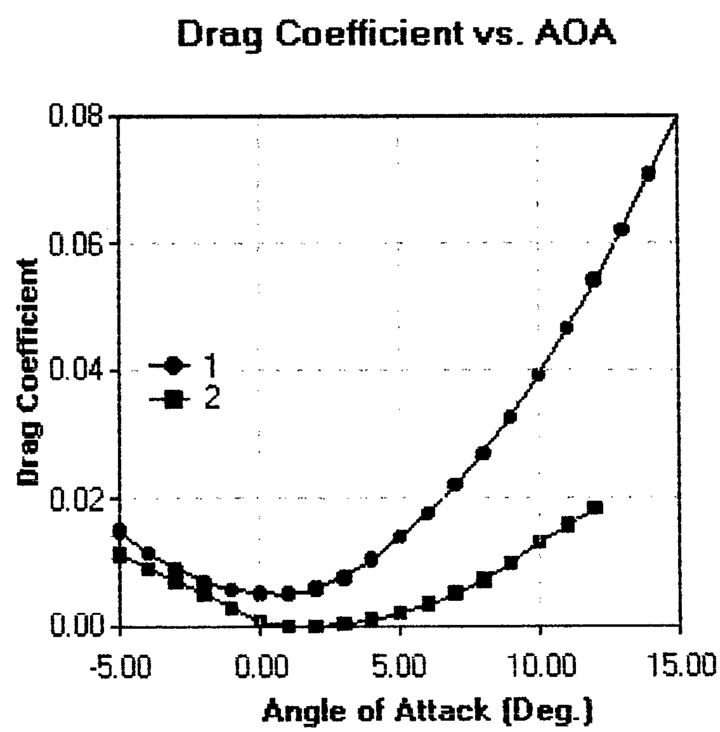


Figure 9 - Cd vs. AOA (Wings-1, Fuselage-2)

5.2 Performance

In the performance calculations it is necessary for the aircraft to remain within its structural and aerodynamic limits. When the aircraft experiences accelerations, the loading of the aircraft changes, this is referred to as the load factor, n. In order for the plane to be gaining altitude, the lift must greater than the weight of the plane, so using the equation, L = n*W, where n is the load factor, gives a lift greater than the weight. As the lift increases the forces and moments experienced in the aircraft also increase, so at certain load factor, the forces are too great and some component of the structure will fail.

The spar, as the backbone of the wings, is subject to load factors more than any other component. Due to the sweep, twist and taper of the spar, it was decided that the finite element (FE) method would be the most accurate analysis method. This was due to the complex coupling effects of bending and torsion under various load conditions. A FE model of the spar was made with the Pre/Post Processor FEMAP and analyzed using MSC/Nastran. It was determined, with a factor of safety of 1.5 and the fact that the wing skins and torsion pin are not included, that the spar could withstand a 3g loading with full payload and fuel. In the performance calculations, the climb rate and turn radius will both be limited by this load factor of 3. A summary of the results in the following sections can be viewed in Table 9 below.

Aircraft Performance	
Cruise Speed	60 mph
Takeoff Distance	112 ft
Climb Rate	764 ft/min
Turn Radius	99.7 ft
Endurance (Average Current)	5.6 minutes
Stall Speed	44 ft/s
Rated Aircraft Cost	**7.995**

Table 9 - Aircraft Performance Summary

5.2.1 Takeoff Distance

The takeoff distance allowed is less than 200ft; however, since time is a large factor in the flight score, the takeoff distance desired is much less than 200ft, since the distance is directly related to aircraft speed. It was determined for takeoff that the rotate/stall speed needed was **43.75 ft/s**, while the actual takeoff speed was **48.1 ft/s**. In order to approximate takeoff distance using an integration of Newton's second law over the ground run of the aircraft, the induced drag was assumed negligible. This integration of the acceleration can be reduced to the following equations from reference 1:

$$\left(\frac{1}{2gK_A}\right)\ln\left(\frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}\right) \quad \text{with} \quad K_T = \left(\frac{T}{W}\right) - \mu\,, \quad \text{and} \quad \frac{\rho}{2(W/S)}\left(\mu\, C_L - C_{D_o} - KC_L^2\right)$$

Using the takeoff speed and the equation below from reference 1 where S_G is the ground roll and K_T and K_A are thrust terms and aerodynamic terms respectively, the takeoff distance was calculated.

$$S_G = \left(\frac{1}{2gK_A}\right)\ln\left(\frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}\right) \text{ where,}$$

$$K_T = \left(\frac{T}{W}\right) - \mu \quad \text{, and } K_A = \frac{\rho}{2(W/S)}\left(\mu C_L - C_{D_0} - KC_L^2\right)$$

A ground roll of **102 ft** was calculated for a rolling resistance factor (μ) of 0.05 and a total takeoff distance of **112 ft** with a rotate factor. This is well within the takeoff constraint and the likelihood of winds being present during the competition will shorten the actual takeoff distance.

5.2.2 Rate of Climb

The rate of climb (R/C) is the vertical component of the aircraft velocity. It can be expressed in terms of weight (W) and velocity (V). W_D is a function of thrust (T) and drag (D), therefore yielding:

$$\frac{R}{C} = \frac{(T - D) \times V}{W}$$

The best climb rate will occur at the velocity for maximum lift to drag ratio, L/D_{max}. Although the thrust produced by the propeller and corresponding power available changes with the airspeed, an approximation was made to show that the maximum lift to drag depends only on the values for the parasite drag coefficient, C_D and the wing's aspect ratio, AR.

$$\frac{L}{D_{max}} = 0.886\sqrt{\frac{AR}{C_D}}$$

Using these formulas, the rate of climb for the aircraft was calculated to be **764 ft/min**.

5.2.3 Turning Radius

The turn radius calculations began with the assumption that the turn will be made at the cruise speed of 60-mph.

$$n = \frac{\left(V_{cruise}\right)^2\left(C_{L_{max}}\right)}{2\left(W_{aircraft}\right)}\left(Area_{wing}\right)$$

This equation gave a load factor of 3 for the turn. From this, and geometry, a bank angle of 70 degrees and turn radius value of **99.7 ft** was calculated.

5.2.4 Estimated Mission Performance

Endurance characteristics are essential performance indicators for an aircraft designed to participate in timed, limited fuel flight mission. The range figure of TLAR 3.5 was based on the power available and the power that the flight profile will require, so that a determination of the number of sorties possible can be made.

The power available is the number of cells multiplied by the power rating for each cell. This equated to 1 battery pack multiplied by 2300 milliAmp-hours for at total of 2.3 Amp-hours. This allows **3.45 minutes** of flight time at the maximum current setting of 40 Amps, which subsequently gave a cruise range of **3.5 miles**.

While this flight time was less than the flight period, an Excel spreadsheet was used to determine the detailed flight plan and current setting breakdown for the entire mission profile, with gliding and powered back cruises to be ample for the completion the mission. The average current draw during the mission was calculated to be **24.7 amps**, which allows for a mission flight time of **5.6 minutes**.

Using an Excel worksheet, the estimated time to complete one mission was **4.85 minutes**. From this time and a payload of 12 softballs, the final mission score will be **3.7**.

The following table shows the RAC calculations.

Rated Aircraft Cost (RAC) = (A * MEW + B * REP + C * MFHR) / 1000 = 7.9952

Manufacturers Empty Weight Multiplier (MEW)				A= $	100
Total Weight w/o payload =	12	Lbs			
				MEW =	11.6
Rated Engine Power (REP)				B= $	1500
# of engines =	1				
Battery Weight =	2.6	Lbs			
				REP =	2.6
Manufacturing Man Hours (MFHR = SUM WBS)				C= $	20/hour
WBS 1.0 Wings					
Wing Span =	8.3	Ft	8hr/ft		
Max Chord =	1.7	Ft	8hr/ft		
# of Control Surfaces =	3	c.s	3hr/c.s		
				WBS 1.0 =	88.76 hr
WBS 2.0 Fuselage					
Fuselage Length =	3.3	Ft	10hr/ft		
				WBS 2.0 =	33 hr
WBS 3.0 Emperage					
# of Vertical Surfaces =	0	v.s	5hr/v.s		
# of Vertical Surfaces w/ =	0	v.s	10hr/v.s		
# of Horizon Surfaces w/ =	0	h.s	10hr/h.s		
				WBS 3.0 =	0 hr
WBS 4.0 Flight System					
# Servo or Motor Contro =	3	Ser	5hr/ser		
				WBS 4.0 =	15 hr
WBS 5.0 Propulsion Systems					
# of engines =	1	Eng	5hr/eng		
# of props =	1	Prop	5hr/prop		
				WBS 5.0 =	10 hr
				MFHR =	146.76
				RAC =	7.995

Table 10 - Rated Aircraft Cost

Using the RAC above, equation for the competition score shown below, the paper score from last year's competition and the competition scoring equation shown below, a final competition score of **120** was estimated.

Flight Score Equation: Flight Score = (Total Laps Flown + Total Balls Carried) / Time

3.717 = (6+12)/4.84

Competition Score: SCORE = (Paper Score * Flight Score) / RAC

120 = (86 * 3 * 3.717)/ 7.995

5.3 Static Stability

The calculations focused on determining the exact placement of aircraft CG and attaining a good Stability Margin. This margin is the geometrical distance between the location of the CG and the location of the neutral point. The neutral point was determined with the following equation along with design parameters defined earlier in the paper.

$$x_N = \frac{l_r}{4} + \frac{2b}{3\pi} \cdot \tan\phi_{c/4}, \quad \textit{for taper ratio} \geq 0.375$$

The neutral point was determined to be at **2 inches** behind the CG. From this and the following equation, the Stability Margin, σ, was calculated to be 0.1.

$$\sigma = \frac{x_N - x_{CG}}{c_{mac}}$$

Table 11 calculates the center of gravity of the aircraft from the weights of the components and their location in the plane. The aircraft is symmetric across the centerline, so there is no moment. The calculation of the moment around the y-axis (parallel to wingspan) was not important, so it was calculated.

Total Length (in)	40		
Item Name	**Item Weight (oz)**	**Arm (in)**	**Moment (in-oz)**
Main Wing	40	20	800
Fuselage Shell	32.88	15	493.2
Fuselage Spar Support	10	8	80
Main Landing Gear (x2)	16	20	320
Nose Gear Structure	5	3	15
Engine (x1)	21.12	4	84.48
Battery Packs (x1)	45	15	675
Propeller (x1)	8	1	8
Wiring Harness	2	18	36
Speed Controller	6	4	24
Receiver	1.65	22	36.3
Main Wing Servos (x2)	2	22	44
Elevator Servo	1	35	35
S-Ball Payload	80	14.00	1120
S-Ball Payload Tie-Downs	0	14.00	0
Empty	0	14.00	0
Total Weight S-Ball Payload	270.65 Oz		3770.98 in-oz
Total Weight No Payload	190.65 Oz		2650.98 in-oz

	Heavy Payload	No Payload
MEW	9.1Lb	9.1lb
Gross T/O Weight	16.9Lb	11.9lb
CG from Nose	**13.93in**	**13.90in**
Distance of CG ahead of Xn	2.0in	2.0in

Table 11 - Weight and Balance

5.4 Systems Architecture

The Systems Architecture describes the electrical components used to fly the aircraft including the batteries, servos, receivers, and handset. The type of motors, batteries, receiver and handset used in the aircraft are shown in table 12.

Given the values obtained for the torque required by each servo, the HS-545BB servo, capable of 62-73 $^{oz}/_{in}$ of torque manufactured by Hitec RCD Inc. was chosen for the elevons, and elevator.

Systems Architecture	
Motor	Graupner Ultra 3300/7
Servos	HS-225MG
Batteries	Sanyo 2400
Receiver	HPD-07RB (PCM)
Handset	Prism 7X (PCM)

Table 12 - Systems Architecture

6 Drawing Package

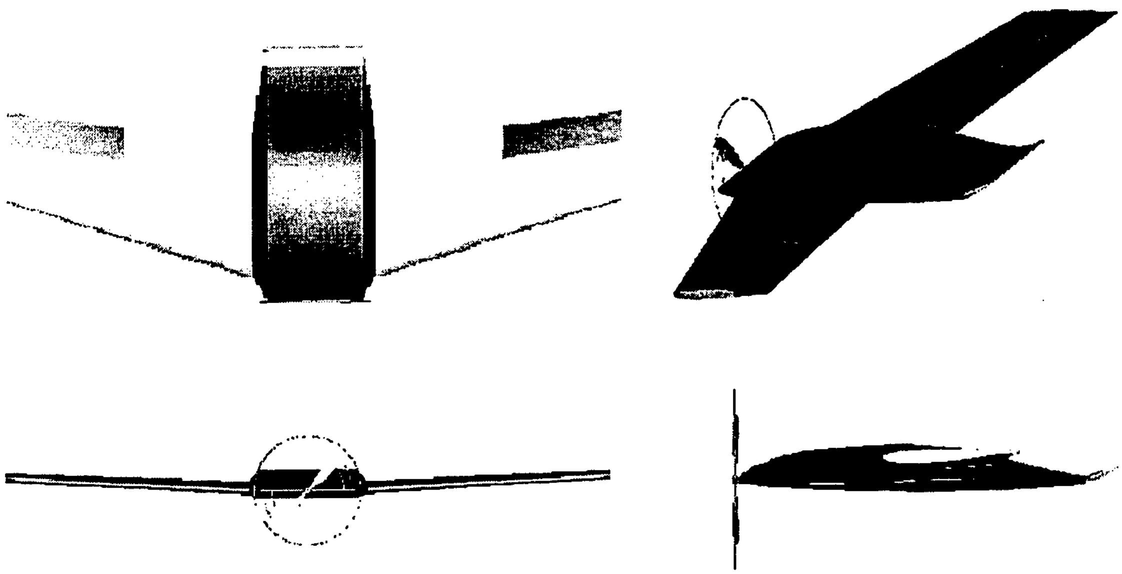


Figure 10 - Three View of Aircraft

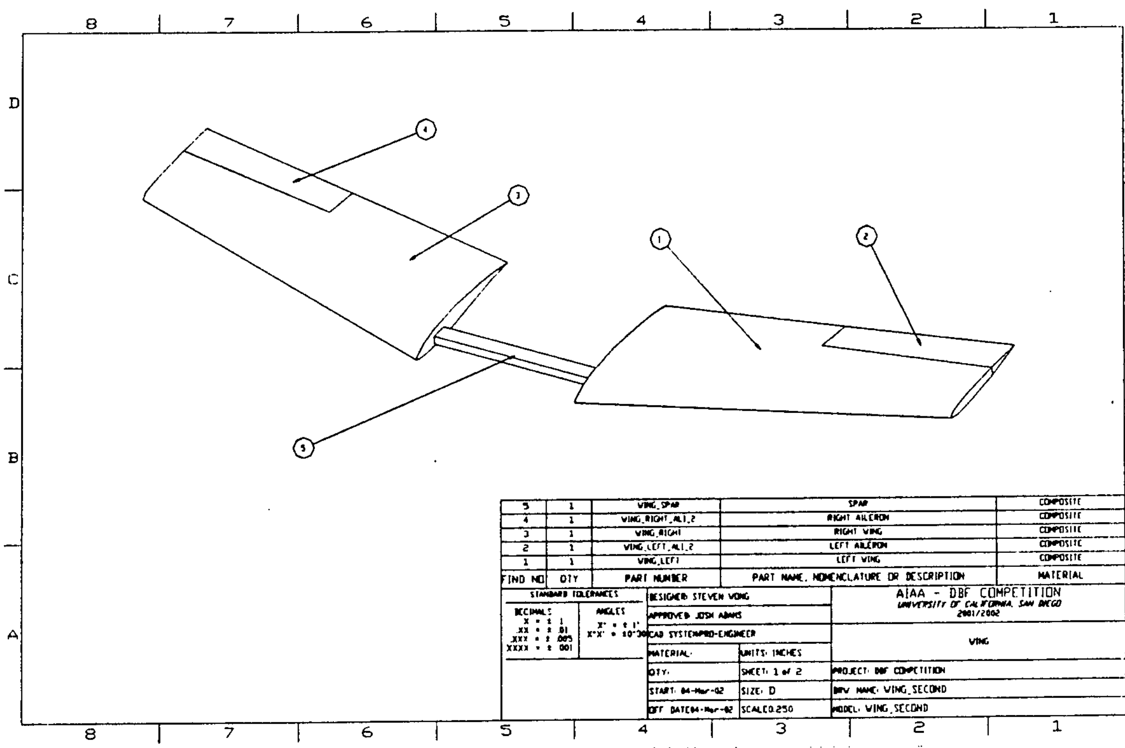


Figure 11 - Wing Assembly Isometric View

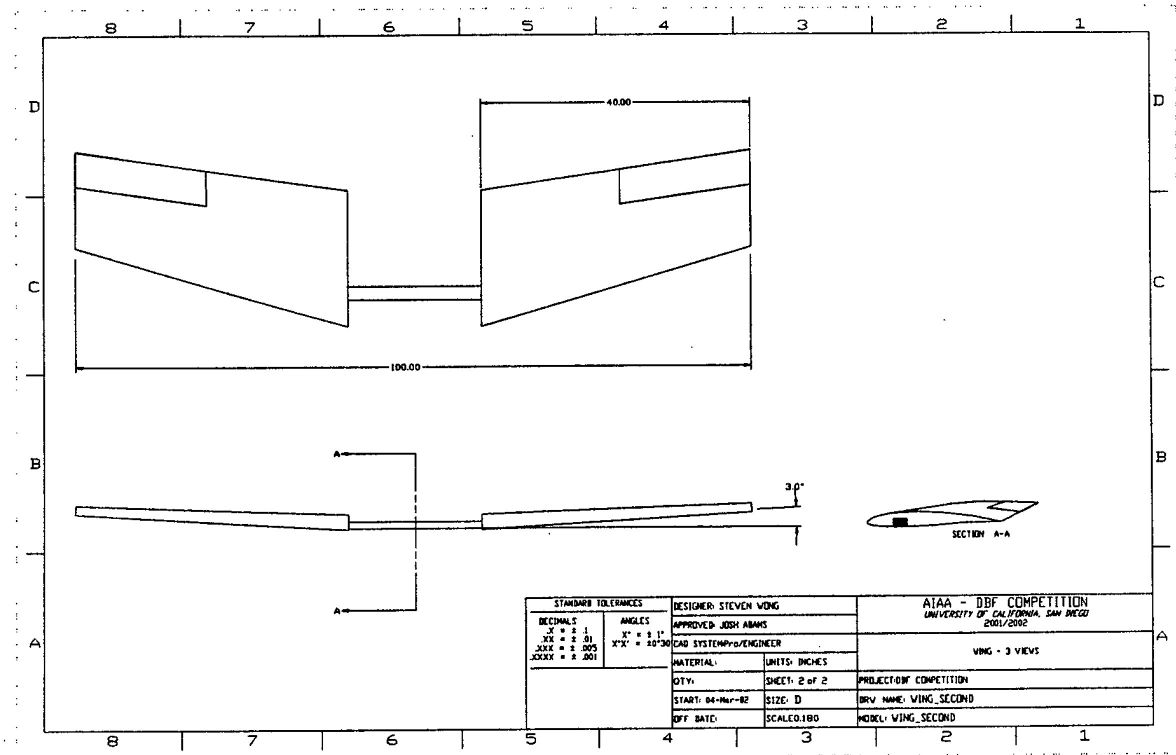


Figure 12 - Wing Assembly

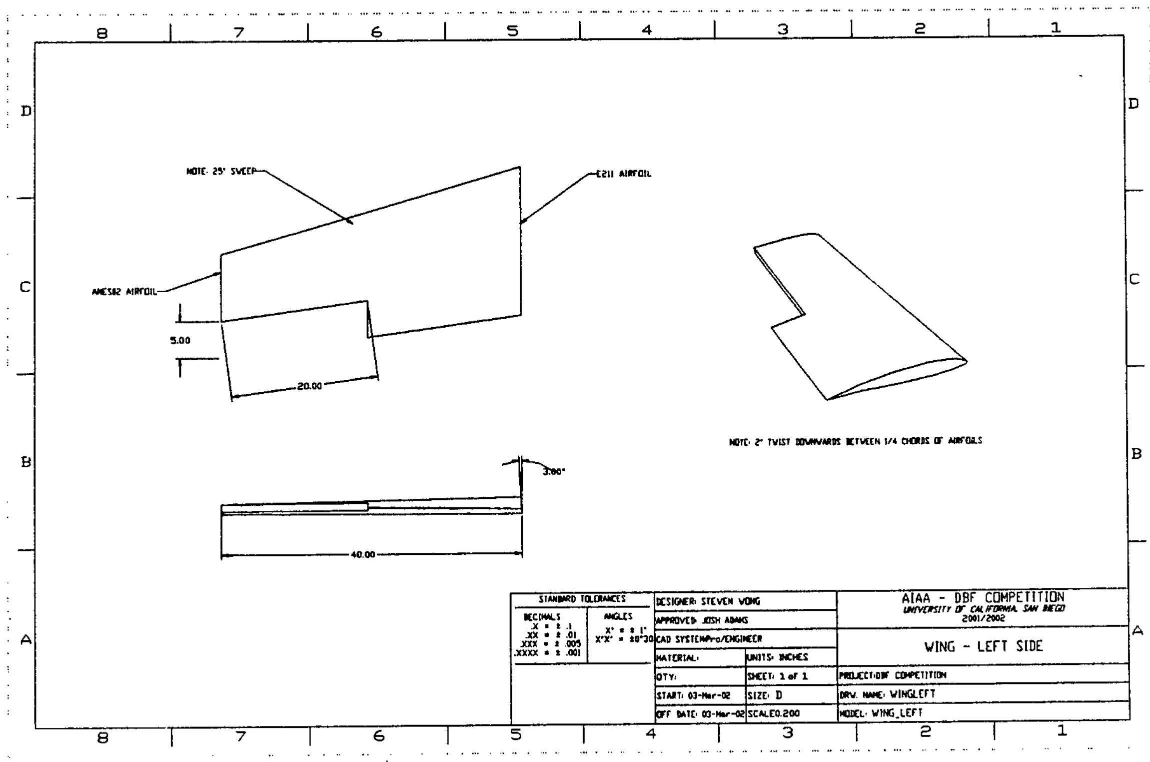


Figure 13 - Wing Detailed View

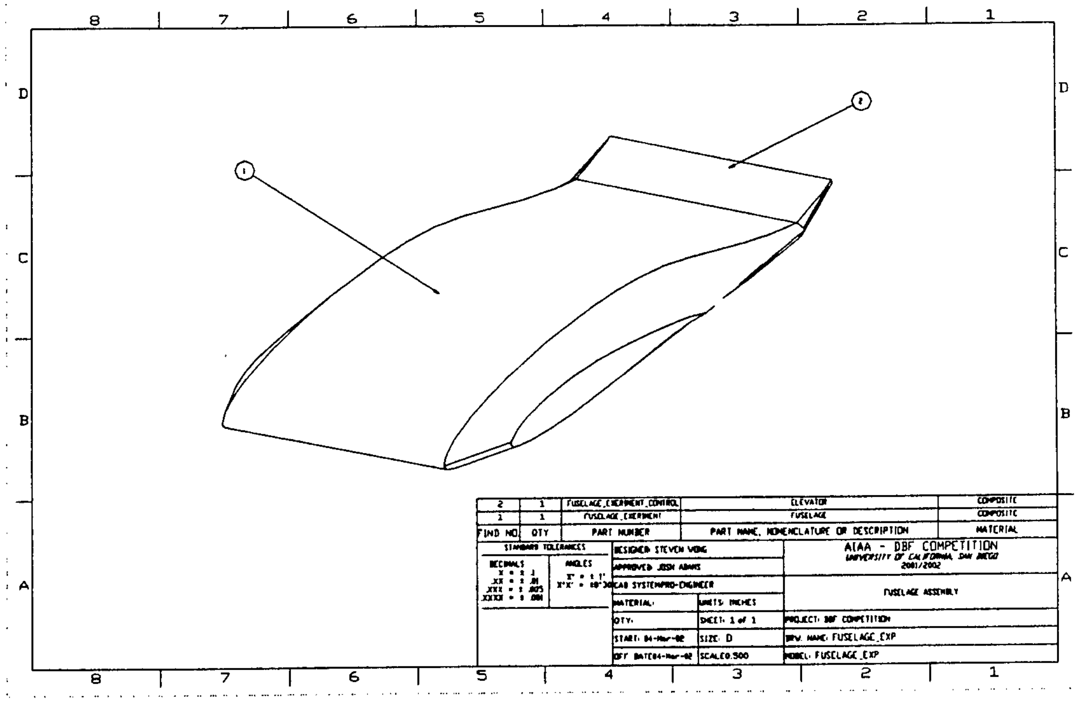


Figure 14 - Fuselage Isometric

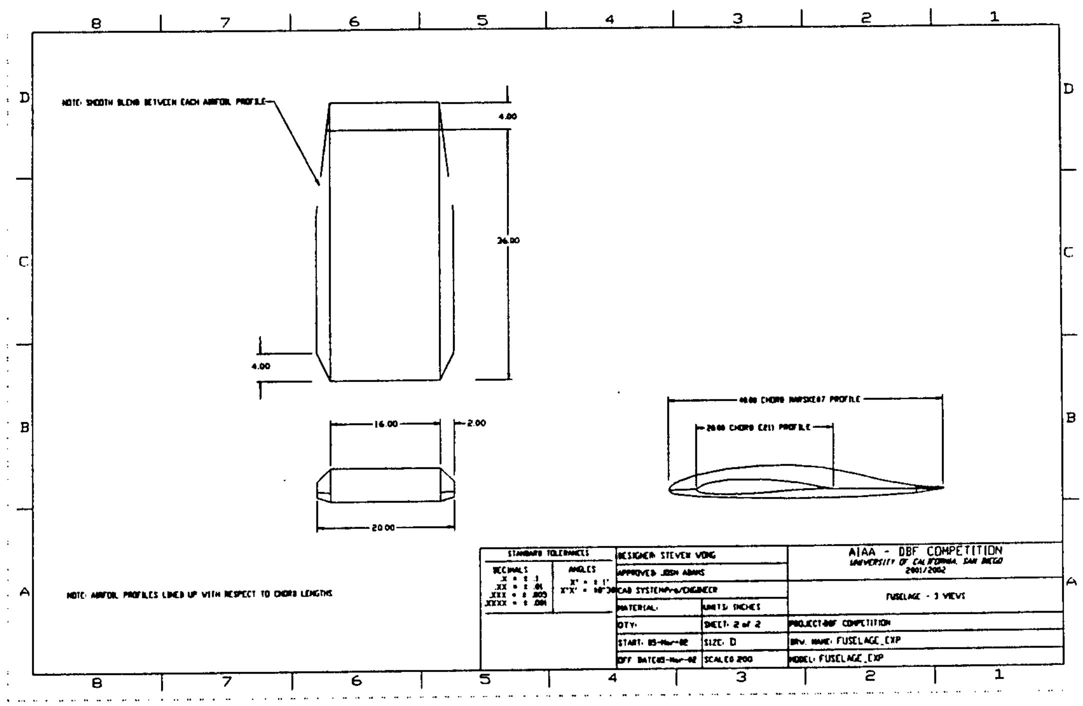


Figure 15 - Fuselage Detailed View

7 Manufacturing

		Strength to Weight Ratio	Cost	Complexity	Time Requirements	Durability	Total
		x3	x2	x2	x1	x1	
Spar	Plywood I-Beam	2	2	1	1	3	16
	Carbon Foam Core	**2**	**1**	**2**	**2**	**3**	**17**
Wings	Plywood Rib	1	2	0	0	2	9
	Foam Core Composite	**2**	**1**	**2**	**2**	**3**	**17**
	Hollow Composite	2	1	2	1	3	16
Fuselage	Plywood	1	2	1	1	2	12
	Foam Core Composite	**2**	**1**	**1**	**1**	**3**	**14**
Landing Gear	**Steel Piano Wire**	**2**	**2**	**2**	**2**	**3**	**19**
	Carbon Composite	2	1	1	1	2	13

Table 13 - Construction Technique Figures of Merit (scale 0-3)

7.1 Figures of Merit

The figures of merit for construction focused mainly on the ability of the team to complete the project rather than performance. As a result, only one of the FOMs affects RAC and flight scores in any way. The others are geared towards the team's supplies of manpower, expertise, and money.

7.1.1 Strength to Weight Ratio

The strength to weight ratio of the aircraft structure is of primary concern as weight affects almost all of the aircraft's flight characteristics. It is also important for the structure to support the loading that occurs during landings and high G turns. This factor was given a weight of x3 to reflect its impact on many aspects of the design process.

7.1.2 Cost

The team had a limited budget to work off of so the price of materials was a large factor to consider. Foam core composites perform exceptionally well but the cost of carbon/fiberglass, epoxy, and foam can be several times that of conventional wood or metal. This FOM was given a weight of x2 to reflect the significance of keeping costs down so that the team would have funds left to get to the competition.

7.1.3 Complexity

The Complexity FOM covers both team skill/experience level and tool availability. The team has limited access to machine shops and scheduling time is problematic. Since the project participation is

voluntary, work is restricted to those times in which members can meet. This is often during the evenings when campus facilities are not available. Therefore any construction technique that requires extensive use of machinery or power tools would not receive a favorable rating.

Individual construction experience varies from person to person. Many of the competition veterans have had experience with TLAR I and TLAR II and the methods used in them. Therefore those manufacturing processes that are most similar to previous years would get a higher rating.

This FOM was weighted at x2 due to the impact of team experience on the project. The team consults with, but is not directed by, more experienced and trained engineers. Therefore the limit of the teams experience is general indication of the limit of the project.

7.1.4 Time Requirements

The time requirements are important for a number of reasons, which include part retries, construction flexibility, and flight testing changes.

Part retries involves remaking parts that have been incorrectly constructed or that have been damaged. If a part requires intensive labor over a long period, it is not as feasible as a readily replaced part. Therefore a higher rating goes to the technique which is faster.

Construction flexibility describes the need to make small adjustments during the construction/assembly of the aircraft. The team builds everything by hand and thus has very generous tolerances. Parts which cannot be modified readily to fit are not advantageous.

The flight testing changes are those large scale changes which might need to be made following the first few flights of the aircraft. Higher scores go to the construction technique that allows for easy modification or replacement.

7.1.5 Durability

Durability is a less important FOM that the others to a certain extent. The parts do need to withstand several flights and also assembly and transportation. However, they only need to last for a few months until the competition is over. Higher scores go to the methods that can endure multiple flight test and competition sorties without structural failure or threat of structural failure.

7.2 Manufacturing Overview

7.2.1 Wings and Spar

Since the wings are the most important component of an aircraft, the decision was made to build the wings and spar first. All other parts would be built around the wing structure. A foam core composite was selected as the technique to be used in construction as the team had extensive experience with it. It also judged to be the best method to maintain accuracy in the wing specifications.

Airfoil templates will be made from formica (due to its low thermal conductivity and ease of shaping) for the upper and lower surfaces of the wing. The templates will account for the taper, aerodynamic twist, and geometric twist of the wing. A hot wire cutter will be used to cut the core of the wing from low-density EPS foam. The spar core will be cut from the wing core as a straight beam at the quarter chord. An extra section will be cut and added to the spar to account for the fuselage pass through.

The spar will be laid up with a single layer 14K tow uni-directional carbon caps to stiffen it in bending. Then it will be wrapped in single layer 12K bi-directional carbon at 45° to account for torsion loads. The layers at the root of the wing will be doubled for added strength.

Before the wing is glassed over, the ailerons are cut and shaped, and the servo mounts are installed. The wiring channels are also cut before glassing occurs. When the wing is prepared, the leading edges and trailing edges are epoxied to the spar and a single layer of fiberglass oriented at 45° is laid up over the entire structure.

7.2.2 Fuselage

The fuselage will be constructed around the wings once they have been completed. Again, foam core composite was selected as the preferable construction technique. It allows for accuracy and strength while not sacrificing weight.

The bulkheads, side panels and payload tray will all be cut and shaped from 1/8" low-density foam sheets. They will be covered in bi-directional carbon and reinforced with 14K tow uni-directional carbon. When set, they will be bonded together around the wing assembly. The 2" fairing between the side panel and the wing root will be cut from the same foam and bonded between the two structures. Holes will be drilled for wiring and to reduce unnecessary weight. A hard point will be constructed in each fairing for the landing gear. The skin will be made of fiberglass laid up over a foam mock-up of the fuselage. The foam will be removed when the epoxy has set and the skin will be bonded to the bulkheads and fairing. Hatches will be cut into the top skin to allow access to the motor and the payload. The fuselage elevator will be made in the same manner as the ailerons.

7.2.3 Landing Gear

The landing gear, unlike the other components, will not be made from composites. This is due to the fact that composites are complex and time consuming. In previous years, landing gear needed to be replaced several times. It would not be feasible to replace composite landing gear.

The landing gear struts will be made out of 1/8" steel piano wire. The wire can be bent using pliers and a vice grip making them easy to modify and to replace should they break. The wheels will be store bought inline skate wheels machined to reduce weight and to conform to the piano wire. Simple store bought EM brakes will also be installed. The struts will fit into the hard points in the fuselage fairing and be secured by plastics tabs and nylon bolts.

Description	Cost/Unit		Unit	Unit Description	Total Cost	
Bi-Directional Carbon Cloth (4.8oz/yard^2)	$	15.00	5	Linear Yards	$	75.00
Uni-Directional Carbon Strands	$	10.00	2	Square Yards	$	20.00
Bi-Directional Fiberglass Cloth (1.0 oz/yard^2)	$	10.00	7	Linear Yards	$	70.00
Epoxy Resin and Hardener	$	30.00	1	Quart	$	30.00
EPS Foam (1 lb/ft^3)	$	25.00	2	3"x36"x60"	$	50.00
High Density Blue Foam (2 lb/ft^3)	$	50.00	1	3"x36"x60"	$	50.00
Tape/Paint/Glue/Other consumables	$	100.00	1		$	100.00
Graupner 3300/7 Electric Motors	$	300.00	1	Motor	$	300.00
HS-545BB	$	35.00	3	Servos	$	105.00
Recievers	$	75.00	1	Reciever	$	75.00
Sanyo 2400 Batteries	$	2.00	40	Battery	$	80.00
Hysteris Brakes	$	25.00	2	Brake	$	50.00
Axel/Wheels/Bearings/Nose Strut	$	45.00	1		$	45.00
Expendable Tools	$	125.00	1		$	125.00
Piano Wire	$	10.00	3	Landing Gear	$	30.00
Softball	$	5.00	12	Payload	$	60.00
Total					**$**	**1,265.00**

Table 14 - Cost Summary

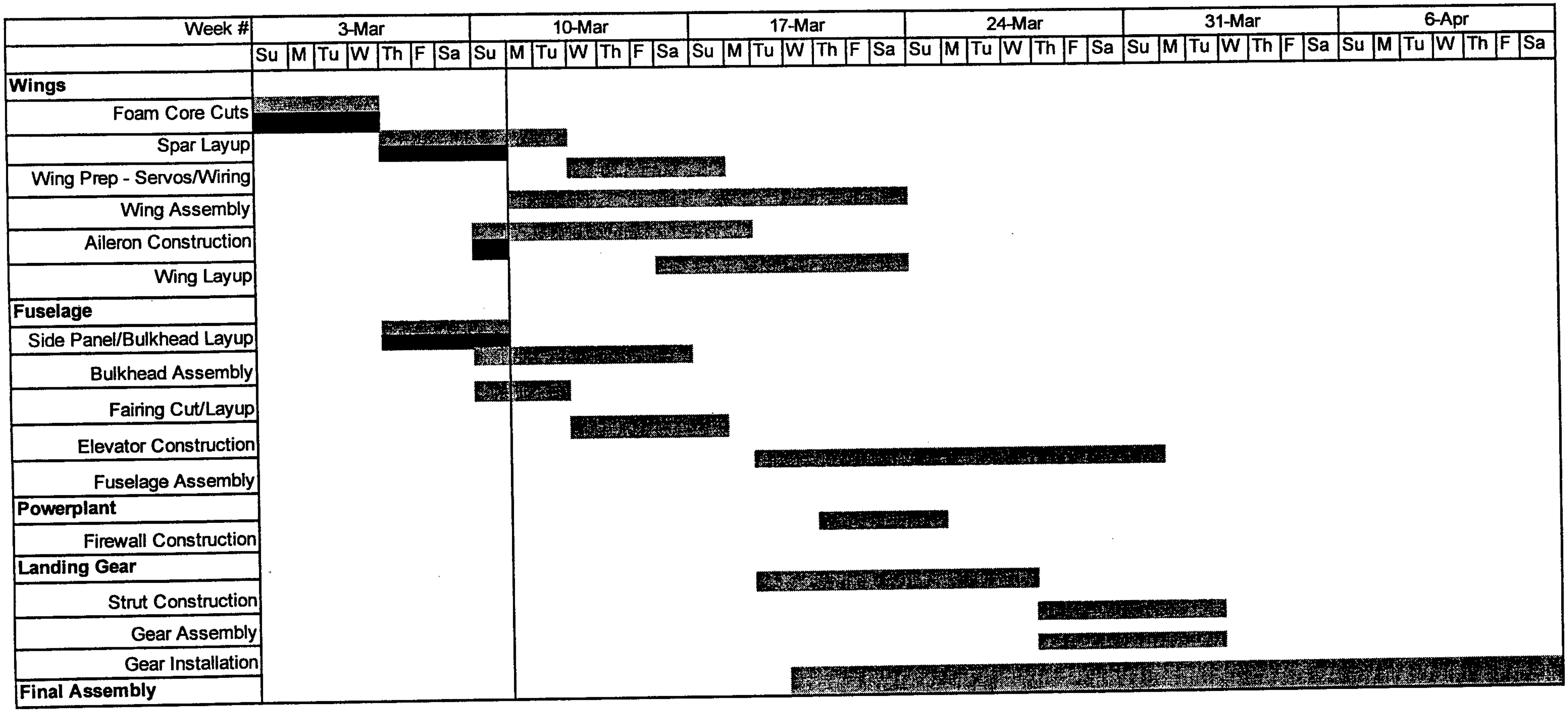


Table 15 - Manufacturing Milestone Chart

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Department of Aerospace Engineering

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Wichita, Kansas
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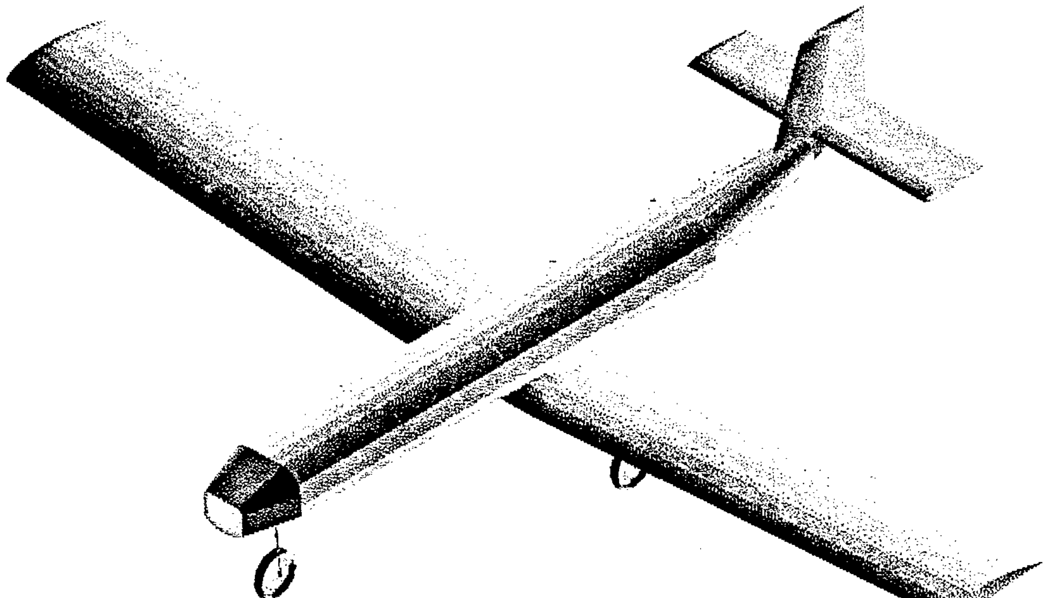
AIAA/Cessna/ONR Student Design/Build/Fly Competition



The WSU Swallow

DESIGN REPORT

March 12, 2002

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1. EXECUTIVE SUMMARY

The AIAA/Cessna/ONR Design Build Fly Competition is an annual event of undergraduate aerospace students from around the world creating original electric powered R/C airplanes capable a flying a given cargo around a particular course. This year, the cargo is between ten and twenty-four softballs, and the course is around two pylons spaced 1000 feet apart. The mission consists of three phases: empty with turns, full with turns, empty without turns. Take-off is limited to 200 feet. The entire mission must be completed within ten minutes in order to avoid a penalty, and score will increase greatly with reduced time.

The primary objective of the team was to design and build a plane capable of earning the highest possible total score. Total score is a function of written report score, flight time, number of softballs carried, and the "rated aircraft cost" according to a complex function provided by competition designers. The task of designing an airplane that is not only flyable but is optimized to these criteria is very complex. To accomplish it, the team followed processes outlines in Raymer, Aircraft Design: A Conceptual Approach.[19]

1.1. Conceptual Design Phase

The objective of the conceptual design phase was to study a spectrum of basic aircraft alternatives and to select a basic configuration for more advanced design. Beginning in September of 2001, about one and a half months was spent researching previous designs, brainstorming new designs, and analyzing these designs both qualitatively and quantitatively with specially designed tools.

Design Alternatives

A wide range of alternative for wings was considered ranging from the totally conventional to the extremely exotic. Some of the more notable concepts investigated were a monoplane, bi-plane, canard, three-lifting-surface, blended wing-body, and a Custer-channel wing. For propulsion, tractor and pusher props were considered in tandem with multi- or single-engine designs. Multiple fuselages were considered for span-loading purposes as well as various lifting bodies. Empennage configurations included conventional, T-tail, V-tail, and multiple vertical surfaces. Retractable and fixed gear were debated in both tricycle and tail-dragger configurations.

Design Tools

For initial comparison and trade-studies, a simple tube frame structure was assumed to model structures and loads and to compare the many designs. A "rubber engine" model was used for propulsion, and basic aerodynamic principles used for identifying performance characteristics. To analyze and link these results, the team specially made an Excel-based design tool called GENESIS (acronym for General

Engineering Numerical Evaluating and Systems Integration Software) consisting of separate sheets for each major design area. These areas linked results to a central main page where they were used by all other relevant calculations in other areas. In this way, circular dependencies were created and Excel was programmed to iterate to a solution (up to 1000 iterations) each time any design change is made. Versions of this design tool were developed for each configuration selected to allow analysis in depth. Each was briefly optimized and then compared to the figures of merit for rated aircraft cost, weight, speed, complexity, and final predicted score.

Conceptual Results

Even before in-depth analysis was completed, many of the more exotic concepts considered were calculated to have only minimal potential when working on the scale of an R/C airplane. In exchange for a very small improvement in score, the complexity of design and building would have increased dramatically, so these designs were rejected. Also, based on these studies, landing gear was selected to be of the tricycle type for better ground handling. It was further decided to use fixed gear rather than retractable gear because the drag savings, at such low airspeeds, was not judged to outweigh the complexity of retractable gear. Multiple engines were ruled out because the negative effect on the rated aircraft cost function dominated the calculated increase in speed gained. Further analysis, using the GENESIS design tool, proceeded with the monoplane, bi-plane, and three-surface, each one with a conventional and V-tail version. It was calculated that the performance for all three was basically the same, and the rated aircraft cost function was significantly lower for the simpler designs. Therefore, the final decision was made that the very simple and conventional monoplane with a single tractor engine was to be the Wichita State University entry. A further payload study, evaluating a smaller payload while going faster around the course, suggested that the number of softballs carried was the dominant term in the score function and the payload should be the maximum number of softballs allowed.

1.2. Preliminary Design

The purpose of preliminary design was to size major components, improve weight estimates, and refine analytical tools to find more accurate performance and stability data. The team divided itself into the four major development groups of aerodynamics, propulsion, structures, and stability and control, to facilitate this design step.

Xfoil, a readily available airfoil analysis program, was used to compare a wide range of candidate airfoils. The Wortmann FX-63-137 airfoil[26], specifically designed for low Reynolds number and low power flight, was chosen for its high coefficient of lift, consistent low drag at a range of flight conditions, and gentle stall characteristics. The wing was then sized using LinAir[14], a commercial potential flow software, to predict three-dimensional effects on wing performance. Results from LinAir for a matrix of wing dimensions and loadings, were used in the ever-evolving GENESIS design tool, and a contour of final

scores with respect to wing parameters developed to optimize wing loading and aspect ratio. Material selection considered load estimates, strength-to-weight ratios, and manufacturing issues. Metal, carbon fiber, wood, and foam were among the types of materials considered for various load-bearing elements of the design. Metal and carbon fiber were ruled out in favor of wood and a foam construction for their lightweight and ease of working properties. Sizing of primary structures such as the wing spar was accomplished using classic strength of materials and fundamentals of structures analysis methods. The fuselage was sized to hold the maximum number of softballs while presenting the minimum frontal area. Performance analysis refinement included using basic electrical circuit relationships to predict electric power system limits and battery requirements, while the "rubber engine" method was continued. Take-off field length and its impact on power required, as well as statistical efficiencies of system components were also modeled in the GENESIS design tool. Stability concerns such as initial empennage sizing, control surface sizing, basic stability criteria, rotation angle for take-off, and fuselage length (as a result of empennage location) were considered using fundamental methods in combination with extensive GENESIS tool usage to maximize or minimize figures of merit including total expected score.

1.3. Detailed Design

At the beginning of the spring semester, detailed design commenced with the purpose of optimizing all aircraft components with respect to final score, as well as accurately predicting aircraft performance and developing construction schemes. The team continued to operate in its parallel and extensively cross-linked fashion, though use of the GENESIS design tool used in conceptual and preliminary phases waned as more complex methods were applied.

Structural sizing was completed in depth for a variety of wing spar cross-sections and a solid rectangular section of laminated spruce wood was chosen. A stressed skin fuselage structure with a non-load-bearing removable top was also designed after rejecting a keelson design and a design with a small hatch for loading and unloading. Vehicle drag assumptions made in preliminary design were supported by detailed analysis of fuselage drag using a combination of fundamental fluid mechanics methods, Xfoil[31], and LinAir Pro[15], an upgrade from the LinAir used earlier. Total vehicle aerodynamic performance was estimated using the LinAir Pro potential flow software. An Astro 640S motor was selected for its close similarity to the specifications of the desired "rubber motor" resulting from preliminary performance analysis. Propeller selection was facilitated with the help of a blade-element theory program called Airscrew. MATLAB[16] and Simulink[23] are software suits that were used extensively to optimize handling qualities and crosswind performance.

The final aircraft appears simplistic at first but is the result of careful and thorough engineering in all aspects of its design. It carries twenty-four softballs tightly in eight bays, has a nine-foot wingspan, cruises at forty-eight mph, weighs fifteen pounds empty, and can operate in a thirty-mph crosswind

component. It is expected to complete each sortie in less than seven minutes, and earn a single flight score of 4.38 for each flight, while having a rated aircraft cost of only 8.84.

2. MANAGEMENT SUMMARY

2.1. Team Architecture

The first priority after formation of the team was to divide the design project into four distinct engineering disciplines: aerodynamics, structures, propulsion, and stability and control. Other anticipated tasks included communications, fabrication, computer programming, and technical report writing. It was further discussed, and agreed, that each member of the team should take chief responsibility for one of the engineering disciplines and one administrative duty according to our personal strengths. In addition, each member chose a cross-discipline to work on under the supervision of the chief engineer for that section. The breakdown of accountability is as follows:

Member Name	Primary Engineering	Secondary Engineering	Other Tasks
Michael Boccia	Propulsion	Structures	Fabrication Crew Chief
Eugene Heim	Stability and Control	Propulsion	Information Technologist
Marcus Mosley	Aerodynamics	Stability and Control	Editor of Technical Reports
Melinda Schwasinger	Structures	Aerodynamics	Administrator
Kevin Pfeiffer	Propulsion Assistant	Structures Assistant	Fabrication
Tom Farrell	Materials	Miscellaneous	Fabrication
Matt McCarthy	Materials	Research	Purchasing
Seth Lamble	Stability and Control Assistant	Aerodynamics Assistant	Fabrication

Table 2.1.1. Primary and secondary divisions of responsibilities for each team member.

Michael Boccia is qualified for the propulsion lead, structures assistant, and fabrication crew chief due in part to his experience building and flying model RC planes. He has also completed the undergraduate curriculum for both propulsion and structures. Due to his years of experience, Mr. Boccia will also serve as the pilot for testing and for the competition.

Eugene Heim was selected to lead the stability and control effort because of his undergraduate/graduate work in the discipline. He also spent the past summer working in the Vehicle Dynamics Branch at NASA Langley. Mr. Heim is qualified to work on the propulsion aspect of the project because he has completed the undergraduate propulsion curriculum. He also has experience using Excel, Simulink, MathCad, MATLAB, Fortran 77/90, and AUTOCad, therefore he was selected to be the information technologist.

Marcus Mosley was chosen to be the aerodynamics lead because he has taken both graduate and undergraduate courses in the field. He also has industry experience as a student intern in aerodynamic performance at Raytheon Aircraft Company. Mr. Mosley has completed all requisite stability and controls courses; therefore he was chosen to assist Mr. Heim in the stability and controls aspect of the project. He also has experience writing technical papers for the Federal Aviation Administration. For this reason, he was selected to be the editor of all technical reports.

Melinda Schwasinger's qualifications lend her to serving as the structures lead, aerodynamics assistant, and communications facilitator. She has taken both undergraduate structures classes and several graduate classes including a Finite Element Analysis class. During her summer program with NASA Goddard, she performed solid mechanics analysis and structural design. Ms. Schwasinger has completed all undergraduate aerodynamics classes, which qualifies her to work with Mr. Mosley on the aerodynamics portion of the project. She has also served as an administrative facilitator on projects at Wichita State University as well as in industry at Cessna Aircraft Company and NASA Goddard.

For the four seniors of the group, this project served as their senior design project for academic credit, and thus, took leadership positions and personal responsibility for project success. The four underclassmen assistants were volunteers and helped out in areas that followed their interest and experience as needs arose. The resulting design going to the competition is a product of all involved working as a team.

2.2. Configuration Management and Accountability

All major decisions were made democratically, and minor ones were delegated to the appropriate individual. If not obviously falling under one of the above areas of responsibility, then a member was appointed to address the issue. A team leader was selected, though his tasks were minimal. It was for

him to make the agenda for regular team meetings and to cast a tie-breaker vote in any decision, given the even number of members on the team. Also, the team leader acted as the team's representative to the faculty. For leadership, the project was divided by semesters. During the fall semester, leadership responsibilities were assumed by Eugene Heim because of his enthusiasm for the project and his leadership experience in other organizations. In the spring semester, leadership was transferred to Michael Boccia because of his experience in building and flying model airplanes. It was believed this experience would give him superior judgment in areas of building and other small details likely to arise and requiring a swift decision.

All work completed by any member of the team must be presented to the other member sharing that discipline. That person will check the work in depth and approve the work done or make recommendations for improvement. If approved, he will "sign off" on the work, thereby accepting responsibility for its accuracy. After approval, the engineer will summarize his results to the entire team at the next regular meeting for review/discussion/approval.

To facilitate communication and document sharing, two devices were implemented. First, a "Yahoo! Briefcase" account, an internet file storage service, was opened in the team's name. Here, all members kept daily backup copies and archives of all documents, spreadsheets, programs, tools, correspondence, etc., relating to the project. In this way, all files could be accessed by any team member instantly from any internet-connected computer at any location. The second device was a document of "Shift Notes." To a shared Word document on the school network, members would add brief summaries of their work in-progress each time as work is completed. This approach kept all members informed continuously of the efforts of all other members, and had the result of keeping the regular meetings fairly short.

2.3. Schedule

Task Name	Start	Finish
WSU Design Build Fly Project	**Thu 9/6/01**	**Sun 4/28/02**
Conceptual Design Phase	**Thu 9/6/01**	**Thu 11/15/01**
Research	Thu 9/6/01	Fri 9/21/01
Analytical Tooling	Thu 9/6/01	Mon 10/1/01
Tooling (Actual)	Thu 9/6/01	Thu 11/15/01
Comparitive Analysis	Mon 10/1/01	Fri 10/5/01
Comparitive (Actual)	Mon 10/8/01	Fri 10/19/01
Configuration Selection	Fri 10/5/01	Fri 10/5/01
Selection (Actual)	Fri 10/19/01	Fri 10/19/01
Preliminary Design Phase	**Mon 10/8/01**	**Thu 1/31/02**
Wing Design	Mon 10/8/01	Fri 11/16/01
Wing Design (Actual)	Mon 10/8/01	Wed 12/12/01
Power System Design	Mon 10/8/01	Fri 11/9/01
Power System (Actual)	Mon 10/8/01	Mon 11/12/01
Load Path Design	Mon 11/12/01	Wed 12/12/01
Load Path (Actual)	Mon 11/12/01	Mon 1/21/02
Empannage Sizing	Mon 10/15/01	Wed 12/12/01
Empannage (Actual)	Mon 10/15/01	Thu 1/31/02
Detail Design Phase	**Mon 1/7/02**	**Tue 3/12/02**
Performance Optimization	Mon 1/7/02	Thu 2/7/02
Performance (Actual)	Mon 1/7/02	Tue 2/26/02
Handling Qualities Opt.	Mon 2/4/02	Thu 2/28/02
Handling (Actual)	Mon 2/4/02	Thu 3/7/02
Structural Refinement	Mon 1/7/02	Thu 2/28/02
Structural (Actual)	Mon 1/21/02	Fri 3/8/02
Design Report Preparation	Mon 2/4/02	Sun 3/10/02
Design Report to AIAA	Tue 3/12/02	Tue 3/12/02
Manufacturing	**Mon 2/4/02**	**Thu 4/25/02**
Fabrication	Mon 2/4/02	Fri 3/29/02
Engineering Mod Time	Mon 4/15/02	Thu 4/25/02
Flight Operations	**Mon 4/1/02**	**Sun 4/28/02**
Test Flight Program	Mon 4/1/02	Fri 4/26/02
Contest	Fri 4/26/02	Sun 4/28/02

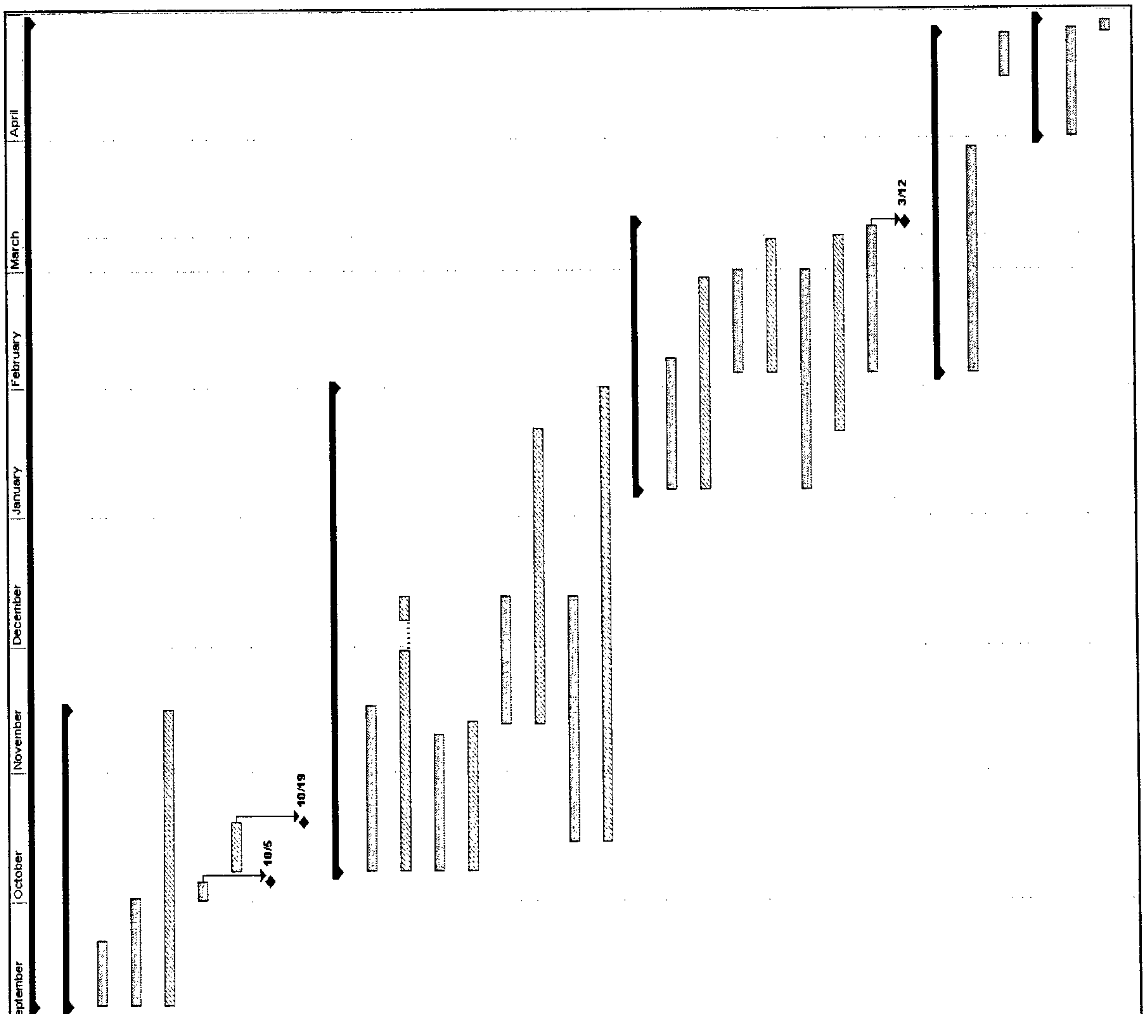


Figure 2.3.1. Planned and actual schedule encompassing entire development process.

3. CONCEPTUAL DESIGN

3.1. Fundamental Design Approach

The fundamental approach of the design team was that as is outlined in Raymer's Aircraft Design: A Conceptual Approach[19]. Initially the team listed the requirements constraints of the project. A study was then conducted to determine aircraft parameters to optimize in order to maximize score. Next preliminary weight and mission profiles were generated based on statistical estimates. An initial configuration was then conceived. From this configuration, performance parameters were analyzed and then optimized through preliminary and detailed design. Construction on the craft followed this optimization.

At the conclusion of the construction stage, the initial flight-testing will begin. Depending on the results of those tests, simple modifications to the plane may be necessary. Following modification to the craft, the craft will be ready for the competition.

3.2. Evaluation Tool

Even before a basic aircraft configuration was selected, an evaluation tool had to be developed that could appropriately analyze potential designs and predict their success in terms of chosen figures of merit leading to a competitive score. Desired to be as thorough and accurate as possible, it would have to integrate as many different aspects as possible, from the major areas of aerodynamics, propulsion, stability and control, and structures. Beginning as a single function to calculate rated aircraft cost, it quickly evolved into an all-inclusive basic design tool to evaluate mathematical functions, provide efficient communication of progress and results among team members, and to link all efforts in such a way that any change created by one member would automatically update all relevant functions and results in the other team members' areas, preserving conformity. The platform decided upon was Microsoft Excel, as opposed to developing a new computer code. Chosen for its familiarity, flexibility, ease of use, personalization possibilities, and inherent mathematical functions, Excel also provided a clear and organized working environment in which to manipulate the data and see results in real time. One main spreadsheet contains the results of all relevant variables related to airplane dimensions, performance, and other parameters as they are calculated or chosen. This sheet is organized by sections including performance, aerodynamics, propulsion, weights, and structures. Additionally, new spreadsheets within the same workbook are created as needed by the team members to perform analysis and store information regarding stability and control calculations, aerodynamic analysis, airfoil properties, weights and CG tracking, structural sizing, fuselage design, propulsion optimization and performance calculations, wing size optimization, and score prediction. All these sheets are links to named variables on the main sheet and output their results to the main sheet so that they can be used in relevant analysis in other sheets. All this cross-linking quickly leads to coupled problem solving, however. For instance, a change

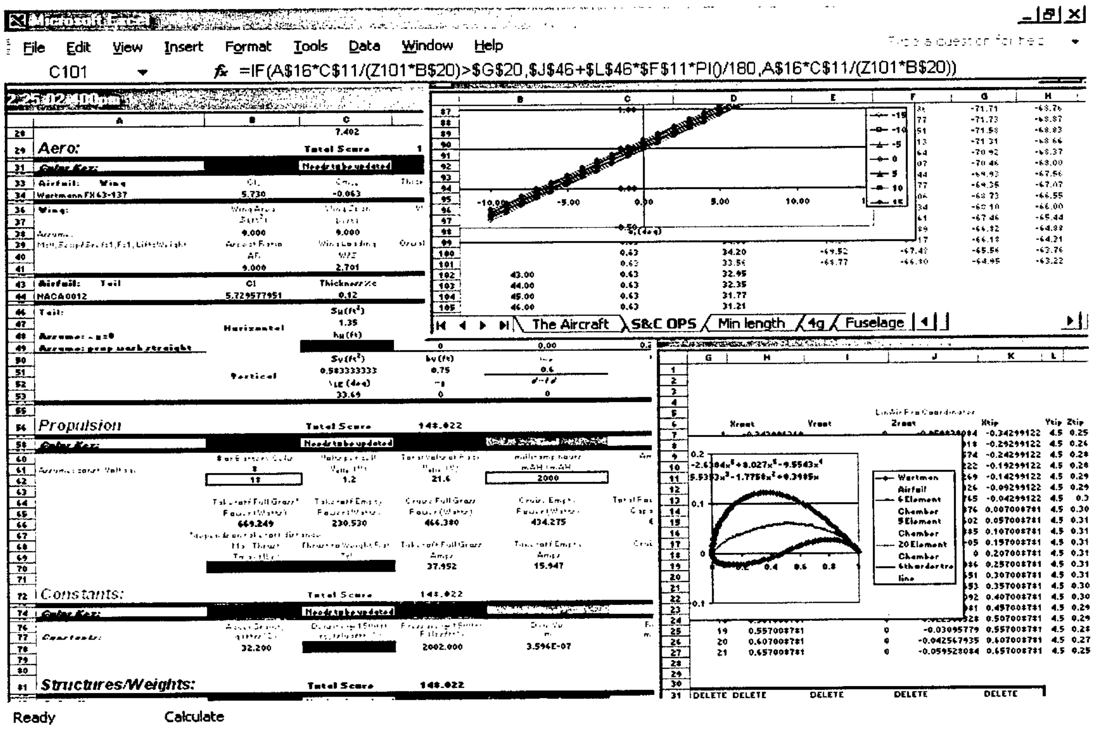


Figure 3.2.1. Screen shot from the Excel-based GENESIS design tool developed by the DBF team specifically for this project.

in weight will automatically effect a change in take-off power required. In the event that this increase in power necessitates an extra battery, then that will, in turn, affect the total weight, which will increase the power required, etc. Excel was so configured, then, to solve such circular expressions using a built-in iterating scheme set to a maximum of 1000 iterations. Each time any variable is changed by any user, each circular function, often several hundred separate and nested circles, is recalculated up to 1000 times or until a steady state is reached, whichever occurs first. In most cases this takes less than a second. The result of this innovative and, as far as the team is aware, original organizational approach, was named GENESIS, for General Engineering Numerical Evaluation and System Integration Software. The pay-offs were near total and instantaneous integration of design efforts among team members in a continuously dynamic fashion, and a minimum of redesign effort necessary because of a change in a different engineering area. GENESIS was especially valuable in the early design phases when sizing and initial performance estimates were constantly in flux, and continued to be the primary means of sharing and comparing engineering efforts throughout preliminary and detailed design.

3.3. Alternative designs investigated

The basic premise for the development of the design tool was a very basic and generic aircraft possessing "one of everything." A single, rectangular wing, a single engine, conventional empennage, no landing gear at all, and a basic tubular aluminum structure of one spar for the wing and one keelson for

Figure 3.3.1. Design concepts evaluated.

the fuselage. Once the master spreadsheet design tool was developed to a sufficient level of complexity, it was then easily modified for multiple wings, fuselages, engines, and even more exotic configurations such as the Custer-channel-wing, tailless, joined wing, and flying or blended-wing-body. These were the result of several brain-storming sessions of the entire team, along with some suggestions by members of the Wichita State University faculty. Figure 3.3.1 shows freehand sketches of many of the configurations initially suggested.

3.4. Design Parameters Considered

Wings:

The wing is quite possibly the major feature in determining the overall success of the airplane. Much attention was given to alternative wing configurations, and research was done to discover the latest technology and trends in wing design. Alternatives considered were a monoplane, biplane, tandem wing, canard, three-surface (canard plus lifting tail), joined wing, flying wing, and Custer channel wing. The Custer channel wing was an intriguing design that uses the prop-wash to accelerate the air over the upper surface of the wing, fashioned into a half-duct, to produce lift even at very low forward speeds. In fact, in one case, Willard Custer accidentally became air-born while taxiing his prototype at only eight miles per hour[29]. This, however, was ruled out before analyzed to the figures of merit because no analytical text or tools could be found, and such a design would have too small a wing to glide very well in the event of a power failure. Other such exotic designs were excluded from numerical analysis based on qualitative decisions of complexity, feasibility, and potential benefits. The joined wing concept, like the channel wing, contained too many unknowns and would require too much original testing before the benefits could even be quantified. The stability of a flying wing or a blended wing body is very difficult to achieve and even more difficult to control, and was ruled out of the basis of complexity. Perhaps future teams can build upon what is established here and pursue some of these ideas. Other multiple wing configurations were believed to provide a benefit in empty weight as well as performance over conventional designs and were selected for further analysis.

Engines

For early analysis, the "rubber engine" method, as described in Raymer's Aircraft Design: A Conceptual Approach[19], was employed using realistic linear relationships between power available and engine and battery weights, without having to pick an actual engine and battery pack. The choice then was to use either a single, powerful engine or multiple smaller engines. A quick sensitivity study showed that to increase the number of engines and propellers by one each would reduce overall score by 10.3%, even after accounting for performance increases, making engine configuration the most weighty factor in score next to the competition flight time (13% per minute). It was so decided that the number of engines and props would be fixed at one, as long one engine would be capable of providing enough thrust.

Empennage:

Alternatives considered for the empennage included a conventional tail, a T-tail, and a V-tail. The benefit of the T-tail over the conventional is not in aerodynamic efficiency or weight, but in it's interaction with the wing downwash at high angles of attack and with the vertical tail in uncontrolled spins. The decision whether or not to have a T-tail was then postponed to preliminary design. The V-tail, for which there was specific provisions made in the rated aircraft cost function, could provide some benefit in that respect. Therefore, each configuration studied also included a separate V-tail version.

Fuselage:

The concept of span-loading the wing(s) with two fuselages, as in the manner of the P-38 or the Rutan Voyager, was considered to have significant savings in structural weight. However, further structural analysis showed the savings, though significant in full-sized aircraft, to be much smaller in model aircraft. Further, when competition officials were consulted about the ball configuration rules, it was discovered that the balls must be at a minimum of two abreast in each fuselage, effectively doubling the cross-sectional area and increasing form drag. This concept, then, was ruled out as too costly in terms of weight and performance.

Landing gear:

Landing gear configurations were considered purely conceptually. A tricycle configuration was believed to have good ground handling while the tail dragger to have slightly less drag in flight. Also, the mechanics of connecting a tail wheel to the rudder would be easier than a nose wheel. However, when confronted with the requirement to take-off in a cross-wind up to thirty miles per hour, stability and control evaluation strongly pushed the decision toward a tricycle arrangement, which was the ultimate conclusion. Retractable gear was ruled out for complexity and rated aircraft cost for extra servos.

3.5. Figures of Merit

To compare design concepts, it was desired to include as many factors as possible such as the rated aircraft cost, speed around the course, power required, as well as complexity and practicality. The last two were difficult to quantify and would be used only qualitatively in the final decision. A mathematical sensitivity study, was conducted to discover which factors were weighted most in the total scoring function in terms of percentage change in score with respect to an incremental change in each parameter. The result showed the most important factors, in order, are as follows: written report score, flight time for each sortie, number of engines, total battery weight, number of balls carried, and empty weight. All other factors had less than two percent affect. Of those, written report score and number of balls carried were set to constant values at their maximums among all configurations. Optimization of the number of balls carried would be conducted in preliminary design. Also, the number of engines was fixed

to one, as discussed above. The remaining figures of merit used to compare concepts were then flight time, battery weight, empty weight, rated aircraft cost, and, of course, total expected score.

3.6. Quantitative Comparison

Copies of GENESIS were altered to calculate the benefits and costs of these configurations in parallel, and their results compared. This first-order analysis, however, had some key limitations. The weight of the aircraft was calculated to affect the power required for take-off as well as battery consumption, but the difference in structural weight among the various configurations was so slight as not to necessitate the changing of the engine or the addition/subtraction of a battery from the battery pack. Also, it was realized that analysis of the drag savings in multiple wing concepts, due mostly to tip vortex effects, was beyond the capability of the design tool to predict, and thus the performances predicted were equal. Vortex analysis being beyond the feasibility limit of this project, this limitation was accepted. The result is, as shown in Table 3.6.1, is that the final predicted score is almost totally a function of the rated aircraft cost.

	Mono	Mono V-Tail	Tandem/ Bi-Plane	Tandem V-Tail	3-Surface	3-Surf V-Tail	Joined Wing
Total Score	**151.473**	**153.857**	**139.606**	**141.103**	**134.385**	**132.224**	**168.181**
	1.00	1.55	-8.50	-7.35	-12.72	-14.56	9.93
Written Report Score	**100.000**	**100.000**	**100.000**	**100.000**	**100.000**	**100.000**	**100.000**
Total Flight	**13.161**	**13.161**	**13.161**	**13.161**	**13.161**	**13.161**	**13.161**
Single Flight Scores	4.387	4.387	4.387	4.387	4.387	4.387	4.387
Number of Balls:	24.00	24.00	24.00	24.00	24.00	24.00	24.00
Laps Flown:	6.00	6.00	6.00	6.00	6.00	6.00	6.00
Total Mission Time:	6.838	6.838	6.838	6.838	6.838	6.838	6.838
Rated Aircraft Cost	**8.689**	**8.554**	**9.428**	**9.328**	**9.794**	**9.954**	**7.826**
Empty Weight	14.556	14.209	14.114	14.114	14.092	14.092	14.151
	0.00	-2.44	-3.13	-3.13	-3.29	-3.29	-2.85
Rated Engine Power:	2.143	2.143	2.143	2.143	2.143	2.143	2.143
Total Battery Weight:	2.143	2.143	2.143	2.143	2.143	2.143	2.143
Number of engines:	1	1	1	1	1	1	1

Table 3.6.1. Comparison of conceptual design alternatives from the GENESIS design tool.

The joined wing concept was included in the comparison study to see the potential benefit of an exotic design, even though it was already decided that it was too complex to be quickly and easily implemented. It may be an idea that a future team, building on work first established by this one, may consider. For this year, having very little to build on, a generally conservative approach and a philosophy valuing simplicity was adopted. This approach further enforced the rated aircraft cost model being the dominate figure of merit in the selection processes. The qualitative analysis also revealed that a conventional configuration (monoplane, tractor prop) with a V-tail provides a negligible (1.15%) increase in score over the conventional configuration. Finally, all things considered, a conventional design proved to be the optimum choice.

In addition to the basic configuration, the desired payload to achieve the highest score had to be determined before design could proceed. It was realized that a plane carrying fewer softballs, though earning fewer payload points, can go much faster and therefore earn higher competition points. To decide on a desired payload configuration, a trade study was performed. A varying number of balls was calculated with its influence on total score. For each condition, motors, batteries, fuselage length, and wing geometry were manually re-optimized in GENESIS. Despite the highly coupled nature of these variables, the resultant relationship, as shown in Figure 3.6.1, is nearly linear. The payload carried is given such a high weighting factor in the score, that the more softballs carried, the higher the total score, regardless of performance trade-offs.

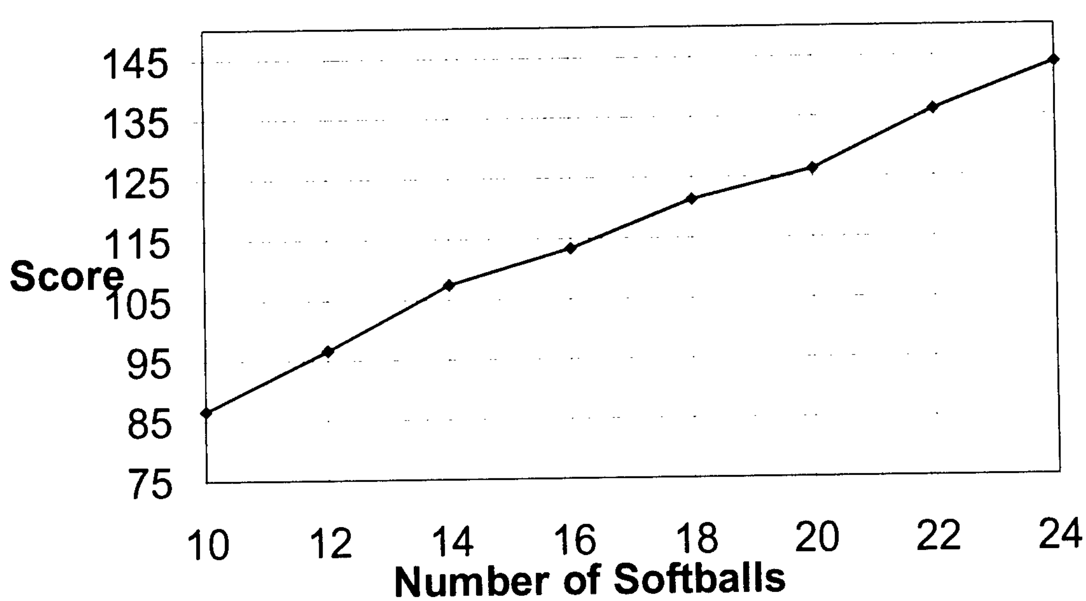


Figure 3.6.1. Score variance with respect to payload capacity. The maximum number of softballs was chosen at the target payload in order to maximize score.

4. PRELIMINARY DESIGN

The selection of a basic configuration, being a monoplane with a conventional tail, tricycle landing gear, and a payload of twenty-four softballs in a more-or-less tubular fuselage, powered by a single engine and prop, marked the end of the conceptual design phase and the beginning of preliminary design. At this time, tasks were divided among the four major disciplines, each one working largely autonomously. Due to the small size of the team, each member took responsibility for their own primary and secondary specialties, and coordinated efforts through the team leader, who himself also had a full share engineering responsibilities in addition to a leadership role.

4.1. Aerodynamics

Airfoil Selection

The first duty of the Aerodynamics department was airfoil research and selection. This posed a challenge due to the low Reynolds numbers at which the model will be flying, which was first estimated to be around 300,000. A large survey was conducted of appropriate airfoils. Some of the airfoils that showed promise were the NACA 4414, and the Wortmann line of low-speed airfoils. Some of the Wortmann airfoils produced lift coefficients upwards of 1.9, but at very high drag and a narrow range of acceptable angles of attack. Eventually, the Wortmann FX-63-137 was selected. It's $C_{l,max}$ and $C_{d,min}$ is more favorable than the NACA 4414, at 1.6 and 0.012 respectively. In addition, the drag "bucket" is relatively flat for a large

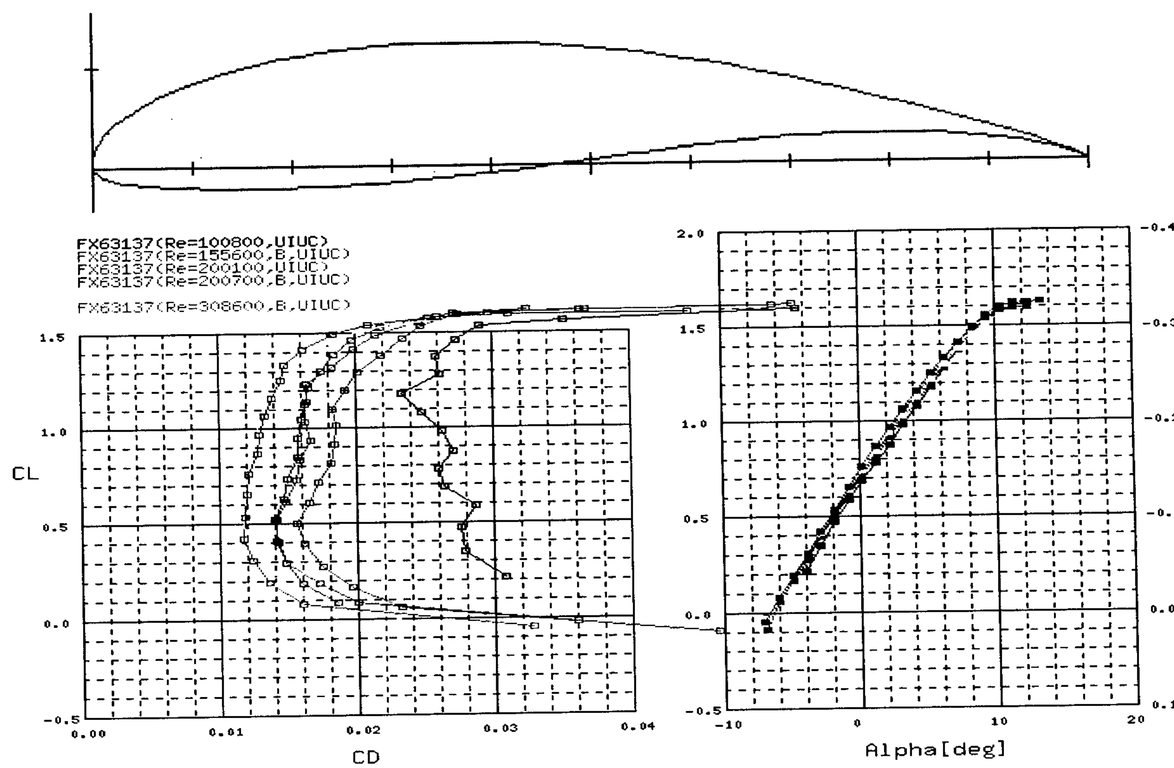


Figure 4.1.1. The Wortmann FX-63-137 airfoil with published experimental data.[26]

range of alpha's giving very consistent performance at different loadings (i.e., a laden or un-laden Swallow). Also, at the point of stall, the lift levels off while the drag increases sharply, rather than the lift dropping off suddenly. This should lead to a gentler stall characteristic, which the pilot would appreciate. Furthermore, the 13.7% thickness of the airfoil allows plenty of room for a robust structure, as opposed to some of the more exotic and highly cambered airfoils. The drawback of the Wortmann is the pronounced trailing edge cusp and the very thin trailing edge. These characteristics are present in most high-performance, low-Reynolds-number airfoils, presenting challenges that will be faced in fabrication. However, it was decided that the payoffs are worth the extra attention in keeping to tolerance in the construction phase.

Wing Sizing by Score Optimization

Figure 4.2.1. shows the first iteration of score optimization, which was the method for sizing the wing, as planned from the beginning. On the Y-axis is the wing area, on the X axis is the wing chord, and thereby span, aspect ratio, etc. On the Z axis, which is represented by colorized contours, is the final total score

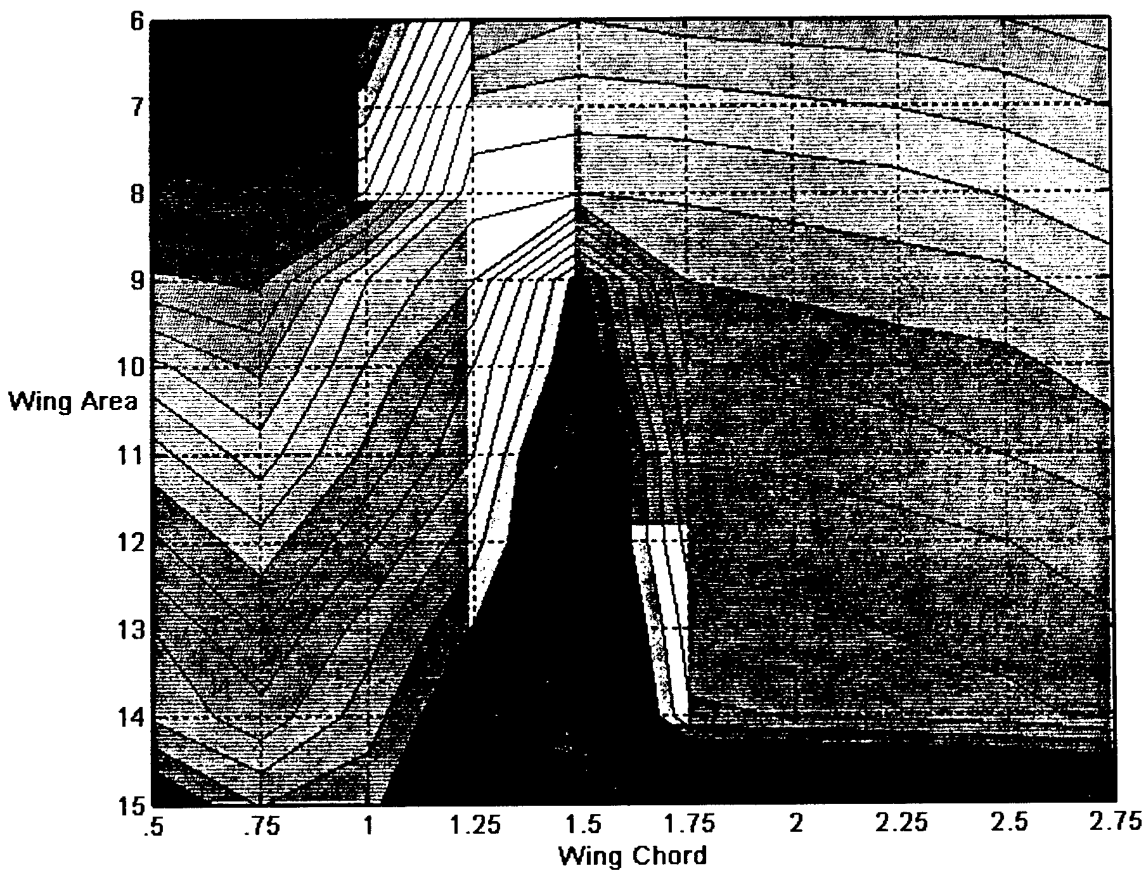


Figure 4.1.2. Contour of total score with respect to wing geometry. Used to size the wing to 9 ft^2.

expected. The score takes into account all factors of the cost function, as well as performance of the airplane and how it leads to an estimated time to complete the course. Earlier attempts at optimization could only take into account the cost function, which resulted in a linear contour and trivial solutions. Finally, when Aerodynamics teamed with the Propulsion department, and after the actual motor selection had been narrowed down to two, hard numbers for performance were obtained and an estimation of flight time made. The largest factors effecting score are flight time, which dictates a small, fast wing; and battery weight, which suggests a large, glider-like wing. For this study the number of softballs, also a major contributor to score, was kept at a constant twenty-four, the maximum allowed.

In this graph, two areas were invalidated for two different reasons. The shaded area on the left represents those conditions that exhibit a wing Reynolds number low enough that the desirable airfoil characteristics of the Wortmann airfoil are lost. Obviously, this occurs on wings with very small chord-lengths, but also depends on the flight speed of the aircraft, which accounts for it's slightly curvilinear nature. The shaded area on the right represents those conditions where required take-off power could not be produced with the selection of motors provided by the Propulsion department. This occurs with wings of low aspect ratio (high induced drag), or just excessive size and weight. The area left is a narrow corridor of possibilities where the plane can not even get off the ground. Within this corridor it is an easy matter to see the highest score occurs at the smallest possible wing. Further iterations using higher weights showed that the shaded areas come together at the top of the graph, and the maximum score can always be found at that intersection.

Also utilized, in order to more accurately predict performance trade-offs with different wing geometries, were extensive tables of aerodynamic results. LinAir, a linear potential flow aerodynamic modeling software, was used to analyze 100 different wing geometries in a 10x10 matrix, which corresponds to the optimization plot. Data for Alpha at stall, max CL, CD, and Oswald Efficiency Factor, where recorded in tables in the Excel-based GENESIS design tool. Excel was then configured to automatically search the tables and select the appropriate values for each wing geometry. As the tables show, the range of values was quite large and had a sizable impact on performance predictions, and further contributed to the non-linear nature of the final graph. A similar process was implemented for the horizontal tail.

This iterating fully coupled process of optimizing was facilitated by the use of the GENESIS design tool developed for conceptual design. Unfortunately, automating the optimization of the wing as shown would have required a database 100-times as large as the current one, which would make it too large to fit on a CD-ROM. So the optimization was done manually by the Aerodynamics department, which was a long and tedious process, but with good results.

There was a total of 3 iterations of wing optimization. Each time there was an improvement in weight estimates, another optimization routine would be performed and new optimum wing geometry selected. As illustrated, with each iteration, the "target" was narrowed and the "bull's eye" became more precise. The final wing geometry is a 9-ft span and a 1-ft chord. To guard against future unforeseen changes in weight invalidating the wing, the current configuration includes a 4-pound safety factor. As long as the weight does not increase more than 4 pounds, or 27%, from the current projected weight, the 9-ft wing will be ideal. In the unlikely event that the prototype is constructed underweight, then further analysis coupled with flight testing may lead to a reduction in the span. The span should easy to modify since it is a straight rectangular wing of simple construction.

Horizontal and Vertical Stabilizer Airfoil Selection:

For the horizontal stabilizer, the requirements from the Stability and Control department were to have an airfoil with low drag at the resulting Reynolds numbers, influenced by small chord length, and one with a C_l at alpha=zero between zero and negative two degrees, so that it can be inverted and create negative lift for trim at very low trim drag. The Aerodynamics department added the constraints that it must be easy to build, thick enough to be robust in structure, and have a very low drag "bucket" about the lower C_l range.

First studied was the NACA 0009, having a C_l at alpha=zero of zero and being equally effective in pitching upwards as well as downwards. Xfoil analysis[31] showed that at Re's of 200,000 and less, the estimated range of the stabilizer, separation occurred at very low AOA's. If the thickness of the airfoil was increased, attached flow could be maintained to twelve degrees, which was decided to be reasonable, and the NACA 0012 was selected for further investigation. Also studied was the Rolf Girsberger RG15, having a C_l at alpha=zero of exactly –2 degrees and a thickness of about 9% and performs best at Re's below 300,000, it seemed to match initial concepts perfectly. Its C_l was low at about 1.0, and its $C_{d,min}$ was low at about 0.008. However, two other factors lead away from the RG15: the thin profile would make a foam-only construction, favored by structures, difficult, and the realization that construction would be kept simpler if the horizontal and the vertical stabilizers happened to be similar. Returning to the NACA 0012 in Xfoil, which is the obvious pick for the vertical fin for the separation issue mentioned above, it was found that the $C_{d,min}$, which occurs at alpha=zero for a symmetric airfoil, was also about 0.008 at a Re of 200,000, and it's $C_{l,max}$ and $C_{d,max}$ were similar to the RD15 as well. The NACA 0012 also had the added benefit of a more robust thickness for structural purposes. Stability and Control indicated that the necessary elevator deflection for cruise trim with the symmetric airfoil would only be about 1 degree, so no great trim drag would be incurred over the RG15. So the NACA 0012 was settled upon for both the horizontal and the vertical stabilizers.

4.2. Stability and Control

The time restriction, structure and contest location requires the aircraft to be able operate in a wide variety of conditions. The team does not have the luxury of choosing when and where they want to takeoff or land. Severe or erratic wind conditions, the air temperature, and even the proximity of the crowd and officials must be taken into account. The underlying goal is to design an aircraft that maximizes the competition score equation while ensuring safety and performance.

The major concern is the maximum allowable wind speed of the competition, 30-mph. Depending on the direction, wind speeds of 30 mph can be helpful, a hindrance or even devastating to the performance of the aircraft on the takeoff roll. A 30-mph headwind would facilitate an easier liftoff. The reverse would prolong the takeoff roll. A 90° 30-mph wind gust could spell disaster.

Aside from the takeoff the next most dangerous leg of the mission is the final approach. The aircraft must be able to trim any sideslip encountered. If the conditions required, the aircraft needs to be capable of crowd avoidance.

All static and dynamic aerodynamic derivatives necessary to analyze longitudinal performance were estimated using methods outlined by Raymer[19], Roskam[21] and Etkin and Reid[8]. Those calculations were then incorporated into the team's GENESIS design code.

Horizontal Tail

In order to be scored as a horizontal surface and not as a "wing" the rules limit the span of the horizontal tail to 25% of the maximum wingspan. The overall wingspan of the aircraft is 9 ft. Thus a quarter of the wingspan would be 2.25 ft. As indicated in the rated aircraft cost equation, the total cost of any horizontal surface greater than 25% of the maximum wingspan would be more than 18 hours. The cost of anything less than 25% is 10 hours per horizontal surface. This results in a 2.5% increase in the total score by restricting the horizontal span. Multiple horizontal surfaces were not considered due to the expected increase in weight, complexity and manufacturing time.

Vertical Tail

Neglecting any effects from the wing-body, any vertical surface aft and above the c.g. will increase lateral directional stability. However, trim in yaw becomes difficult with extreme sideslip angles ($\beta > 45°$) that may be encountered during crosswind conditions. The pilot may have to crab into the wind on approach if such extreme crosswinds are encountered. Side force generated by the forward section of the fuselage will have to provide yaw control during the takeoff and landing rolls. The available side force of the fuselage was the limiting factor on the size of the vertical tail and was a function of the overall length.

Length

The overall length was also a factor in the rated aircraft cost. To determine the minimum length, the distance from the c.g. to the aerodynamic center of the horizontal tail was determined as a function of horizontal tail volume. The GENESIS design code then calculated the minimum overall length required to trim a 4-g pull-up with a given horizontal tail volume. The results showed an all-moving horizontal stabilizer would allow trim of a 4-g pull-up up to a static margin of 26%. This will allow for a large range of c.g. locations due to unforeseen changes in construction. However, the short coupling of the wing and horizontal stabilizer raised concerns. A sensitivity study concluded the overall length was insignificant in the score equation. A 1-ft increase in fuselage length results in a mere 2.2% decrease in the overall score. The emphasis on achieving trim at the minimum fuselage length possible was then restricted to a more conservative 30 to 40% elevator.

Flight Time

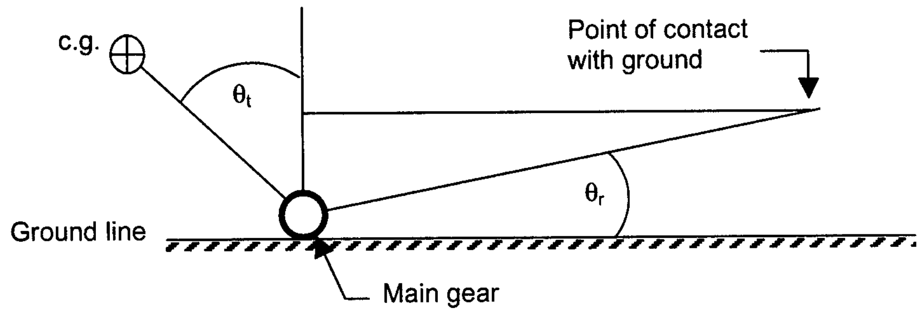
Obviously a major objective is to minimize the rated aircraft cost. However, even more important is to minimize the flight time of the aircraft. The flight time is a major factor in the score equation. Shaving a single minute off of the total flight time means a 15% jump in the score. Minimizing the flight time is a major concern. A great deal of time could be saved by any increases in performance. Take for example the turn radius. The mission requires three sorties each with two laps. Four of these laps require a complete turnaround in the downward leg. This results in ten complete circles. Taking these turns at a higher load factor will reduce the turning radius and thus the total distance of the course. However speed is lost in these turns and will have to be made up in the straight-aways.

Weights

The empty weight was not much of a concern. Aside from the 55 pound maximum and the extra required lift to take-off, a 1 lb increase in weight results in a mere 1% decrease in the score. However, a 1 lb reduction in the battery weight required results in a 17% leap in the score. Minimizing trim drag in the cruise condition is a major concern.

C.G.

In order to maintain stability and minimize variations in handling qualities between loaded and unloaded sorties, the center of gravity (c.g.) of the aircraft was fixed to coincide with the c.g. of the payload (24 softballs). This c.g. position was possible by using the batteries as a ballast to achieve any c.g. offset needed. However the physical dimensions of the fuselage limited this range to ±22 inches. This in turn imposed a maximum possible horizontal tail volume of 0.68 (with no weight from a vertical surface)



.**Figure 4.2.1. Tipback angle, θ_t verses rotation angle, θ_r.**

Take-Off

The required tail area for take-off rotation and the maximum rotation angle were also of concern. Both of these constraints were influenced by the tipback angle, θ_t as described by Raymer[1] (Figure 4.2.1.). To ensure the aircraft would not rest on the tail if it were somehow tipped backwards during rotation on take-off, the maximum rotation angle, θ_r was limited to 70% of the tipback angle. The rotation needed to achieve C_{Lmax} of the wing (10.5 degrees) was then set as the minimum rotation angle. This resulted in a θ_r of 16.5 degrees and a θ_t of 23.6 degrees.

Control Authority

To minimize trim drag during cruise for both the loaded and unloaded sorties the average angle-of-attack and elevator deflection for trim was restricted to ±1 degree off the point of minimum airfoil drag, -2 degrees and ±1 degree of deflection respectively. According to Roskam[2], any elevator deflection beyond 30 to 25 degrees would stall the horizontal tail. To maintain trim during a 4-g pull-up the maximum elevator deflection was limited to 20 degrees. To ensure elevator control during post stall the stall angle-of-attack of the horizontal stabilizer was required to be at least 5 degrees greater than that of stall angle-of-attack (10.5 degrees) while the elevator deflection required for trim at the maximum lift coefficient was limited to 15 degrees.

Performance Objectives

Performance objectives in one-way or another directly relate to maximizing the score. Take for instance stability. If the aircraft is not stable it will be inherently difficult to fly by remote control. This would most likely increase the flight time and most certainly increase the workload on the pilot. The same is true for the handling qualities of the airplane. An intense short period or rapid change in angle-of-attack due to a

small perturbation of the elevator may result in total loss of control. Great care must be taken during the design process to ensure optimal performance characteristics. Final sizing of the empennage and control surfaces were dictated by a modal analysis in the detailed design phase.

4.3. Propulsion

Propulsion preliminary design began with the identification of three main figures of merit:

1. Required power for all flight conditions (take-off, cruise, full gross, etc.),
2. Required current draw to meet the power requirements, and
3. Battery weight to supply sufficient voltage and total capacity required to complete a sortie.

Power Required

Calculation of required power for level flight was determined using simple aerodynamics. The aircraft parameters used are weight, wing area, aspect ratio, span efficiency factor, maximum lift coefficient, and zero lift drag coefficient. Using methods from Anderson[3], and assuming flight equilibrium (cruise), C_L and C_D were calculated varying V from zero to 100 feet per second. Once the actual drag is know for each speed, the thrust required to overcome this drag is simply equal to the drag. When thrust is multiplied by velocity, power required can be determined for level flight. This was repeated for both empty and full gross conditions.

Power for take-off was calculated using methods outlined in Anderson[3]. It consists of one simple equation:

$$T = \frac{1.44W^2}{dg\rho SC_{L,\max}} + D + \mu(W - L) \text{ where:}$$

T = thrust

W = weight

d = take-off distance

g = gravitational acceleration

ρ = density of air

S = wing area

$C_{L,max}$ = self explanatory

D = drag

μ = coefficient of friction in the wheels

Again, multiplying the calculated thrust by the velocity gives power required for take-off.

Current Requirements

One equation is necessary to calculate the required current. The parameters that need to be known are the power required for the given condition, the voltage applied to the motor, the no-load amps of the motor, and the internal resistance of the motor windings. Using the equation

$$P = (V - RI)(I - I_0)^2 \quad \text{where:}$$

P = power required

V = applied voltage

R = internal resistance

I = current

I_0 = no-load amps

Rearranging and solving, the current required can be determined for each flight condition.

Battery Selection

Based on the voltage required by the motor, and the total capacity required from the batteries, the correct battery configuration can be chosen. The capacity used by the aircraft over the entire three-phase course can be determined by multiplying the current drawn by the total time over which that current is being drawn. For example, for take-off, the motor draws 40 amps, and that condition lasts for 30 seconds. The total capacity required for take-off is 1200 amp-seconds. At cruise, the motor may only draw 20 amps, and the lap may last for 90 seconds. The capacity used for cruise is 1800 amp-seconds. All of these are summed for the entire flight to get a total capacity required. For all three phases, the total capacity required is 6120 amp-seconds. Batteries list capacities in milliamp-hours, so a simple conversion shows that 6120 A-s is equal to 1700 mA-h. This is a common capacity for NiCd cells.

Motor And Propeller Selection

Contest rules, for safety purposes, dictate that all teams must use a commercially available propeller that is completely unaltered with the exception of clipping the tips to a shorter length. Propeller selection, then, reduces down to comparing standard available model aircraft parts with a given motor in combination with expected flight speeds and thrust requirements. For preliminary design, the "rubber engine" method used in conceptual design was furthered until more accurate information was available about aircraft weight, drag, maneuverability, etc. The selection of a specific, commercially available motor would be postponed until the detailed design phase, assuming that one conforming closely to required specifications can be found. Also, a propeller would be selected similarly after the motor. For preliminary purposes, it was assumed, using statistical trends, that propeller with an 80% efficiency rating could be found to match any motor selected.

4.4. Structures

Wing Spar

To progress from the simple thin-walled tube frame used in conceptual design to a more realistic main spar, a number of reasonable assumptions were made at the onset. These included Linear Elastic, Isotropic, Isothermal Material Response assumptions. Also made was Euler-Bernoulli assumption stating planar sections prior to deformation remain planar and perpendicular to the centroidal axis. Further, to greatly simplify analysis and build in an additional factor of safety, the spar was sized to carry all loads alone, neglecting the contribution of the wing's foam core. This allows the use of very low-density and lightweight foam, and also avoids what would be a very difficult 3-D finite element analysis. However, it is assumed that the foam will sufficiently transfer all aerodynamic forces to the spar, located at the 1/4-chord.

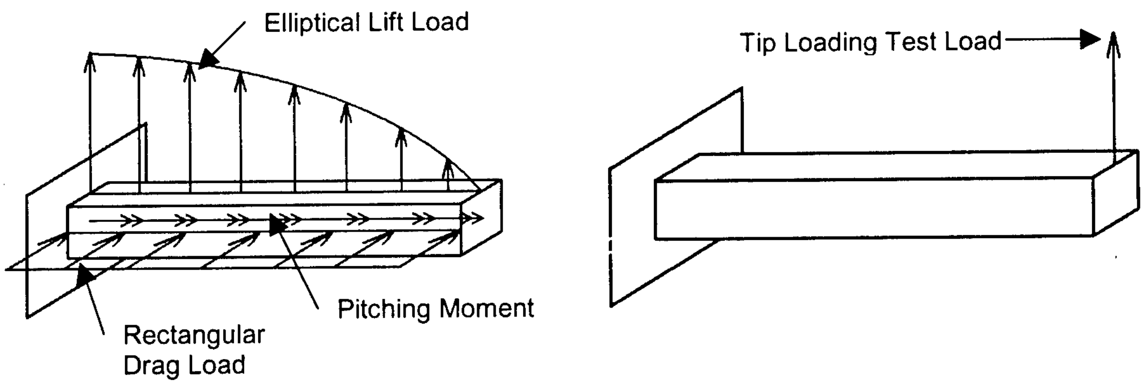


Figure 4.4.1. The two possible design loads for the wing.

The wing spar had to be designed for two specific and unique loadings, as shown in Figure 4.4.1. The first is the actual flight loads of lift, drag, and torsion due to pitching moment. The lifting load is modeled as a nearly elliptical distribution, which was derived by the LinAir aerodynamic analysis software. The lifting load is calculated to be full gross weight, neglecting any lift contribution from the fuselage, times a load factor of four. This 4g design condition was selected to include a healthy factor of safety. The second loading condition is the tip-loading which will serve as the test of structural integrity at the competition site prior to flight. This roughly simulates the force of a 2.5 load factor on the wing root, but is significantly different as it acts on the outboard wing sections. In outboard sections, the shear and bending from the point load at the tip will significantly exceed flight loads from a more-or-less elliptical lift distribution.

Composite materials were not considered for this project since the team lacks experience in working with them. A metal structure was also ruled out since jigging and riveting are time consuming processes and error-prone. Therefore for the small scale of this project, wood was determined to be the

optimum class of material. Basswood, spruce, and Douglas Fir were considered for the main load bearing elements. Douglas fir was rejected due to the unavailability, poor splitting characteristics and cost. Spruce was selected instead of basswood for several reasons. The strength of spruce is higher while the density remains the same. Spruce is also available in the local area in full wingspan lengths (9 ft span).

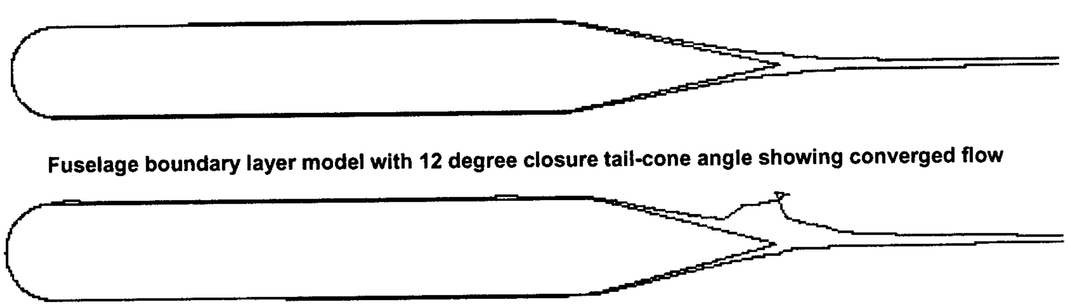
5. DETAIL DESIGN

5.1. Aerodynamics

After initial aerodynamic sizing was completed in the preliminary phase, the object of aerodynamic design efforts was to optimize performance by careful drag reduction efforts, as well as concurrently predict aerodynamic performance and key coefficients as accurately as possible. A sensitivity study using the GENESIS design tool from preliminary design showed that a change of 0.001 in total drag coefficient results in about a five-point change in final score, due to a flight time increase of about six seconds and additional batteries required. Obviously, careful drag management will be an important factor in optimizing the competition score.

Tail-Cone Taper Optimization

The type of drag to be most avoided, more so than friction drag, is separation drag. Unconverged flow, in major proportion, anywhere on the aircraft will have disastrous effects in aircraft performance. Aside from the wing, which has received much attention in this area, the aft end of the fuselage is the largest, most critical area where this is likely to occur. Here the flow is subjected to a strong adverse pressure gradient, which encourages separation. Therefore, careful attention must be applied in determining an appropriate angle that the flow can negotiate without causing separation.



Fuselage boundary layer model with 12 degree closure tail-cone angle showing converged flow

Fuselage boundary layer model with 15 degree tail-cone angle showing separated flow

Figure 5.1.1. Converged and non-converged flow on the fuselage dependent on tail-cone taper.

To perform this analysis, the software Xfoil[31] was used. Xfoil, very good at determining forces on a two-dimensional wing section, has with it an iterative boundary layer analyzer used to predict separation on an airfoil. In this case, rather than an airfoil, a series of 140 coordinate points was programmed that describe the longitudinal profile of the fuselage. This two-dimensional profile was then altered to reflect a range of taper angles on the aft-section, or tail-cone. Angle of attack was kept at a constant zero value, and analysis was done for cruise condition only, that portion of the flight profile where parasite drag is most important. A semispherical nose with the same diameter of the fuselage was assumed for the sake of this study, and the tail-cone was modeled as a perfect cone. No radius was applied to the juncture of the tail-cone and the fuselage. Reynolds number was recalculated to be 2.5x10^6, due to the added nose and tail length. The region of possible angles was determined by the region of possible empennage locations currently being studied by the stability and control section.

As the Xfoil boundary layer analysis clearly shows in Figure 5.1.1, at closure angles greater than 14 degrees, the flow is unable to converge at the trailing edge. The analysis used within Xfoil is unable to predict forces "post stall", but it is assumed that the drag is undesirably high. As seen in Figure 5.1.2, the slight change in friction drag is due to a slight change in wetted surface area when using cones of different aspect ratios, and in the portion of the fuselage surface that has a laminar boundary layer as versus a turbulent boundary layer. The trend in the change in pressure drag is to increase as it nears separation. The overall drag coefficient of the fuselage, as shown, is lower with a more gradual taper. At some point, the trend should reverse when the wetted area becomes so large as to overcome savings in pressure drag, but that reversal must not occur until the tail-cone becomes what practicality determines to be overly long. Results from this analysis are two-dimentional, but are conservative and should hold for a three-dimensional body as well.

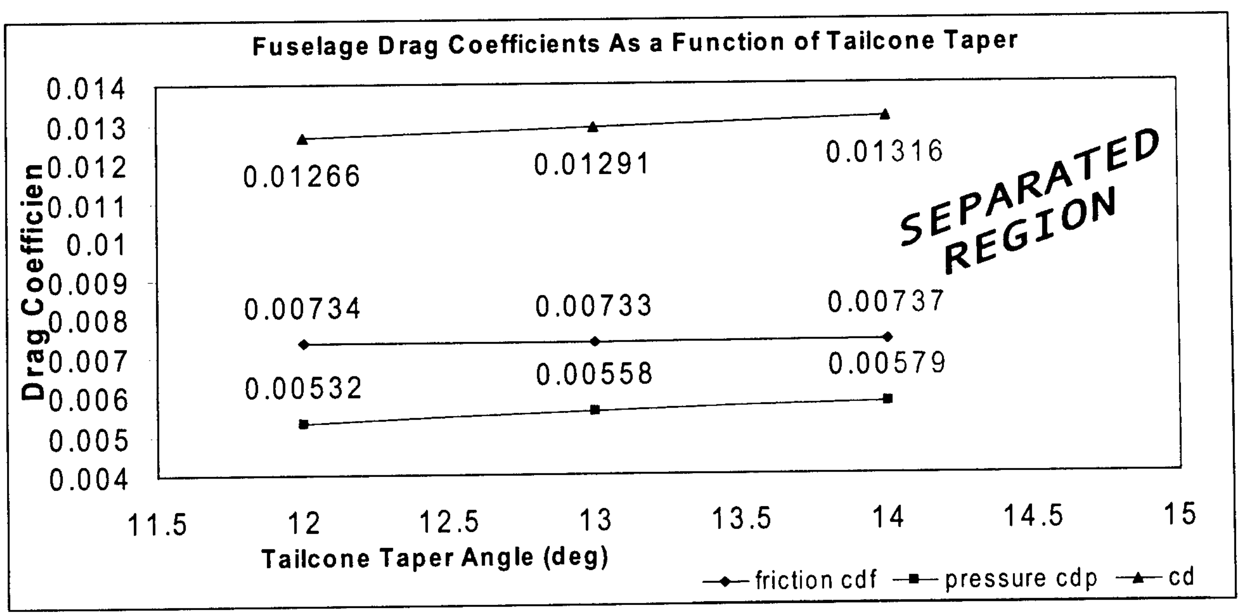


Figure 5.1.2. Fuselage 2-D drag coefficient change with tail-cone taper angle. Data from Xfoil.[31]

Final Aerodynamic Performance Predictions

As mentioned above, after drag reduction, the primary goal of aerodynamic design in the detail phase is to predict aerodynamic coefficients and other performance values as accurately as analysis tools allow. To achieve this, a whole-plane aerodynamic model was developed in LinAir Pro[15], a more advanced version of the software LinAir[14] used in wing sizing. Like LinAir, LinAir Pro idealizes an aircraft shape into a series of many small planar elements. Basic potential flow analysis is then performed on these elements and values calculated for any combination of angle of attack as well as side-slip. The result, though not as accurate as CFD or wind-tunnel testing, is claimed by LinAir developers to be within 10% of actual real-world values, which was deemed satisfactory considering the scope and time schedule of this project. In addition to fundamental limits taken directly from LinAir Pro, a plot of C_D over C_L was generated, and from this was found the $(L/D)_{max}$ as shown in Figure 5.1.3.

Basic Aerodynamic Limits				
CL_0	CD_0	Alpha Stall	CLmax	CDstall
0.448	0.0238	11 deg	1.428	0.0896

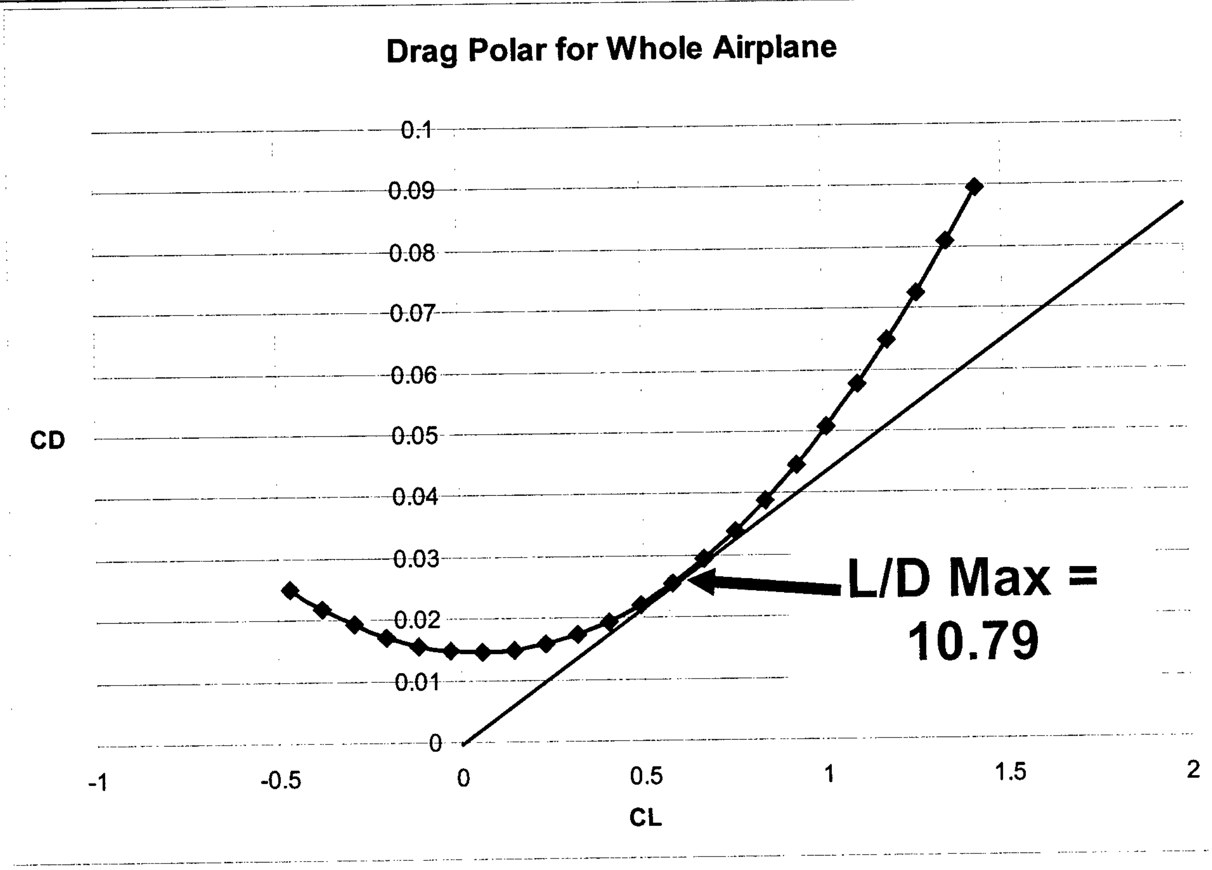


Figure 5.1.3. Total airplane aerodynamic characteristics. Data from LinAir Pro.

5.2. Propulsion

The success of the aircraft depends greatly on the efficiency and power of the propulsion system. In order to optimize the propulsion system for the DBF plane, the correct combination of propeller and motor must be selected. Motor selection was determined by the necessary power requirements calculated from, among other things, wing area, gross weight, take-off distance, and desired cruising speed. The motor was then simply selected from among a catalogue of highly efficient and readily available model airplane electric motors. The motor chosen for the job was the Astro 640G, being the smallest motor that minimally meets power requirements, minimizing weight and cost. In order to deliver this power, a propeller capable of meeting the thrust requirements must be matched to the motor.

Propeller Selection

As required by competition rules, the propeller must be unmodified, except for clipped tips, and commercially available. This limits the range of propeller designs and characteristics available for use. Model airplane propellers are typically not high precision products. They have a basic airfoil shape with sharp leading edges and are not balanced from the factory. Technical data, if available, is questionable due to inconsistencies in the manufacturing process (e.g. the pitch of a propeller could vary between props in the same lot and even between blades on the same prop)! These irregularities and lack of reliable technical information make detailed and accurate analysis difficult and at best, only a reasonably close estimate can be determined. Any results obtained will have to be flight tested to determine best performance.

Motor torque is a limiting factor in propeller size. Fortunately, the Astro 640 can be equipped with a gear reduction unit to address this problem. As the diameter of the propeller increases, the torque required also increases. Increasing the diameter of the propeller increases the thrust produced at the expense of motor torque and rpm. The maximum operating speed of the Astro 640S ("S" for super gearbox) is 6000 rpm. This limit must be considered when choosing a propeller.

Ground clearance must also be considered. Due to the overall size of the aircraft, and location of key structural elements, the landing gear height is limited. Currently, the gear gives approximately eight inches of ground clearance from the belly of the fuselage. This is close to the maximum length of the landing gear. Increasing the length would tend to decrease the ground handling stability creating the possibility of tipping. The location of the gear would have to be moved out into the wings for added stability meaning a re-design of the fuselage and wing spar. This is simply not an option at this point, so the gear height is treated as a constraint.

Finally, the power and thrust requirements for the most demanding modes of flight must be met. Take-off at full gross weight in 170 feet requires the most thrust at approximately 7.7 pounds continuous thrust.

Cruising at full gross weight comes in second requiring 3.5 pounds continuous thrust. Just meeting these requirements is not enough. A small amount of excess thrust, half a pound to one pound, is necessary to account for uncertainties in aircraft performance.

Assumptions

Model propellers are made primarily of three types of materials: wood, nylon, and fiberglass. Nylon props tend to be more flexible and for this reason, are not considered since aerodynamic loads on the prop can cause the prop to change shape, which is undesirable. Fiberglass props are very lightweight though not as available in all sizes. Wood props are perhaps the most common and the most available. For the purpose of optimization, wood propellers will be considered.

While model propellers come in a variety of materials, they generally have only two blades though occasionally there are three bladed props available. Due to availability, only two bladed props will be considered.

Diameter and pitch of model propellers come in standard sizes. Diameters increase by one inch when the overall diameter is less than one foot, and by two inches when the overall diameter exceeds one foot, though non-standard odd sizes do exist. Pitch can vary by one inch when the diameter is less than one foot and by two inches when the diameter exceeds one foot. Instead of calculating an optimum diameter and pitch, the best results will be chosen from available diameter/pitch combinations.

Pitch, for model airplanes, is defined as the number of inches the propeller advances through one revolution measured at 75% of the radius. As I stated before, the pitch can vary between props marked as having the same pitch and even between blades on the same prop. It is assumed that the propeller is designed "as advertised" and that one could find a prop with a pitch that closely matches its stated pitch.

The airfoil shape of model propellers only resembles an airfoil at best. The lower surface is flat with a noticeable but inconsistent curvature on the upper surface. To approximate this airfoil, a NACA 4412 was chosen. This does not match exactly, but is fairly close and since it will be used for each propeller diameter/pitch combination, a reasonable trend in performance can be determined. Choosing an exact airfoil shape provides critical data for determining propeller performance such as lift and drag.

Analysis

In order to analyze propeller performance, a program called Airscrew developed by David Ambrose[4] was implemented. It is a simple blade-element-theory program that approximates the performance of propellers and rotors. It is well referenced and uses equations no more complicated than those seen in Wichita State University Aerodynamic Theory, Propulsion, or Rotor Aerodynamics courses.

Inputs to the program include diameter, rpm, number of blades, pitch, flight speeds, and altitude. More detailed information about the airfoil sections is also required. Local chord, drag coefficients, maximum lift coefficients, and lift curve slopes for each section are needed to complete the calculations.

The flight speeds chosen for analysis were from zero mph up to 60 mph so as to encompass the entire flight envelope. An altitude of 1500 feet was also chosen as a good average between the local ground level of approximately 1350 feet and a maximum flight altitude of about 250 feet above ground level.

The program calculates the thrust produced in pounds, the shaft horsepower required to produce this thrust, the advance ratio, coefficient of thrust, coefficient of power, efficiency, speed power coefficient, and helical tip speed. The data of primary importance to the optimization are thrust, shaft horsepower, and efficiency. These factors will be used to determine which propeller will best produce the desired performance.

Standard sizes considered for analysis are 14x10, 16x8, 16x10, 18x6, 18x8, and 18x10. Larger sizes require taller landing gear and smaller sizes produce too little thrust. In addition, the larger sizes also require more power than the motor is capable of delivering. As seen in Figure 5.2.1, the input table to the right remains unchanged for each propeller. Only the diameter and pitch, and rpm if necessary to switch between geared and direct drive, are changed in the "Set Parameters" window. This assures that each

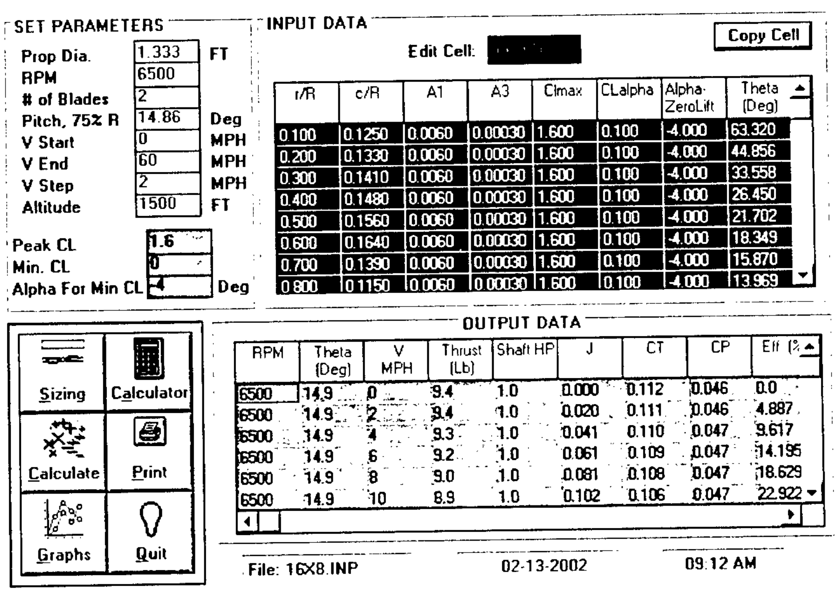
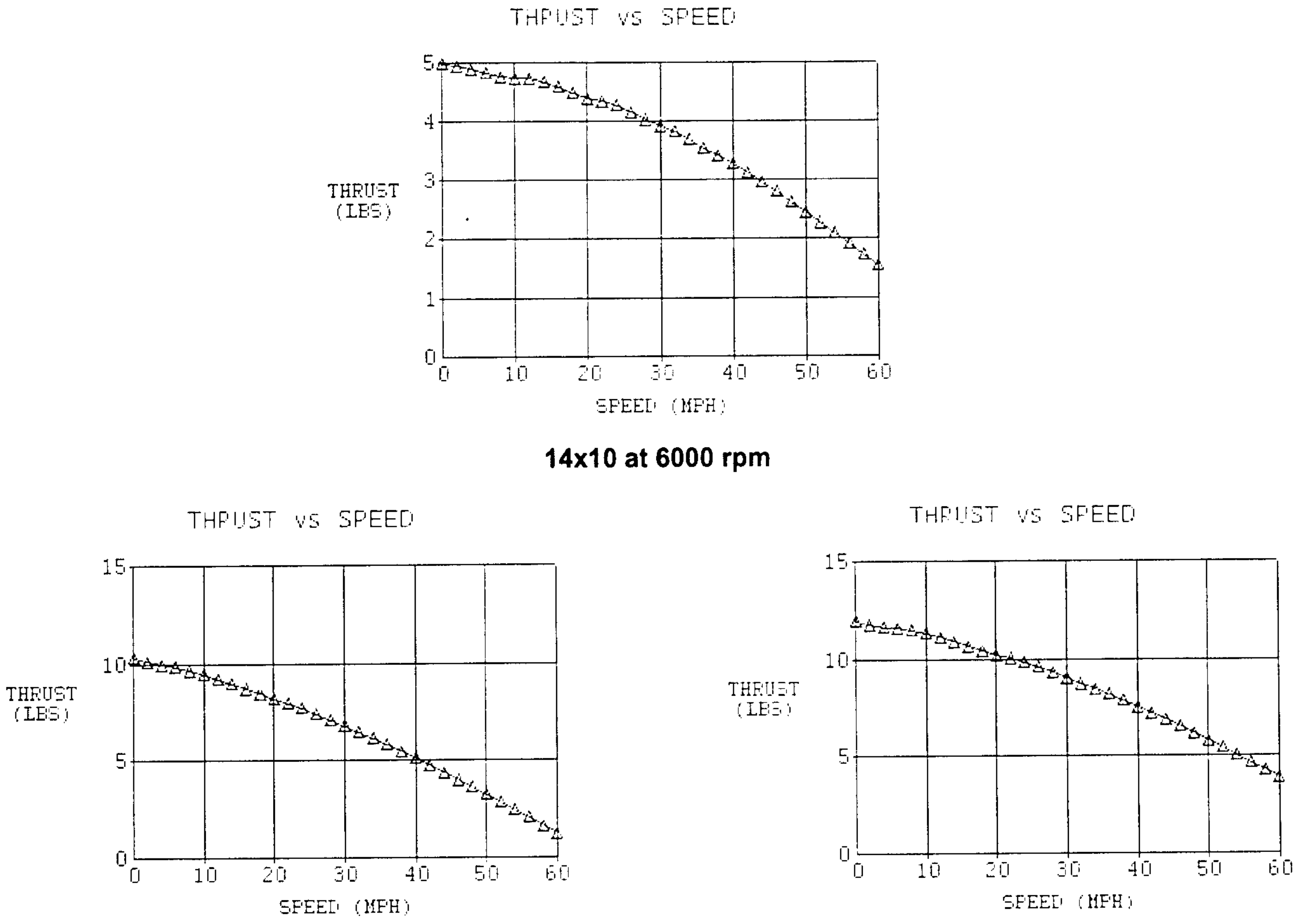


Figure 5.2.1. Airscrew propeller design program input screen.[4]

propeller is aerodynamically equal. The data generated is then plotted and compared to the required values of power and thrust for the aircraft.

Results & Discussion

The manufacturer of the Astro 640S, Astro Flight, Inc., suggests the best propellers for this motor are between 14x10 and 18x8. The results of this analysis show that the best propeller for this aircraft is actually an 18x10. The results for the 14x10 propeller show less static thrust and less required horsepower than the motor is capable of producing as seen in figure 5.2.2. Thrust and power are directly related to the diameter and pitch of the propeller. The smaller the diameter, the less thrust produced and the less power required. However, too much pitch and too large a diameter quickly overwhelms the capabilities of the motor. Furthermore, while increasing the diameter will increase the overall thrust, the pitch acts like the gears of a transmission in a car. A shallow pitch acts like a low gear providing good static thrust and good climbing characteristics, but limiting top speeds. A steep pitch performs best at high speeds but loses effectiveness at lower speeds. Selecting a good pitch for take-off may not be a good pitch for cruise. Since most of the flight will be spent at cruise, a pitch that favors this condition, but provides acceptable take-off performance, is desired. As seen in figure 5.2.2., this provides marginal thrust at take-off and cruise. The fact that the suggested propellers do not provide adequate performance is not surprising. These motors are commonly used on smaller aircraft that do not require large amounts of thrust. By increasing the pitch to 10 inches, better performance is achieved at take-off while also improving performance at cruise (figure 5.2.2.).



14x10 at 6000 rpm

18x8 at 6000 rpm

18x10 at 6000 rpm

Figure 5.2.2. Thrust versus speed curves for three propellers, data from Airscrew code.[4]

Conclusions

The recommended propellers for this motor do not provide adequate performance for the aircraft. Analyzing a 14x10 prop shows that it is insufficient to provide the necessary thrust for both take-off and cruise. Increasing the pitch from a 14x10 to an 18x8 provides the necessary thrust, but just barely, without any room for error. An 18x10 prop delivers sufficient thrust at take-off and cruise with a comfortable margin for error. Based on these calculations, an 18x10 propeller is the optimum choice for this aircraft. For actual flight-testing, a few of the recommended propellers will also be used to verify the calculated results. This analysis provides a good estimate of approximate propeller characteristics that will help narrow down the final propeller used for the competition. Due to many unknown factors inherent in model aircraft propellers, a more accurate result cannot be obtained without flight-testing.

Onboard Instrumentation

In order to fly the aircraft to its maximum potential, the pilot must be able to accurately judge the flight condition and react to changes. In R/C aircraft, the pilot depends on watching the aircraft from a distance and estimating flight performance. But because of the high demands of this competition flight, a higher level of accuracy is called for from the pilot. Therefore, a telemetry system will transmit airspeed, battery voltage, and battery temperature down to the pilot. The airspeed sensor consists of an impeller turning an encoder wheel. A data acquisition circuit will collect information and send it, via radio signals over amateur radio frequencies, down to a laptop PC running a custom-made virtual instrumentation program. The virtual instrumentation program graphically represents the data in a form easily readable by the observer, who will then verbally relay vital information to the pilot. This system will help to maintain a desired airspeed, which will aid in battery conservation. Monitoring battery voltage and temperature can help to warn the pilot of any abnormalities or malfunctions so that he can land the plane immediately should a problem develop.

Final Performance Estimates

The other goal of detail performance design is reasonably accurate performance predictions. From conceptual design stages, performance and its ultimate effect on score has been a driving factor in all areas of design. Coming full circle, after all drag, weight, and stability concerns are integrated into final design, the performance estimates were updated to be as follows:

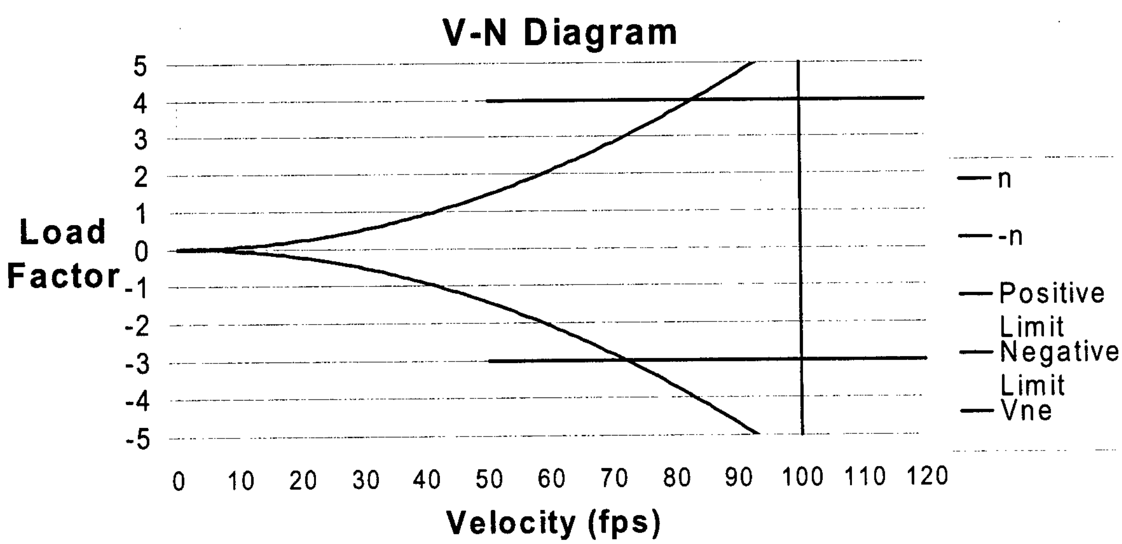


Figure 5.2.3. V-n diagram for final aircraft configuration.

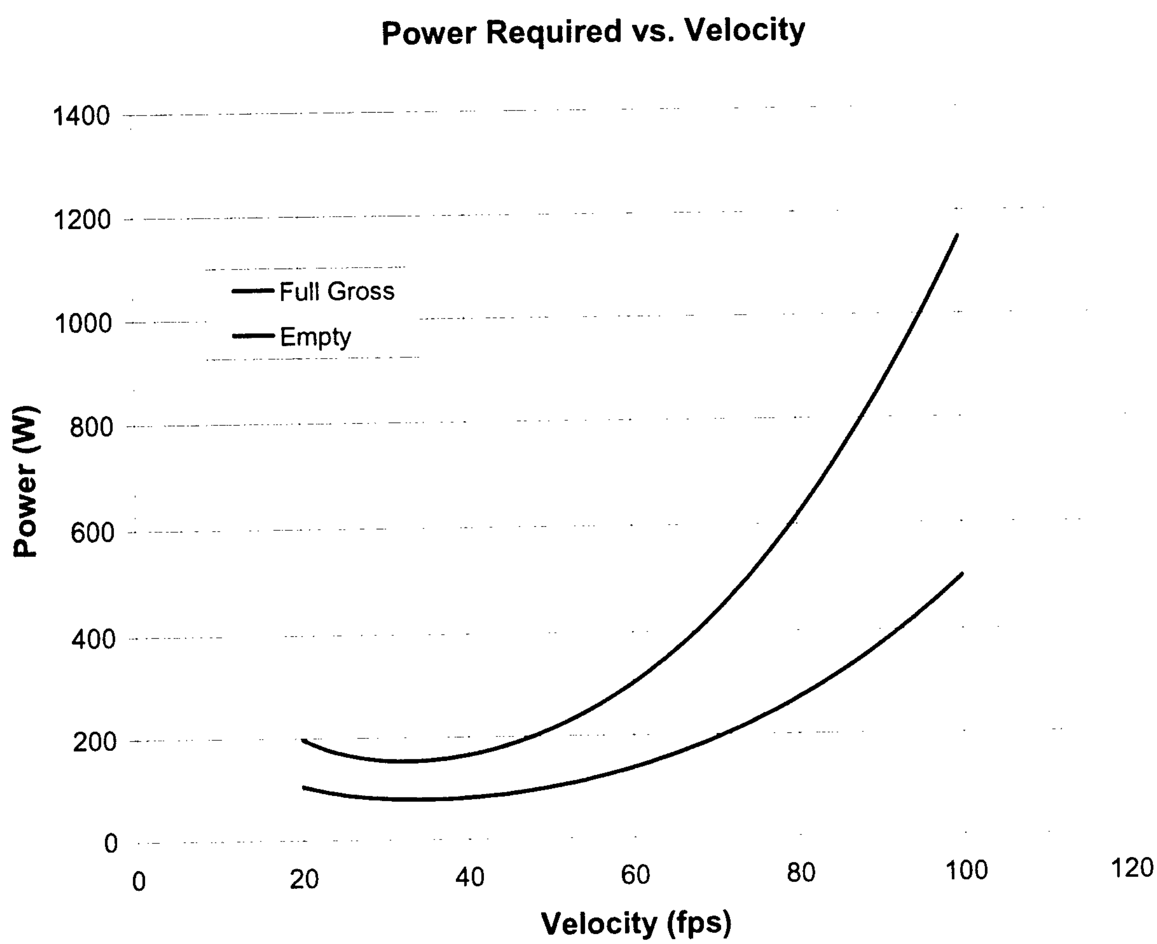


Figure 5.2.4. Power required versus velocity for final aircraft configuration.

Performance Characteristics (Empty)	
Take-off Power Required	200 Watts
Take-off Speed	26 mph
Take-off Distance	170 feet
Cruise Power Required	415 Watts
Cruise Speed	48 mph
Stall Speed	22 mph
Maneuver Speed	57 mph
Never Exceed Speed	68 mph

Performance Characteristics (Full Gross)	
Take-off Power Required	643 Watts
Take-off Speed	34 mph
Take-off Distance	170 feet
Cruise Power Required	448 Watts
Cruise Speed	48 mph
Stall Speed	29 mph
Maneuver Speed	57 mph
Never Exceed Speed	68 mph

Table 5.2.1. Final predicted performance values pertinent to piloting.

5.3. Stability and Control

Performance Analysis

All requirements, constraints, and goals listed above were then integrated into GENESIS for a brute-force method of optimization. Conditional formatting was used to alert the user of any constraints that were broken. A plot of the current configuration was instantly generated and used to perform visual "sanity checks" of the configuration. This visualization method proved to be quite useful and fun. Figure 5.3.1 is given as an example of a constraint violation of the location of the batteries. Note the extrusion out the nose.

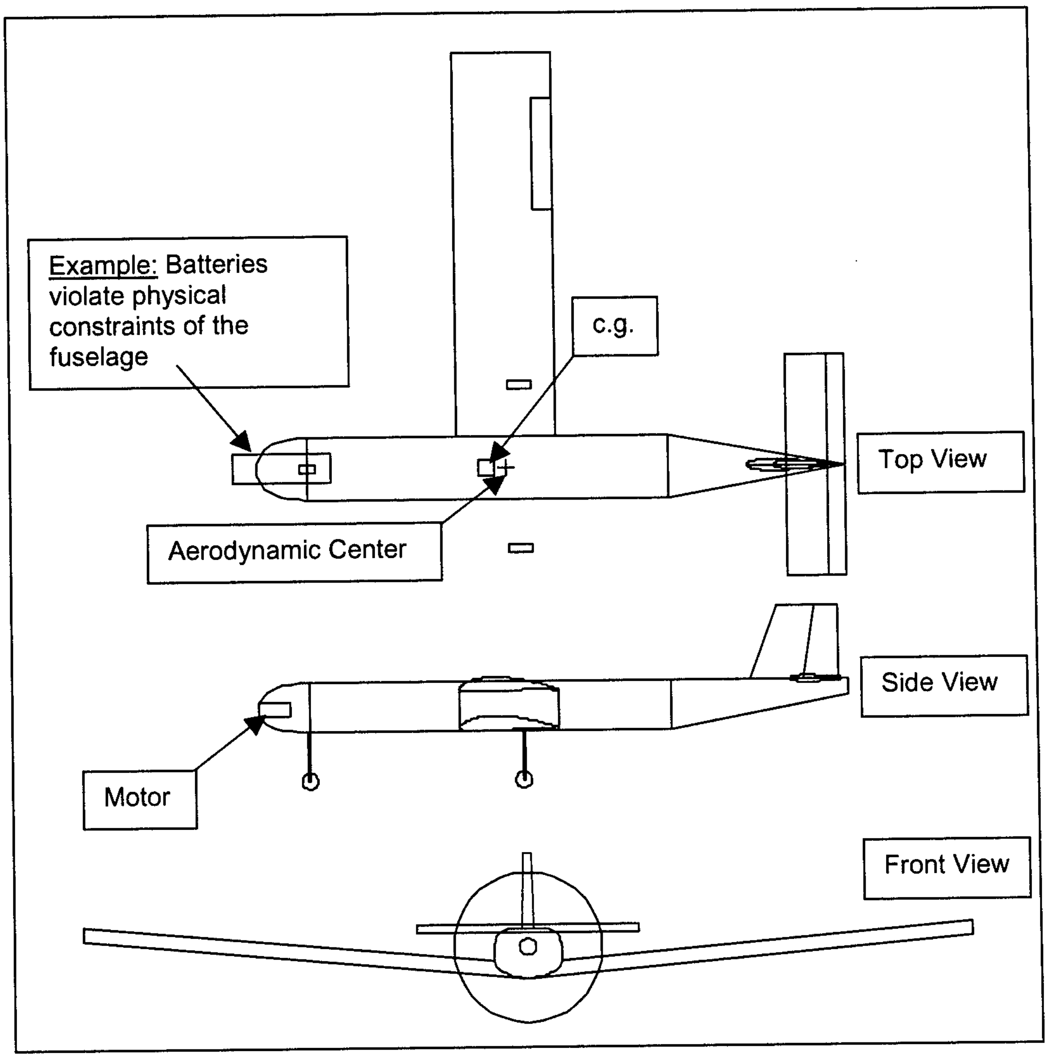


Figure 5.3.1. Configuration plot used for visualization during design, automatically generated by the GENESIS design tool. *Notice the constraint violation of the location of the batteries.*

The next step in the process was to evaluate the resulting trim diagram (Figure 5.3.2). A static margin of 0.20 and an over all length of 6 ft was chosen to distribute the control power. A conservative 31.2% of the horizontal tail was selected as the elevator to provide sufficient control effectiveness for a 4-g pull-up. The span of the horizontal stabilizer was set to 25% of the wingspan (2.25 ft) to minimize the cost to the score with a 0.60 ft chord giving an area of 1.35 square feet. This results in a average cruise angle of attack of -1.94 degrees and elevator deflection of 0.15 degrees. The elevator deflection required for a 4-g pull-up was -20 degrees while the stall angle-of-attack for the horizontal is 19.3 degrees. The vertical stabilizer was given a taper of 0.7 purely for esthetics with a root chord and span of 0.9 ft and 0.75 ft respectively. The total vertical tail area was 0.58 square feet. Raymer[19] suggests ailerons should not extend beyond the outboard 10% of the half-span, as the vortex flow due to the wing tips would produce little effectiveness. This set the outboard limit. The aerodynamics department has requested the inboard section of the ailerons be limited to 35% of the half-span relative to the wing tip. This in turn limits the span of the ailerons to 25% of the half-span. Following the historical guidelines for 25% half-span ailerons the range of chord ratios is from 0.17 to 0.20. Twenty percent chord ailerons where selected to provide a dimensionless roll rate of .12 with 10 degrees of aileron deflection at cruise speed.

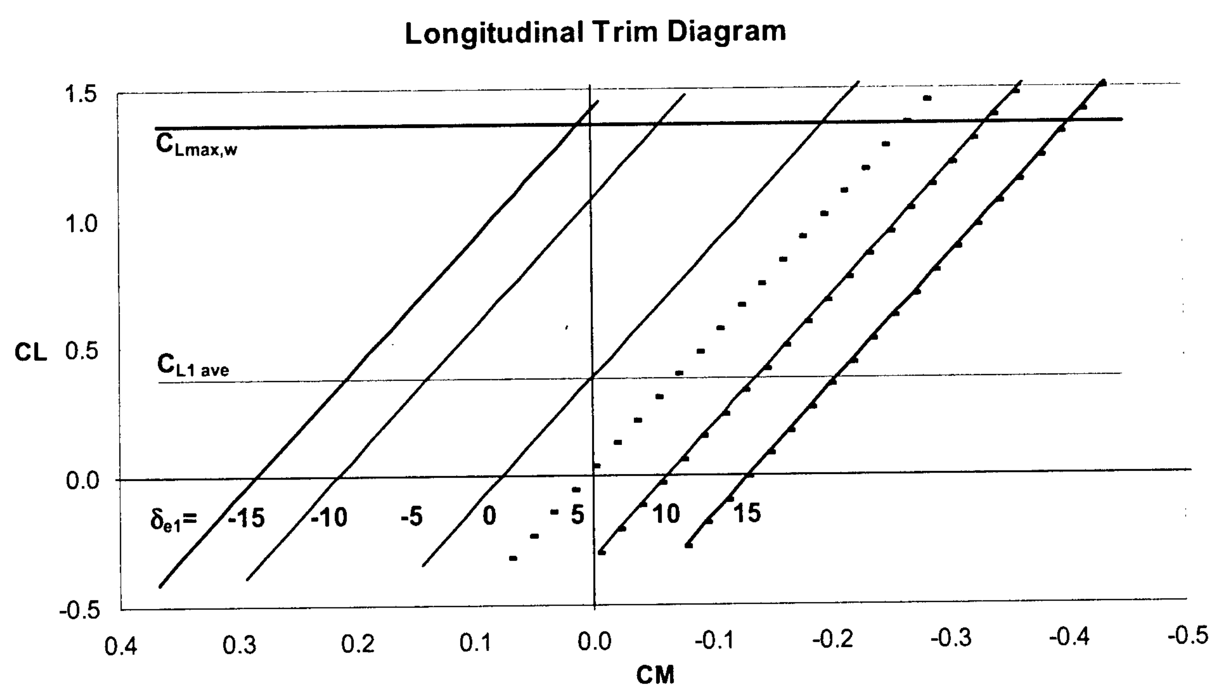


Figure 5.3.2 Longitudinal trim diagram.

Analysis of Handling Qualities

Handling qualities were estimated using methods outlined in Roskam[2]. A dynamic analysis of the modes was then conducted using MATLAB and SIMULINK to quickly view time responses. The changes in drag with the rate of change in angle-of-attack, change in pitch rate, and elevator deflections as well as the coefficient of side force with aileron deflections were assumed negligible during the analysis. Tables 5.4.1 and 5.4.2 show the results. The aircraft is stable in all modes. The Phugoid demonstrates the highest level of overshoot from a step response at 88% for the fully loaded cruise phase. This results in level 2 flying qualities for the Phugoid mode. The response is sufficient to accomplish the mission but will require some work on the part of the pilot to maintain a constant cruise speed. The telemetry system mentioned earlier should assist the pilot. The remainders of the modes consist of level 1 flying qualities and are adequate for completing the mission.

Longitudinal Coefficients and Stability Derivatives (Dimensionless)

C_{L0}	0.543	C_{M0}	-0.032	C_{D0}	0.020
$C_{L\alpha}$	5.085	$C_{M\alpha}$	-1.017	$C_{d\alpha}$	0.169
$C_{L\alpha dot}$	0.707	$C_{M\alpha dot}$	-2.257	$C_{D\alpha dot}$	0.000
C_{Lq}	3.160	C_{Mq}	-11.461	C_{Dq}	0.000

Longitudinal Control Derivatives (1/rad)

$C_{L\delta e}$	0.265	$C_{M\delta e}$	-0.848	$C_{D\delta e}$	0.000

Lateral-Directional Stability Derivatives (Dimensionless)

$C_{y\beta}$	-0.092	$C_{l\beta}$	-0.121	$C_{n\beta}$	0.021
C_{yp}	-0.007	C_{lp}	-0.523	C_{np}	-0.051
C_{yr}	0.063	C_{lr}	1.003	C_{nr}	-0.071

Lateral-Directional Control Derivatives (1/rad)

$C_{y\delta a}$	0.000	$C_{l\delta a}$	0.349	$C_{n\delta a}$	-0.032
$C_{y\delta r}$	0.055	$C_{l\delta r}$	0.175	$C_{n\delta r}$	-0.019

Table 5.3.1. Stability and Control Derivatives for cruise (Stability Axes).

Loaded Flights

Mission Phase	Mode	Dimensionless Eigenvalues	Undamped Natural Frequency (1/sec)	Damping Ratio (1/sec)	Time Constant (sec)
Cruise $\gamma = 0°$ V = 71 ft/sec	Short Period	-4.88 ± 6.80i	8.37	0.58	-
	Phugoid	-0.02 ± 0.64i	0.64	0.03	-
	Dutch Roll	-0.62 ± 2.24i	2.32	0.27	-
	Spiral	-1.64	-	-	0.61
	Roll	-18.80	-	-	0.05
Climb-out $\gamma = 8°$ V = 50 ft/sec	Short Period	-3.02 ± 4.11i	5.10	0.59	-
	Phugoid	-0.03 ± 0.90i	0.90	0.04	-
	Dutch Roll	-0.20 ± 1.82i	1.83	0.11	-
	Spiral	-1.94	-	-	0.51
	Roll	-22.27	-	-	0.04
Decent $\gamma = -4°$ V = 50 ft/sec	Short Period	-3.00 ± 4.07i	5.06	0.59	-
	Phugoid	-0.03 ± 0.91i	0.92	0.04	-
	Dutch Roll	-0.20 ± 1.84i	1.85	0.11	-
	Spiral	-1.98	-	-	0.51
	Roll	-23.50	-	-	0.04

Unloaded Flights

Mission Phase	Mode	Dimensionless Eigenvalues	Undamped Natural Frequency (1/sec)	Damping Ratio (1/sec)	Time Constant (sec)
Cruise $\gamma = 0°$ V = 71 ft/sec	Short Period	-8.68+8.99i	12.50	0.69	-
	Phugoid	-0.02+0.64i	0.64	0.04	-
	Dutch Roll	-0.37+2.50i	2.53	0.14	-
	Spiral	-2.93	-	-	0.34
	Roll	-17.05	-	-	0.06
Climb-out $\gamma = 8°$ V = 50 ft/sec	Short Period	-5.37+5.50i	7.69	0.70	-
	Phugoid	-0.02+0.91i	0.91	0.03	-
	Dutch Roll	-0.26+1.84i	1.85	0.14	-
	Spiral	-2.11	-	-	0.47
	Roll	-14.61	-	-	0.07
Decent $\gamma = -4°$ V = 50 ft/sec	Short Period	-5.36+5.49i	7.67	0.70	-
	Phugoid	-0.02+0.92i	0.92	0.03	-
	Dutch Roll	-0.26+1.84i	1.86	0.14	-
	Spiral	-2.12	-	-	0.47
	Roll	-14.67	-	-	0.07

Table 5.3.2 Dynamic modes for steady state straight-line flight.

5.4. Weight and Balance Worksheet

Before flight, it is vital to know that the aircraft is within its weight and balance envelope. The envelope is a plot of C.G. location on one axis and gross weight on the other. If any flight condition is outside the envelope, then the craft becomes unstable. Measures such as ballasting can be implemented in this event. Figure 5.5.1 shows envelope of range of acceptable C.G. locations measured in percent of the mean aerodynamic chord from the wing leading edge aft. Maximum gross weight is defined to satisfy the 200-ft take off requirement. Points indicate calculated weight and balance for each of two flight profiles.

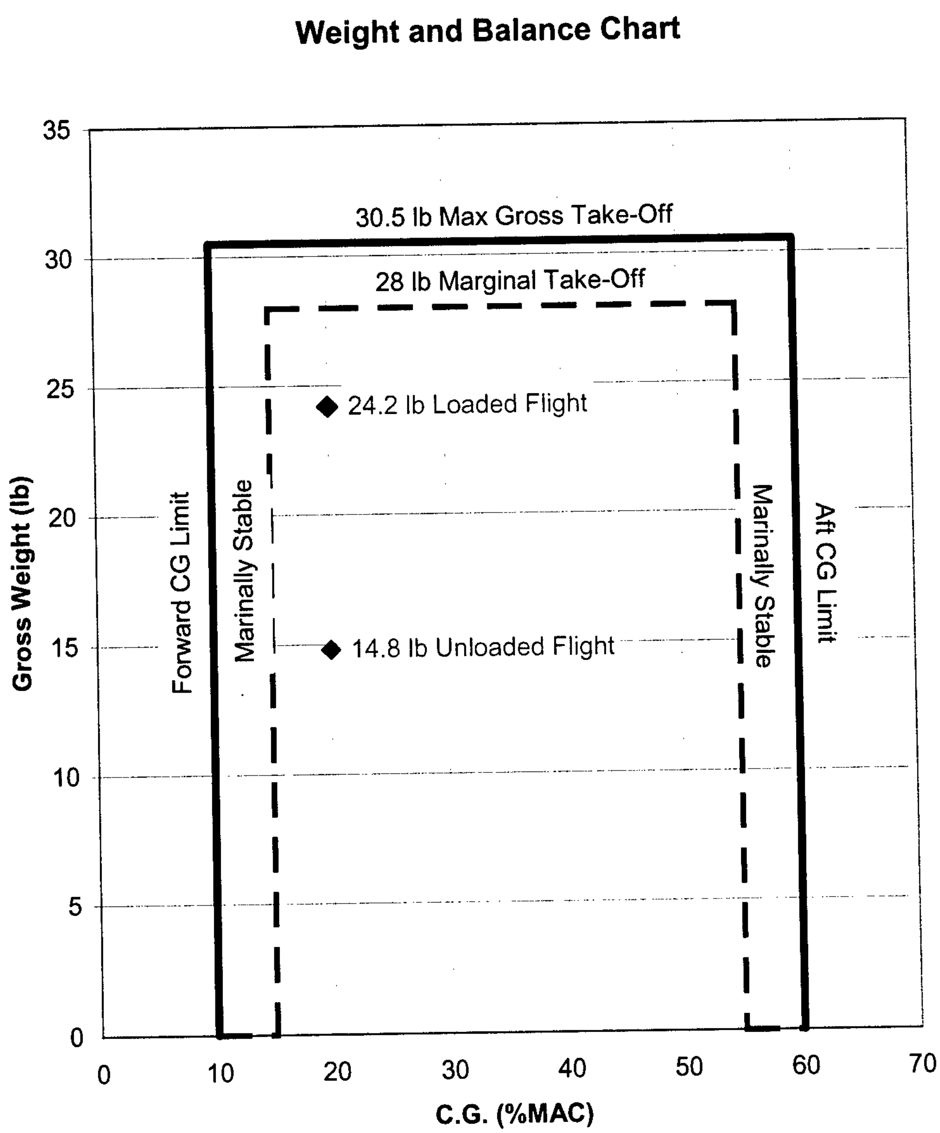


Figure 5.4.1. Weight and Balance Chart.

5.5. Structures

Material selection, cross-sectional shape, and tapering methods were investigated in order to find the most efficient way to reduce structural weight while maintaining structural integrity. The primary design requirement is that the spar must be capable of acting out all of the loads. The foam core of the wing will be capable of assisting in the minimization of the angle of twist due to the torsional load created by the pitching moment. Ultimately, the angle of twist encountered by the wing must be as close to zero as possible. The limit on angle of twist was established to be +/- one degree. The spar must also be easy to manufacture in order to minimize the amount of time needed for construction and the level of skill necessary to produce an acceptable spar. Another stipulation imposed by the geometry of the airfoil is that the spar height must be no more than one inch. If the height of the spar was more than one inch, the depth of the spar is constrained to levels where in some cases a moment of inertia cannot be found to react bending loads without exceeding the boundaries of the airfoil. Material selection is sometimes critical when minimizing weight in the wing spar. In this optimization spruce, basswood, and fir were analyzed.

Spar tapering is also an effective way of reducing weight. Three different methods of tapering were also investigated: sectional tapering, a step-down method of tapering, and no taper. In all methods, the height of the spar was held constant at one inch. In the sectional tapering method, the depth of the spar is calculated from the multi-axial bending equation. The spar was divided into two hundred pieces and the depth was determined to be the minimal depth capable of handling the combined bending in that section of the wing. In the step down method, section cuts were taken at foot intervals along the span and the depth was determined to be the minimum depth necessary to act out the maximum combined bending moment in that section. Also four basic cross-sectional shapes were analyzed: a solid rectangular cross-section, an I-beam, a hollow box-beam, and a solid circular cross-section (see Figure 5.6.2).

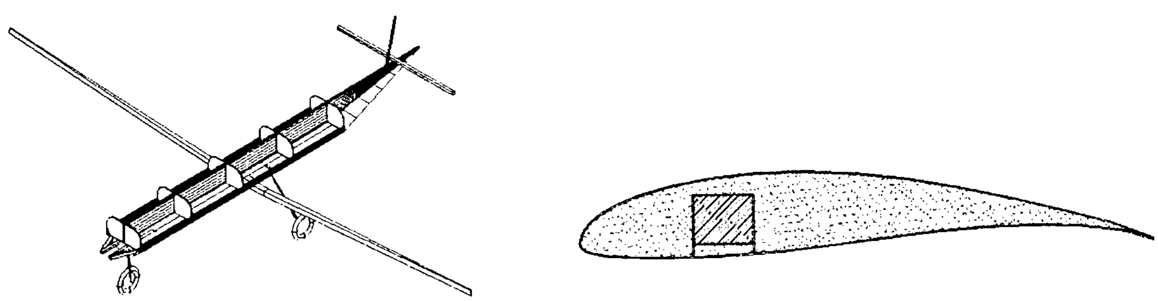


Figure 5.5.1. Overall structural configuration. The left shows the entire airplane main load-bearing elements. The right shows the wing cross-section with the spar. Larger views follow in Drawing Package section.

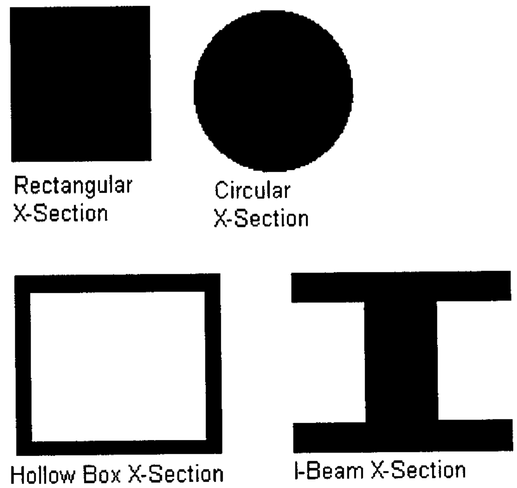


Figure 5.5.2. Different cross-sectional shapes that were analyzed.

Many more exotic cross-sectional shapes exist. The four cross-sections shown above were investigated for a variety of reasons. The rectangular cross-section was chosen for analysis because it may be the simplest to build. However, there was some concern about its ability to react torsional loads. The circular section was selected to investigate its bending load bearing capabilities. Circular cross-sections are known to do very well in torsion, however the circular cross-section often performs poorly in bending situations when compared to a typical rectangular beam. The I-beam and box beam cross-sections were dismissed from consideration due to the difficulty of construction at this scale.

During this optimization exercise, a nine-foot wing with a foot long chord was subjected to multi-axial loading. A new coordinate system was established for the structural analysis of the wing spar. The x-axis points outboard along the spar's centroid from the fuselage. The y-axis runs parallel to the lifting load and points upwards from the fuselage. The positive z-axis points opposite the drag. This system was established in order to simplify the evaluation of loads and stresses acting on the spar.

A roughly elliptic 4g lifting load, shown in Figure 5.6.3, was predicted during flight, however the team must also simulate a 2.5 g tip loading condition by raising the plane by the wingtips off the ground in order to prove structural integrity. The two conditions were analyzed at 200 different sections along the wing. The resultant bending moment diagrams are compared in Figure 5.6.4. The maximum moment generated due to lift at each location is shown in Figure 5.6.5. The loads shown in Figure 5.6.5 were used as the design loads. It was shown that the moment created by the lifting load was dominant on the inboard third of the wing, while the moment generated by the tip loading condition dominated on the outboard sections. In order to increase the effect of the lift on the structural design, wing weight was neglected since it opposes lift. All lifting loads were evaluated by using the lift produced when the coefficient of lift of the wing was a maximum and the velocity was at 83 ft/s, which was determined to be the maneuvering speed.

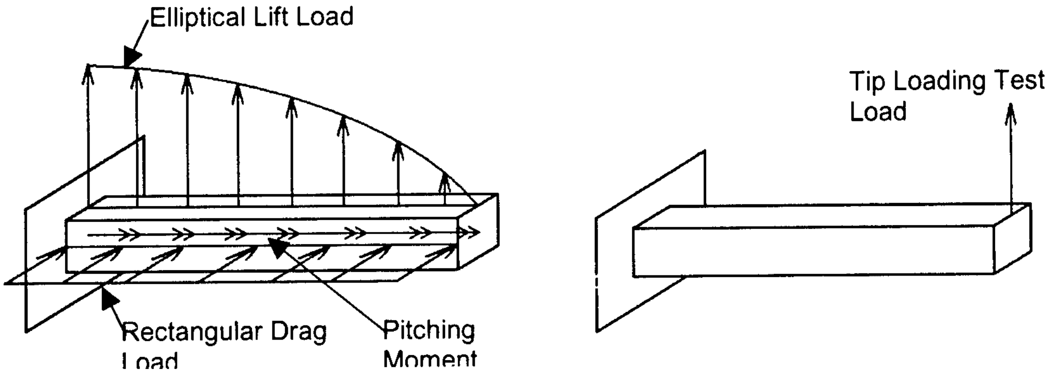


Figure 5.5.3. Two expected loading conditions for the main wing spar. The left shows flight loads, and the right shows tip-loading for structural testing at competition.

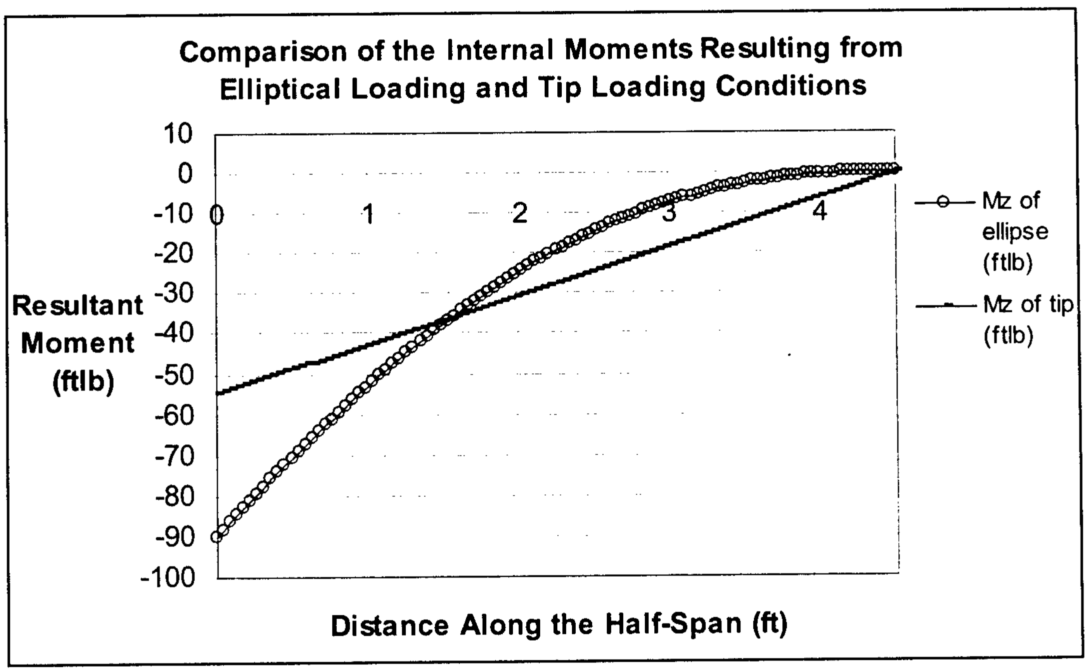


Figure 5.5.4. Maximum bending moment due to lift as a function of span-wise location across the half-span.

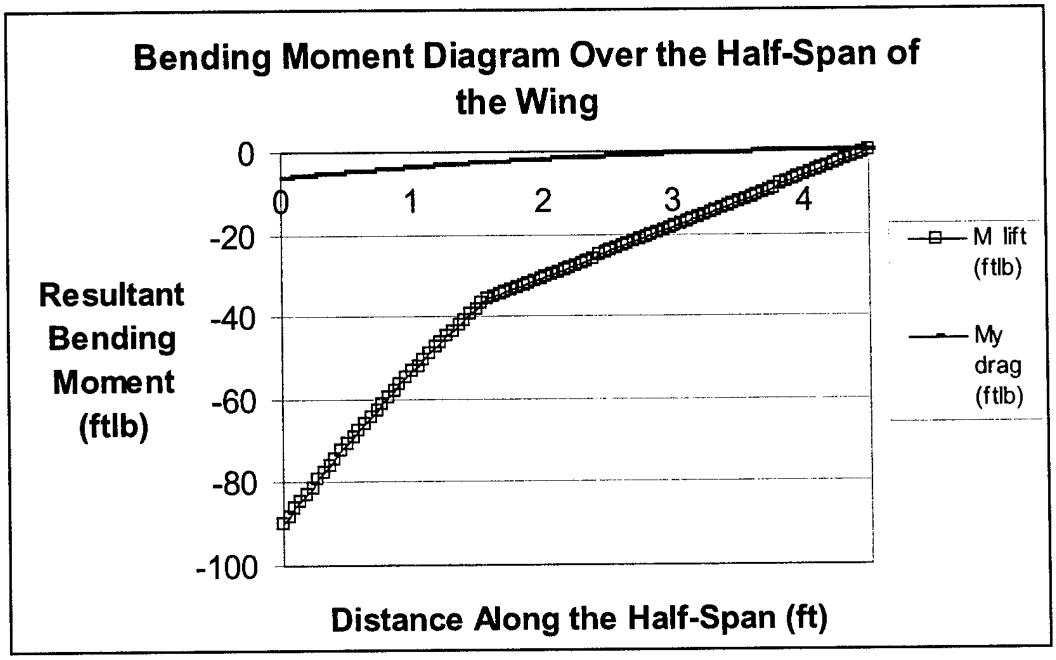


Figure 5.5.5. Maximum bending moment as a function of span-wise location across the half-span, resulting from the maximum lifting moment selected from Figure 5.6.3 and the rectangular drag load.

A section was established in GENESIS in order to expedite calculations and minimize computational errors. The bending equation shown below was used to determine the depth of the cross-section.

$$\sigma := \frac{(-Mz \cdot y)}{Izz} + \frac{(My \cdot z)}{Iyy}$$

σ = The bending strength of the material

Mz = The maximum moment resulting from the elliptical lifting load and the tip loading condition at a given station along the spar.

My = The moment resulting from the drag load at a given station along the spar.

y = The location along the height of the spar to be analyzed. In the case of maximum bending stress, y would correspond to the upper or lower surface of the spar.

z = The location along the depth of the spar to be analyzed. In the case of maximum bending stress, z would correspond to the most forward or aft chord-wise portion of the spar.

Iyy = The moment of inertia about the y-axis. This term is a function of the depth.

Izz = The moment of inertia about the z-axis. This term is a function of depth.

Each cross-sectional shape was analyzed using the function shown above. The three tapering methods were also implemented on each cross-sectional shape. An overall weight of the nine-foot spar was calculated for each combination of shape and tapering method.

The material study was only performed on the rectangular cross-section. The material properties of spruce, fir, and basswood were applied to all three sectional tapering methods and the weight was determined for each resulting spar. The finding was that the basswood spar was heavier than the spruce spar. Availability of large basswood strips was also a problem. For these reasons, spruce was selected as the material to be used in the spar. It is the intention of the team to use a lamination technique in order to improve the strength of the spar. This additional strength will serve as an additional factor of safety. This decision was made since the wings will likely be the most time consuming pieces of the plane to rebuild, in the event of damage. Over designing the wing might cost in weight, but the benefit of not having to reconstruct the wings after a rough landing may prove to be well worth it. For this reason, the other spar shapes were analyzed only with spruce properties.

Three tapering methods were investigated in this optimization study. It was found that the sectional taper consistently offers the smallest cross-sectional depth. For this reason, one can predict the weight of the sectionally tapered spar to be the least. Due to the savings in weight as well as smaller cross-sectional size, the sectional taper method was selected by the group for implementation.

The only decision left to make turned out to be the most difficult. Which sectional shape should be chosen for the Design, Build, Fly plane. Three aspects of each shape would ultimately be the deciding factors: depth, weight, angle of twist of the spar. The spar was sized in order to react all bending loads and keep the angle of twist to less than one degree. In Figure 5.6.6 the resulting depths of the rectangular and circular cross-sections are compared. The weights of the resulting spruce spars are compared in Figure 5.5.7.

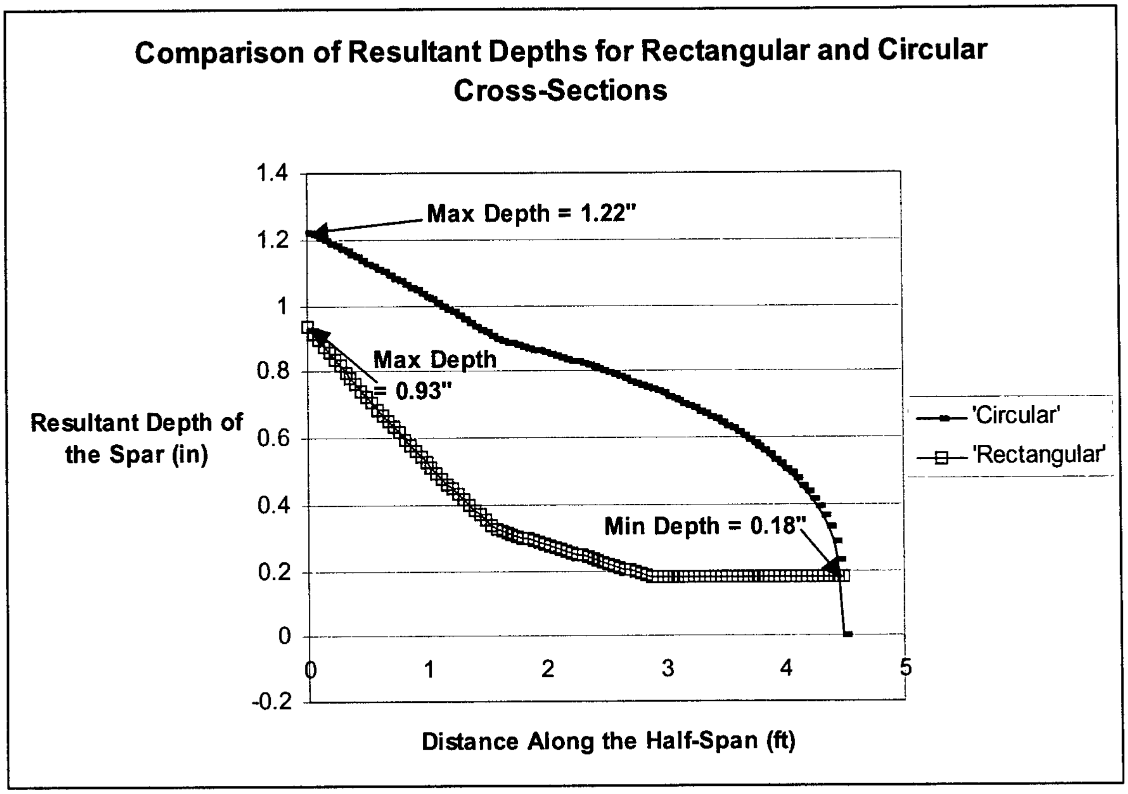


Figure 5.5.6. Derived spar sizes to meet loading demands using circular and rectangular section beams.

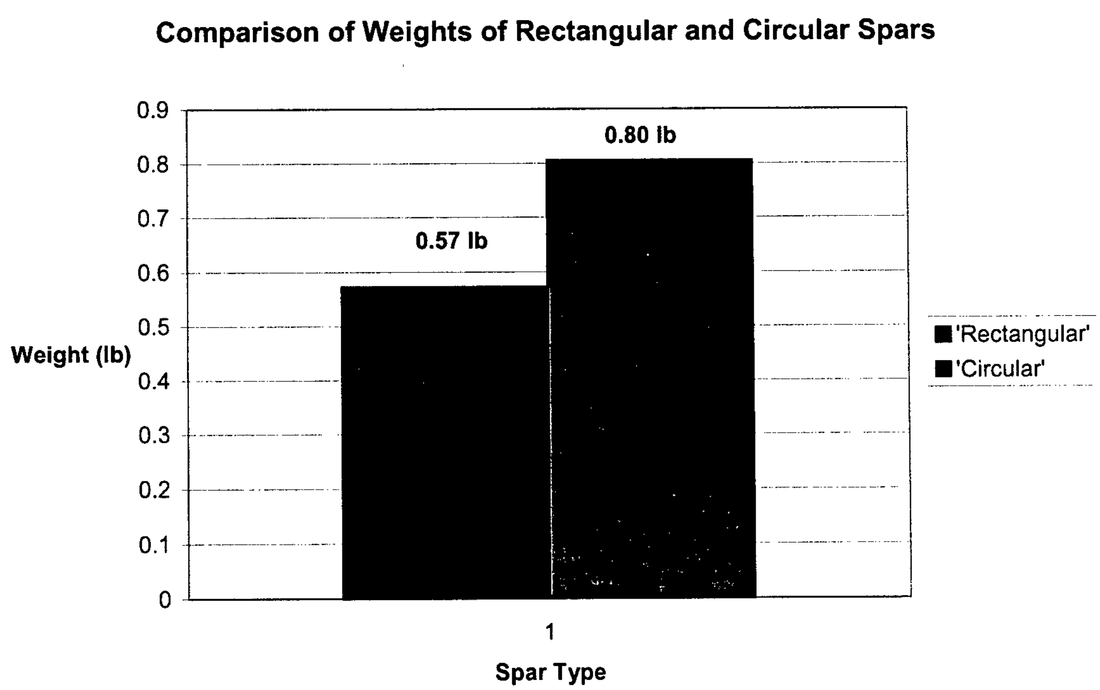


Figure 5.5.7. Weights for main wing spars using circular and rectangular section beams.

When considering the horizontal tail, a rectangular lifting load was analyzed with a rectangular drag load as illustrated in Figure 5. The resulting bending moment diagram is shown in Figure 6. As with the wing, the pitching moment was assumed to be evenly distributed along the centroidal x-axis. The values of lift, drag, and pitching moment were analyzed with the elevators at a maximum deflection of +/- 20 degrees. The speed during this analysis was assumed to be the maneuvering speed. The distributed torque was determined to be the most critical condition, thus the design condition since an angle of twist of less than +/- 1 degree must be maintained. The hinge of the elevator was also analyzed to assure that it could handle the moments generated by the elevator. This was accomplished by analyzing the drag on a flat plate perpendicular to the free-stream, with the reasoning that if the hinge could withstand the drag perpendicular to the free-stream, it would obviously be capable of withstanding the drag at maximum elevator deflection.

The fuselage will be constructed of an inverted T-beam for bending and a combination of lower stressed skin and triangular sill configuration for torsional rigidity. The thickness of the webs in the T-beam was determined to be 1/16" from the results of the multi-axial bending equation discussed above. The loads used to determine the internal reactions were the lift and weights of each component at 4gs and the drag on a flat plate resulting from a thirty mile per hour ninety-degree crosswind. The maximum internal moment occurs just fore of the aerodynamic center of the wing. The torsional loads occur from the torque of the motor and the side force on the vertical tail. This results in a 27.2 inch lb torque. When just analyzing a 1/32" thick stressed skin under-fairing, the resulting angle of twist is 0.000116-degree rotation. The area of concern results where the undercarriage of the spar interrupts the stressed skin. The presence of the spar alone results in a two-degree angle of twist over the inch wide cutout. This results in a displacement of 0.031 inches. This was deemed an acceptable amount of twist on the fuselage. The sills discussed above will add additional torsional rigidity, but their primary purpose is to restrain the motion of the balls in the cargo compartment. The landing gear is composed of a material with a maximum deflection of 1.5 inches under a 25 pound landing load.

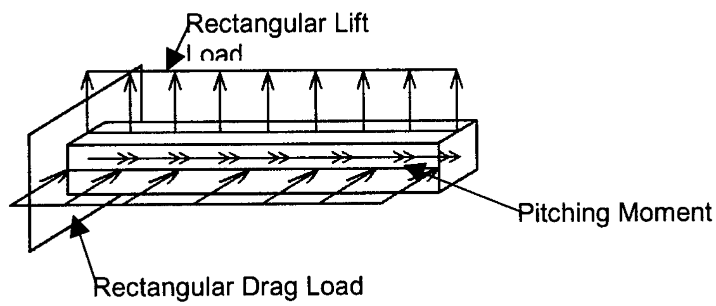


Figure 5.5.8. Analyzed loads for the horizontal stabilizer spar.

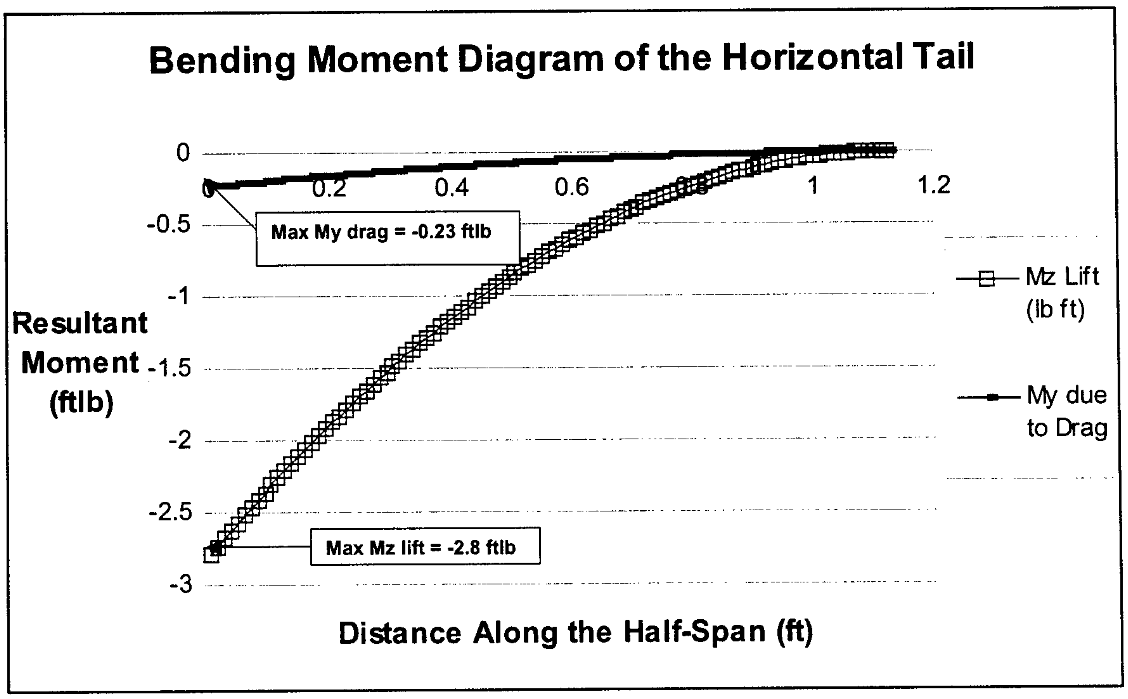


Figure 5.5.9. Bending moment diagram for the horizontal tail.

5.6. Rated Aircraft Worksheet

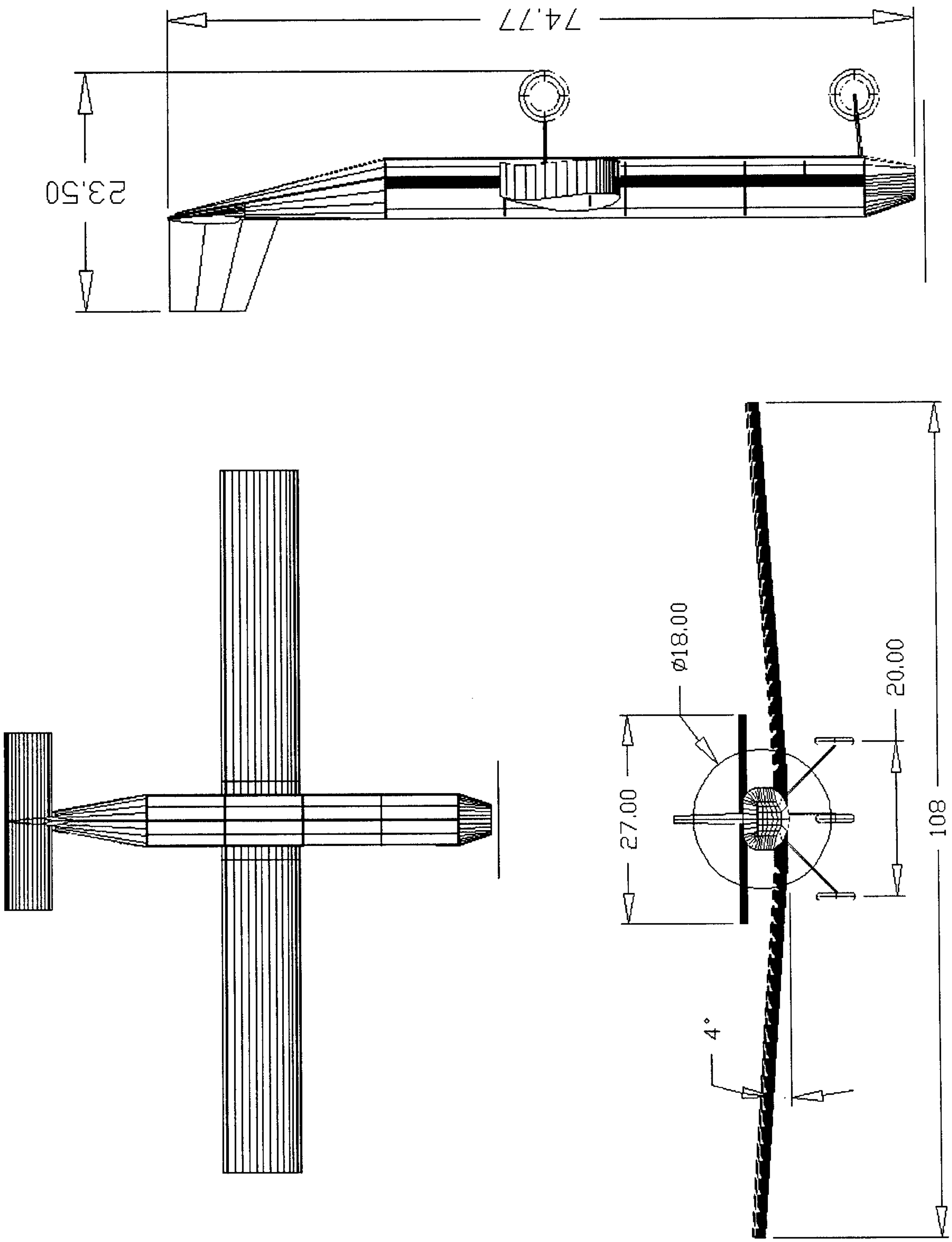
In the conceptual design phase, it was the rated aircraft cost which was the key element in selecting a basic configuration. Throughout preliminary and detailed design, that configuration was upheld by the conservative approach and simplicity-first principals established at the beginning. As the design is finalized, it can be shown that the rated aircraft cost has increased only a small fraction, and performance estimates have become even more favorable. Best final estimates predict the score to be 4.38 for each single flight score, and a rated aircraft cost of only 8.84. Note that at the time of this report, construction is not complete, so the Empty Weight term in the RAC worksheet (Table 5.6.1) is the best calculated weight rather than a measured value.

Rated Aircraft Cost	**8.838**
Manufactures Empty Weight	15.038 lb
Empty Weight	15.038 lb
Material Multiplier	0.00
Rated Engine Power:	2.143
Total Battery Weight:	2.143 lb
Number of engines:	1
Manufacturing Man Hours	206.000 hrs
1.0 Wing WBS:	86.000 hrs
Wingspan:	9.000 ft
Wing chord:	1.000 ft
Control surfaces	2
2.0 Fuselage WBS:	60.000 hrs
Fuselage Length:	6.000 ft
3.0 Empenage WBS:	20 hrs
Vertical without ctrl:	0
Vertical with ctrl:	1
Horizontal with ctrl:	1
Vtail	0
4.0 Flight Systems	30 hrs
Servos and controlers:	6
5.0 Propulsion WBS:	10 hrs
Number of engines:	1
Number of Props:	1

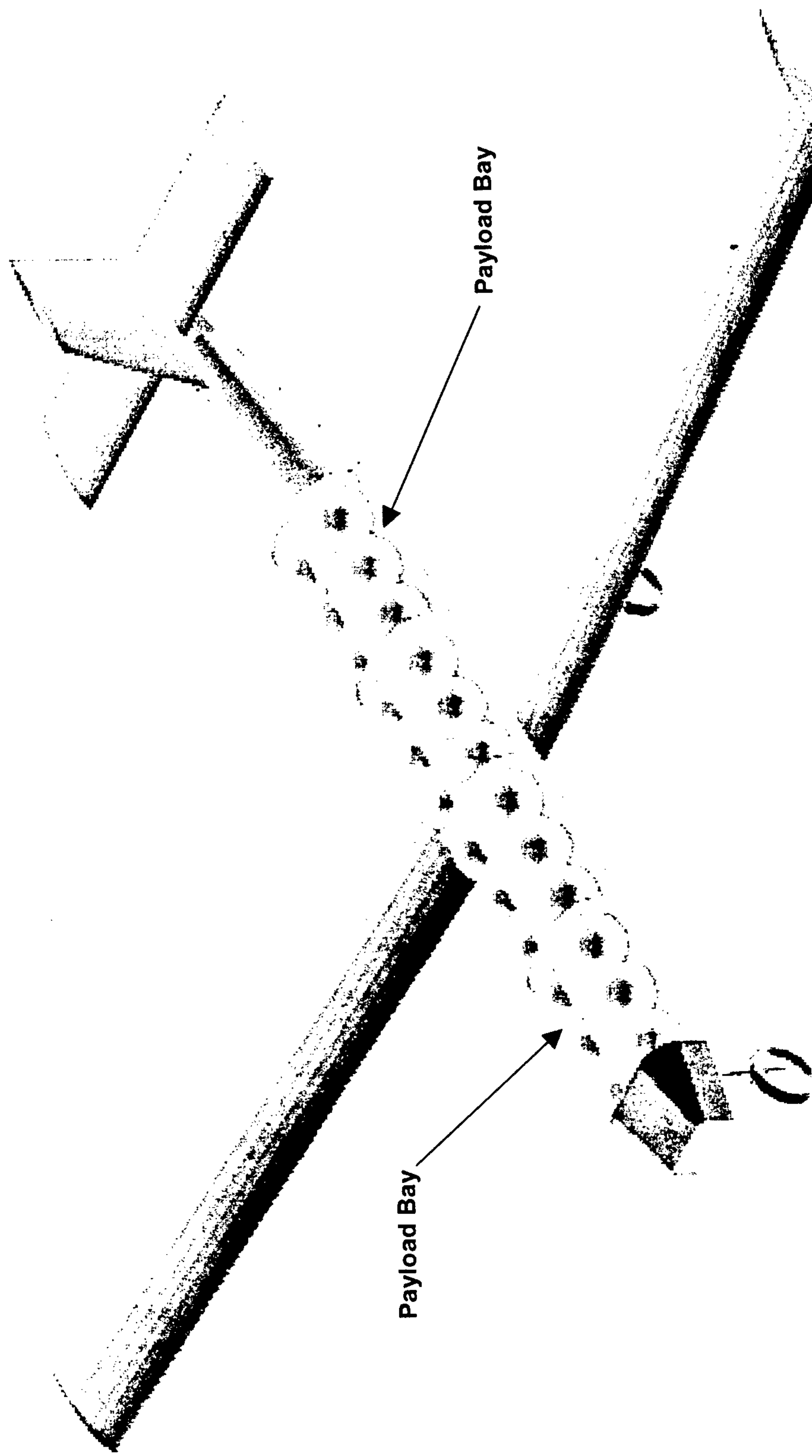
Table 5.6.1. Rated Aircraft Cost worksheet.

5.7. Drawing Package

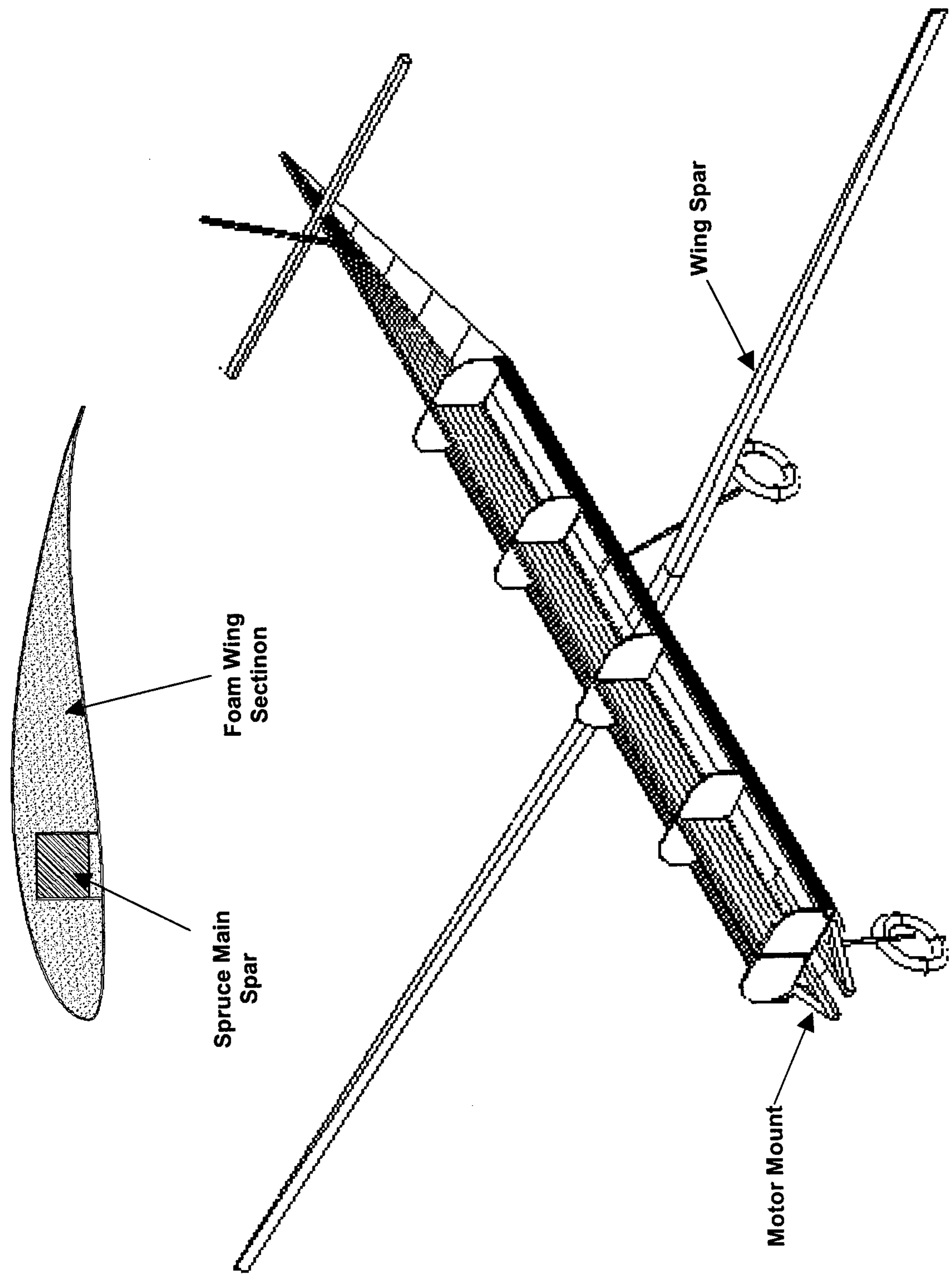
Three-View With Dimensions



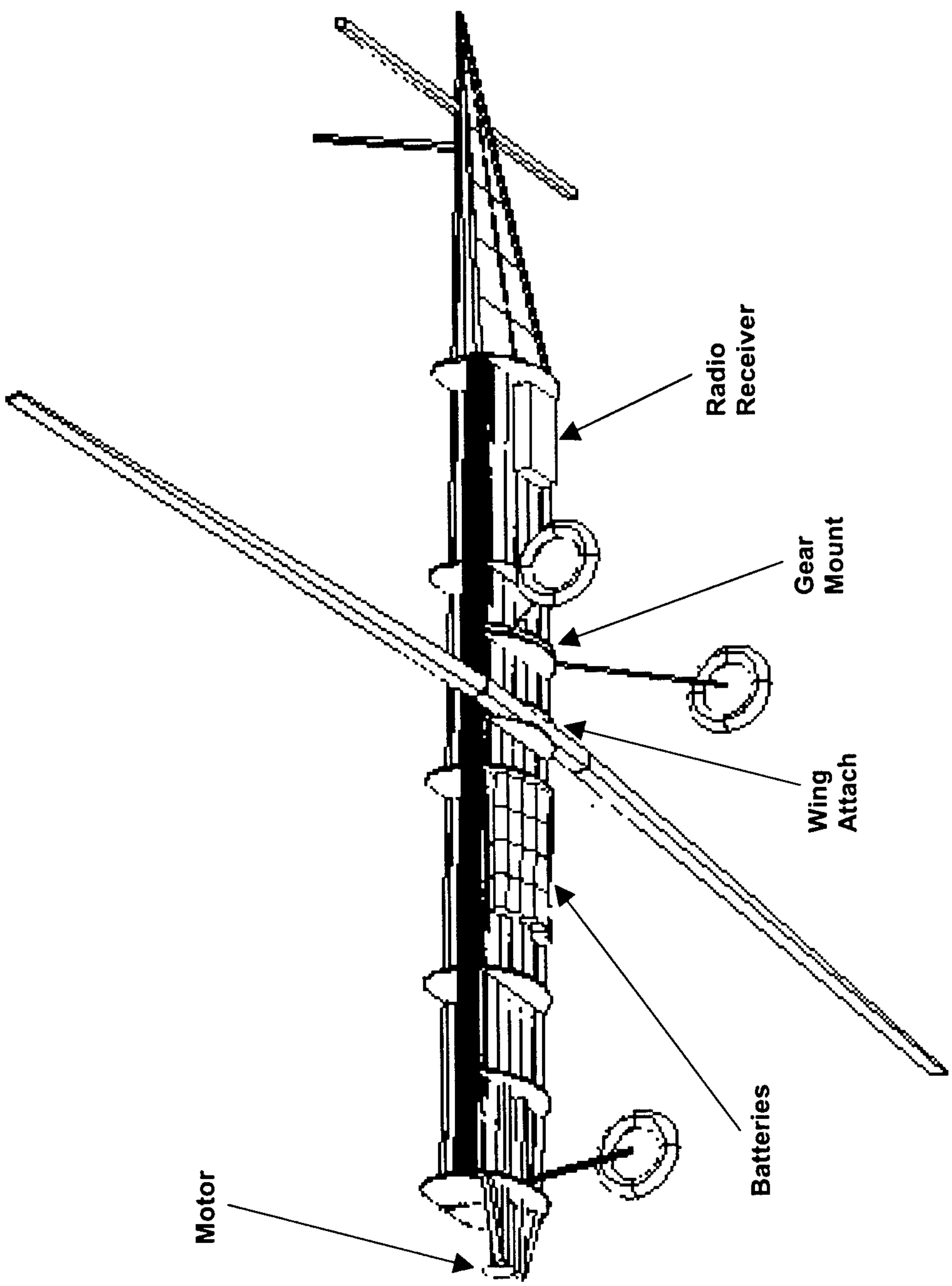
Rendered Isometric View With Payload Doors Removed. Each of the eight bays will carry three softballs for a total of twenty-four softballs.



Isometric View of Load Paths. Highlights wing spars, motor mount, and fuselage stressed skin. Also close-up of wing cross section showing spar mounted inside foam.



Systems Layout View. Underside skin removed to show wing and gear attach points, batteries, radio receiver, and motor.



6. MANUFACTURING PLAN

6.1. Methods

To avoid problems of production delays and excessive difficulty, the team worked to select configurations and design strategies that would lead to an effective and easily producible aircraft. To this end, a number of assumptions and decisions were made early in the design. Structural material selection was one of the early choices made.

In general, the team lacks experience with composites and the tooling necessary for an all-composite design. There would be little enough time for design and development neglecting the learning curve necessary to learn new processes. It was decided early then that composites would not be used. Sheet metal was also eliminated from consideration for several good reasons. Structural parts would be very thin if designed to flight loads, or the airplane would be excessively heavy if thicker material was used. Jigging and riveting structures would also time consuming and error-prone.

Wood construction has several strong benefits to offer. Construction methods in wood are relatively simple and forgiving. At the model scale, wood construction rivals sheet metal and composite methods for strength to weight. Several team members have experience in woodworking, some of that in aircraft structures. Our team also consulted an Airframe and Powerplant aircraft mechanic (A&P) with significant experience in wood structures. This A&P provided valuable advice and holds a stock of aircraft grade spruce and birch that allows quick turn-around from detail design to construction. For these reasons, wood was chosen for construction of the aircraft's primary structures.

As the design developed, details were worked out in the construction of specific components. A major concern was the final design of the wing since that would be one of our first parts built. Three methods of construction were considered. The first was to build the wing with a spar, ribs, and stringers under a skin of mono-cote. The second method was a wing with spar, ribs, and a load bearing skin forming a D-tube. The aft portion of this wing could be wood or could be mono-cote to save weight. The third method considered was a foam filled wing with a spar and wood skin. Laminar sections require a relatively smooth and continuous surface finish to maintain their best properties. A thin, non-load bearing wood skin would have the effect of preserving the aerodynamic shape and smooth surface from incidental damage, and increase structural damage tolerance for impact damage. Also, hot-wire-cut foam wing sheeted with a wood venire lends itself to a straight rectangular wing design, rather than a tapered and twisted wing, which would be slightly more aerodynamically beneficial. Also, at the low Reynolds numbers in which a model plane operates, the benefits of a complex wing were calculated to be very small.

The spar itself was designed to carry all the bending and torsion loads of the wing and transfer them to the fuselage. The spar is a solid rectangular section built up from 4 quarter-inch thick strips of spruce for a total depth or height of one inch. The spar fits into a slot that is hot-wire-cut into the foam of the wing. Construction of the spar is done on a long flat surface. The strips of wood are glued and clamped up against the box extrusions. After cure, the spar is trimmed to width on a band saw and given a linear taper from root to tip.

The fuselage design followed the design of the wing. In conceptual design, a keelson was envisioned in the fuselage with a lightweight payload bay fairing in the softballs to be carried. This would have been the easiest to construct; however, concerns arose about torsional stiffness and weight of the structure. A second-generation fuselage design was formulated to solve these problems, with a cross section of a hollow tube, and a vertical shear-web on the centerline of the section. Manufacturing methods would be a difficult issue with the design, though, because the sides of the fuselage would be thin birch plywood bent to the roughly 4-inch diameter of the softball payload. The curvature might require many bulkheads and could be prone to splitting since the plywood of the curved outer skin would directly bear the weight of the balls. In the end, the fuselage was redesigned a third time with a stressed skin construction made of thin aircraft plywood. However, stresses are diverted away from the top half of the outer shell, allowing a full-length and width hatch for rapid ball loading operations. Construction would be rather challenging compared to the simple keelson design of the conceptual stage, but the benefits of strength and weight were judged to be worth the reasonable effort required. To allow access to the wing attach bolts and to allow storage of batteries, there are access doors in the floor of the cargo bay. The doors must be bolted in at the corners and sides because the floor of the cargo bay is also the top of the "D" tube. Bolting the doors allows them to continue carrying the shear loads caused by torsion loads.

The landing gear attach points will be mounted to bulkheads inside the D-tube of the fuselage, but all the mounting hardware will be accessible from the outside of the vehicle. The plane is arranged this way so that a damaged landing gear may be replaced if necessary. Failed landing gear was one of the major problems that some teams have encountered in the past, and repair time slowed flight turn-around times.

To manufacture this structure, the fuselage is broken into sub-assemblies for simple manufacturing and simple final assembly. The "T" structure and sills of the aircraft are made of 6 flat sub-assemblies that all run the full length of the cargo bay. Each of these can be built on a flat table with no tooling beyond a straight edge.. The sub assemblies glue together to form the upper section of the fuselage. To prevent the vertical shear web of the "T" from collapsing to the side, bulkheads are spaced at intervals to fit in between ball positions. The removable access doors are built into the floor in the sub-assembly stage. The D-tube assembly starts by placing all of the sub-floor bulkheads along the "T" beam. The underside

is skinned while the "T" is supported in a jig to hold the structure straight. Once the D-tube dries, it is an integral part of the structure, and the fuselage will remain straight.

The structure of the tail boom is a continuation of the cargo bay structure. Rather than carrying the D-tube to the tail though, the top of the T-section is triangulated with a truss to provide torsional stiffness. The longerons and bending members of the tail are spliced into the members running the length of the cargo bay. Schleicher, a German sailplane manufacturer has established standards for wood splices in aircraft structures. A linear taper splice can be considered a single piece of wood if the taper ratio is 20 units of length to one unit of thickness. Splices will mate the sections as one. Repairs can also be made with splices of this type. The motor mount is spliced to the front end of the cargo bay in the same way as the tail is at the rear.

Instructions for Manufacturing

Wing Spar

- (4) 1.75" x 0.25" x 10' Spruce Strips bonded together with white glue to form (1) 1.75" x 1" x 10' main wing spar.
- While glue is still wet, clamp middle 8" to table. Slide (2) 4 degree wedges between table and wood adhered wood strips to create dihedral angle.
- Let cure for 48 hrs.
- Remove clamps, trim to specified spar dimensions using a scroll saw.

Ribs and Foam

- Create (2) airfoil templates from coordinates using a band saw and sander. Do not include cutout for spar.
- Cut (8) 1'x1' wing sections using a hotwire foam cutter.
- Custom cut the remaining (1) 1'x1' section after fuselage is constructed.
- Create (11) more airfoil templates using the band saw and sander. Include the correct spar dimensions for the span-wise placement of the ribs.
- Glue templates to the ends of the previously cut sections. Re-run the hotwire over the template including the spar cutout.
- Glue spar into the cutout, ensuring proper alignment of the airfoil sections.

Sheeting

- Sheet the entire wing with 1/32" thick balsa sheeting.
- Use Elmer's glue to attach the sheeting to the foam in small portions of the airfoil allowing time to dry between.

Horizontal Tail Spar

- Use (3) Spruce Strips bonded with Elmer's glue.
- Let cure for 48 hrs.
- Trim to specified dimensions using a scroll saw.

Ribs and Foam

- Create (2) airfoil templates from coordinates using a band saw and sander. Do not include cutout for the spar.
- Cut (2) airfoil sections using a hotwire foam cutter.
- Create (3) more airfoil templates using the band saw and sander. Include correct spar dimensions.
- Glue templates to the ends of the previously cut sections. Re-run the hotwire over the templates including the spar cutout.
- Glue spar into the cutout ensuring the proper alignment of the airfoil sections

Sheeting
- Sheet entire surface with 1/32" thick balsa sheeting.
- Use Elmer's glue to attach the sheeting to the foam in small portions of the airfoil allowing time to dry between.

Vertical Tail Spar
- Use (1) 0.25" x 0.25" x 9" spruce strip

Endcap and Foam
- Create (2) airfoil templates from coordinates using a band saw and sander. Include cutout for the spar. Glue templates to ends of foam.
- Cut (1) airfoil section using hotwire foam cutter.

Sheeting
- Sheet entire surface with 1/32" thick balsa sheeting.
- Use white glue to attach the sheeting to the foam in small portions of the airfoil allowing time to dry between.

Control Surfaces:
- Use hotwire to cut control surface.
- Attach using hinge.

Fuselage
- Construct T-beam and sills using the correct dimensions of spruce.
- Form stressed skin using 1/32" sheeting.
- Create bulkheads
- Construct mounting plate using the specified dimensions.

Final Assembly
- Assemble all pieces using white glue and bolts where specified on the engineering drawing.

6.2. Construction Schedule

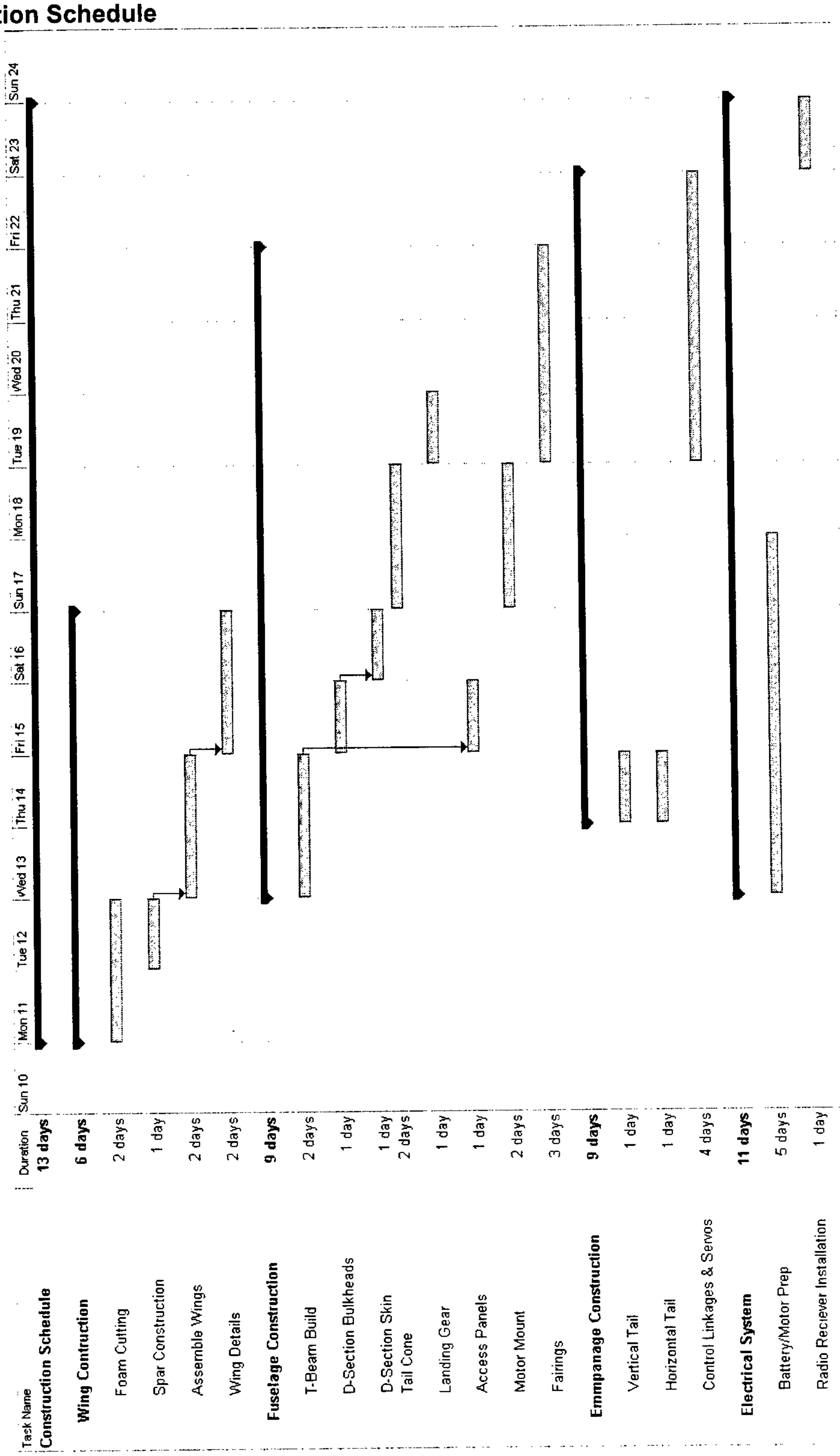


Figure 6.2.1. Planned construction schedule. At the time of this report, construction has not yet been completed, so actual construction times are not available.

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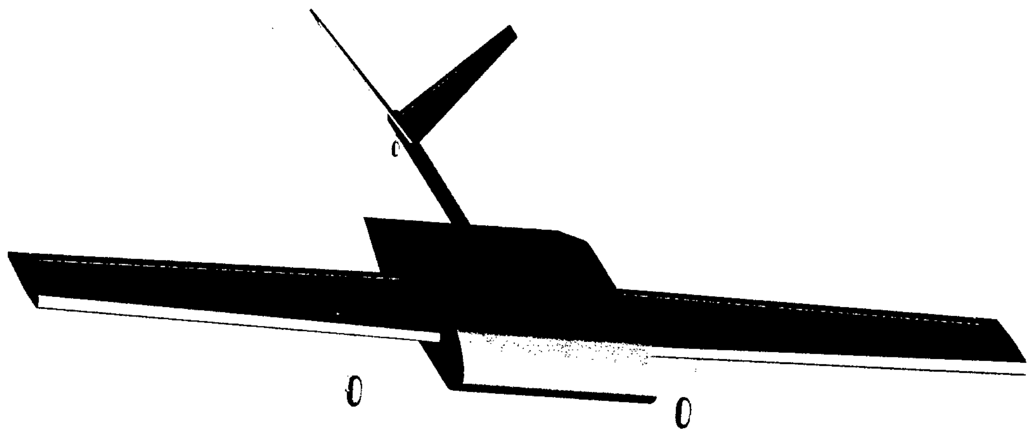
AIAA
Design Build Fly Competition



Submitted by:

University of Arizona
AIAA Student Chapter
Department of Aerospace and Mechanical Engineering
Tucson Arizona

Table of Contents:

1. Executive Summary:

The Student Chapter AIAA from The University of Arizona has set focus on competing in this year's Cessna/ONR Student Design Build Fly Competition. In an effort to build on two year's experience and successes, the students have designed and built an unmanned, elected powered, radio-controlled aircraft that would best perform the specified mission profile, based on the competition rules and guidelines. The following report outlines and details the process taken, decisions made, and design achieved in created such an aircraft.

The aircraft's mission was defined in three parts. The first part consisted of flying two laps around the course with the aircraft unloaded. The second part consisted of flying an identical two laps with the aircraft loaded with up to twenty four softballs. The final part consisted of flying an additional two laps with the aircraft unloaded. On the downwind leg of the first two parts, the aircraft was to complete a three hundred and sixty degree turn in the direction opposite of the base and finals turns. A take off requirement of two hundred feet was required for all three parts, while a ten-minute time limit was also enforced. The flight score was calculated as the summation of the total number of laps flown and the total number of balls carried, divided by the total mission time. The overall score was defined as the written report score multiplied by the summation of the best three of five flight scores, divided by the rated aircraft cost.

The intent of the student team was to design an aircraft, which would best perform the aforementioned mission and produced the best possible score. The goal a balanced design possessing good demonstrated flight handling qualities and practical and affordable manufacturing requirements while providing a high vehicle performance. The engineering process that was used to produce these results was conducted through a conceptual design phase, a preliminary design phase, and a detailed design phase. All three phases are outlined, in detail, in this report.

1.1. Conceptual Design:

The very first step in the design process was to agree upon a conceptual design of the aircraft. The students conducted weekly meetings where the many possible design strengths and weaknesses were discussed. Before work began, much time was spent discussing the achievements and shortcomings of the previous years design. Areas for improvement where then agreed upon and students were then assigned to research them. All team members were required to read the competition rules and guidelines and come to the next meeting with concept ideas and drawings. One by one the concepts were presented and discussed by all members of the team. The team then broke each aircraft into its components, such as wing, empennage, and fuselage configurations, and then tabulated their strengths and weaknesses in a histogram format. The histogram allowed for the best and most popular design alternatives to be selected. The final conceptual design was created based on the histogram output.

The various design alternatives discussed included wing area and geometry, empennage configuration and structure, landing gear, power plant, fuselage size and structure, as well as payload

size and structure. The team compared various wing configurations, such as elliptical, tapered, and rectangular wings. Also discussed were the benefits and weaknesses of mono-wing or bi-wing configurations, as well as low wing, mid wing, and high wing configurations. The team considered various tail configurations including T-tail, V-tail, and conventional style tails. The landing gear designs considered include retractable gear, fixed tricycle gear, and a tail-dragging configuration. The team considered placing motors on the wings as well as pusher/puller configurations. The team discussed the characteristics of multiple motors as opposed to a strong single motor. The students also discussed the fuselage shape and structure, and how the payload would be distributed within it. Several different ball configurations were discussed.

In order to produce the best possible results, a few tools, such as spreadsheets, Internet and library research, and competition flight video analysis were used. Some time was spent creating spreadsheets based on the team's aircraft concept discussions. As mentioned before, strengths and weaknesses of each design were listed and organized in a histogram format. This was done by use of Microsoft Excel Spreadsheets. Also, many team members conducted research on the Internet and in the University's Engineering and Science Library to understand and analyze the different configurations. Finally, much time was spent analyzing the video we had recorded from the various flights of last year's AIAA Design Build and Fly competition. Many of the competitions best aircraft were compared and discussed based on their flight performance and characteristics.

The conceptual design agreed upon was an aircraft that had a single mid-wing. Its fuselage was to be low profile and have the shape of a symmetric airfoil. The aircraft would carry twenty-four softballs and it would hold them in four columns of six rows, single height. The aircraft's empennage would have a V-tail supported by a boom running from the back of the fuselage. The aircraft was to be a tail dragger with retractable main gear. The team agreed that the power plant would be one large motor mounted on the nose of the aircraft.

1.2. Preliminary Design:

Based on the decisions made during the conceptual design phase, several different aircraft models were created and discussed. The model picked to build was based on manufacturing ability, performance, and cost. Many calculations were performed to complete the general sizing of the aircraft so that a working prototype could be created by the end of our first academic semester.

Though many aspects of the aircraft were already decided upon, there were still many alternatives yet to be discussed. First, the wing area, wing loading, wing airfoil, and span of the aircraft were considered. Designs consisting of many different combinations of these variables were created and considered. The power plant was also still being decided upon. Motors considered for the aircraft were the Astroflight Cobalt 90 and Cobalt 60. Both geared and direct drive motors were compared. Several motors were purchased and tested against many different size propellers. Thrust and rpm results from those tests were tabulated. The Graupner motors were considered but none were seriously tested due to lack of performance. Different fuselage shapes were considered. Different symmetric airfoils were

considered based on size, length, and thickness. With these characteristics defined, the sizing of the entire aircraft was completed based on the performance desired.

For this stage of the design, more spreadsheets were created to tabulate and graph data. One of the most significant spreadsheets was one that would calculate the rated aircraft cost as a function of every dimension and property of the aircraft was varied. This was used to as a guide to picking the best aircraft design characteristics. Another significant and useful tool was the University of Arizona Undergraduate Low Speed Wind tunnel (LSWT). A quarter scale model of the prototype aircraft was created and tested in the tunnel on various occasions to obtain various different experimental values. The outcomes of these tests are discussed in detail later in this report. The final and most significant tool employed was the prototype aircraft, which was flown on various occasions. Flight testing of the prototype lead to many design changes and adjustments, which increased the performance and stability of the final aircraft design.

The aircraft that was designed as a result of the preliminary design stage was one, which would carry the full load of twenty-four softballs, and would way only 14 pounds empty. The aircraft had a wingspan of seven feet and utilized Eppler 192 airfoils. The fuselage created was a NACA 0012 airfoil at 17% thickness and a chord of 42 inches. Finally, the aircraft's empennage was composed of a boom tail made out of an aluminum tube with a thickness of 1/16 inches and a V-tail. The total prototype length was 5 feet and was supported by tail dragging gear.

1.3 Detail Design:

For the detail design phase, the prototype aircraft was refined into the final, competition ready product. Many aspects of the prototype needed to be redesigned as indicated by the many flight tests. A few iterations were needed before the aircraft flew with the desired balance of stability and performance. During this phase of the design, more detailed calculations were performed and the actual flight envelope limits of the aircraft were defined. A detailed stability analysis of the plane was performed so that the structural and aerodynamic characteristics of the aircraft could be optimized.

Many of the same tools were employed for the preliminary design phase as were for the detailed design phase. Spreadsheets were used to aid with calculations and optimization analysis. Also, the prototype aircraft was continuously flown, adjusted, and re-flown. Finally, Pro-E and AutoCAD were used to create a digital, scale model of the final design.

Figure 1.1
Project Schedule

Event	Start	Finish	August	September	October	November	December	January	Febuary	March	April
Conceptual Design Phase:	8/16/01	10/11/01									
2001 Project Analysis	8/16/01	9/6/01									
Concept Presentations	9/6/01	9/31/01									
Concept Discussions	9/6/01	9/31/01									
2002 Project Strategy	9/13/01	10/11/01									
Concept Component Determination	9/13/01	10/11/01									
Concept Finalization	9/13/01	10/11/01									
Preliminary Design Phase:	10/4/01	1/11/02									
Component Group Assignment	10/4/01	10/4/01									
Wing Research and Discussion	10/11/01	10/18/01									
Fuselage Research and Discussion	10/18/01	10/25/01									
Empenage Research and Discussion	10/25/01	10/31/01									
Power Plant Research and Discussion	10/4/01	10/31/01									
Landing Gear Research and Discussion	10/25/01	10/31/01									
Prototype Finalization and Assembly	10/25/01	11/11/01									
Prototype Test Flights and Analysis	10/11/01	1/11/02									
Detail Design Phase:	12/20/01	4/11/02									
Aerodynamic Analysis	12/20/01	2/15/02									
Structural Analysis	12/20/01	2/15/02									
Aerodynamic Refinement	1/1/02	3/14/02									
Structural Refinement	1/1/02	3/14/02									
Final Aircraft Construction	1/1/02	3/14/02									
Flight Testing of Competition Aircraft	2/15/02	4/11/02									
Documentation	12/1/01	3/14/02									
Letter of Intent	10/31/01	10/31/01									
Proposal Preparation	12/1/01	3/10/02									
Submission of Proposal	3/10/02	3/10/02									
AIAA DBF Competition	4/26/02	4/28/02									

2. Management Summary:

The University of Arizona design team is composed of several engineering students that vary in age, experience, and areas of expertise. The executive management consists of a President, Project Director, Secretary, and Treasurer. Underneath the executive branch, the other student members are delegated to be responsible for various aspects of the aircraft. The responsibilities of the students are as follows:

President:

Patrick Haley is currently a sophomore who is double majoring in Aerospace and Mechanical Engineering. He is the club/team president and is responsible for everything included in the student AIAA chapter, along with overseeing the DBF project as a whole. Some of Patrick's major responsibilities include managing the organization of the team members, conducting weekly meetings, delegating tasks and responsibilities, and acting as the liaison to the Tucson Professional AIAA Chapter. Moreover, Patrick is in charge of the community and business outreach, which includes public presentations of the project, academic recruitment, and the management of the executive members as well as the entire club.

Project Director:

Keith Brock is a sophomore majoring in Aerospace Engineering. He is responsible for keeping the team on schedule with the project. This includes assigning weekly tasks and goals for the team. He is also responsible for providing the president with detailed itemizations of the parts and supplies needed for the project. Keith is also in charge of presenting weekly progress reports at the club meetings. More importantly, he serves ad the manager of the team members who work on the project. This includes solving all of the day-to-day problems that occur with the project and team members. He serves as a single point of contact for interfacing between project issues, concerns, and their solutions.

Secretary:

Jessica Dooley is a junior majoring in Aerospace Engineering. She is responsible for taking minutes at our club meetings and keeping track of all the suggestions and design issues brought up. She is also responsible for taking attendance at the meetings and making sure that all the members are working on their projects. Occasionally she will be responsible for contacting part suppliers and/or sponsors. All letter writing and industry contacts either go through her or the President.

Treasurer:

Corey Caverly is a sophomore majoring in Electrical Engineering. His responsibilities include keeping track of the club budget, general management of the club spending and procurement, and contacting sponsors and arranging for donations. Corey must plan for fund raising events and keep the club supplied with all the parts needed to build and fly the aircraft.

The general club members are responsible for the implementation of goals and objectives set forth by the contest regulations and tasked assigned by the management structure. The team members have a wide range of abilities, which provides a fun, challenging, and rewarding project environment. This, in turn, provides the club with a range of design ideas and construction techniques. Each team member is assigned to a specific aircraft component based on his/her area of interest.

3. Conceptual Design:

During the conceptual design phase, the goal of the student team was to agree on the general type of aircraft to be created. For this, it was necessary to completely analyze many significant areas of focus. First, the team found it necessary to closely examine the types of aircraft that have been made for the past competitions; both by the University of Arizona as well as other schools. By understanding the strengths and weaknesses of some of the recent successful and unsuccessful concepts, the team was able to start making educated decisions. Once this was complete, the competition guidelines were read by every member and discussed openly. At this point the team was able to discuss some basic strategy for the aircraft so that the design could be best suited for the competition. The students also needed to completely understand different types of aircraft configurations, and how each different configuration affects the aircraft's general flight performance. For this, many different types of aircraft styles were discussed in detail. Finally, each student proposed an aircraft concept and presented it to the team. Each design was broken down and discussed. From this point, the team's concept design was begun as each student's best suggestions were collected and discussed.

3.1 Team Strategy:

Before the any design project can begin, it is very important to specifically define the need and goal of the project. The goal of this team was to create an aircraft that would best perform the specified mission profile, based on the competition rules and guidelines. Because the aircraft's ability to do this is quantified its flight score, the team focused on creating an aircraft which would maximize this value. With this in mind, the previous year's aircraft was analyzed so that its score limitations could be understood.

The University of Arizona finished 13th in last year's competition with an aircraft named Aircat2001. The Aircat was a successful aircraft because it was able to complete three flights with a load of fifteen pounds of steel and seventy-eight softballs. Its heavy lift ability was attributed to its bi-wing, eight-foot span, and high lift airfoils. Because it used an Astroflight Cobalt 90 motor driven by thirty six Sanyo 2400 mAh cells, the aircraft had enough power to produce it's weight in thrust. Unfortunately, the aircraft's major limitation was its speed. Because the aircraft produced quite a bit of drag, many teams were able to fly either much faster or for a longer period of time. Another limiting factor was the Aircat2001's rated aircraft cost. Although the Aircat could lift more then most other aircraft, its cost was higher. Since our aircraft rarely lifted as much as it could, the high cost was unnecessary. On account of these lessons learned, the team agreed that the strategy for winning this year's competition would be to produce an aircraft that could lift the maximum load, be just as fast as the other planes, while having the lowest cost. Minimizing cost meant that the team would have to take chances and trade stability for performance. It was agreed, however, that a balance would between the two would be made. As each aircraft component was discussed, the team made a considerable effort to conserve high performance and low cost.

3.2 Wing Configuration:

An aircraft's wing can take many forms, each having significantly different flight characteristics. For this reason, it was important to discuss and consider all of the options. In the past, the University of Arizona team has designed stabile and high lift aircraft with a rectangular style wing. This year, however, it was determined that the aircraft must be kept maneuverable and aerobatic. For this reason, the team discussed the advantages of creating an elliptical or tapered wing. It is well known that the elliptical wing is the most efficient. It was decided to that this type of planform would be too difficult to produce uniformly. The tapered wing is known for having better maneuverability then a rectangular wing, and it was believed that it could produce just the necessary amount of lift. The stability was compromised with since a tapered wing often results in poor handling characteristics at low speeds, but the performance was greatly enhanced at the same time.

The bi-wing design of the previous year's plane produced a significant amount of lift at low speeds, but it also added significant amounts of drag. The wing added wing area of the aircraft also increased the cost. Since this competition didn't require the aircraft to lift more then ten pounds of payload, approximately the weight of twenty-four softballs, the planform didn't need to produce nearly as much lift. Having considered both, the mono-wing was the preferred configuration.

3.3 Fuselage Configuration:

The fuselage is perhaps the most important of all aircraft components. Although designs exist that have no fuselage, most conventional aircraft rely on the fuselage to carry the payload and possess the majority of the structure. For this reason, much time was spent discussing the fuselage design. As mentioned before, last year's aircraft had significant amounts of drag due to its fuselage's profile. This was necessary, however, to support the bi-wing configuration. This for this year's design, the team focused on keeping the fuselage low profile and thus low drag. Choosing a fuselage which encompassed the minimum amount of volume would also reduce the affect of side force when flying in a cross wind, which is expected in Wichita, Kansas. To obtain these properties, the team decided that the fuselage should have the profile of a symmetric airfoil, or at least resemble the shape. The only variable left would be how to distribute the payload.

The competition guidelines clearly specify that the softballs must be stored single height and at least doublewide. For this reason, the team considered holding the softballs either in pods under the wing, or within the fuselage. It was of some concern that the pods would add unnecessary drag so that configuration was no longer considered. With the balls being held within the fuselage, there were several ways they could be configured. The team considered holding the balls in either two columns of twelve rows, three columns of eight rows, or four columns of six rows. Since we wanted the aircraft to be short in length, and easy to load and unload, we quickly eliminated the two by twelve configuration. Of the two remaining configurations, the three by eight configuration would have a longer fuselage yet would allow for more then affective wing area at the same span. The four by six configuration would allow for a shorter fuselage and it would be much easier to produce. In addition, the entire payload could be kept

closer to the center of gravity and the aircraft would be able to fly just as well without the payload as it would with it. For these reasons, the aircraft was designed to hold it's payload internal to the fuselage in the three by six ball configuration.

3.4 Empennage Configuration:

With the wing and fuselage configurations defined, there were several tail configurations, which could be chosen for the aircraft. Several members suggested tapering the fuselage all the way back and mounting a conventional style tail. This was a preferred method because it would be the easiest to produce and it would be very successful in stabilizing the airplane in pitch and yaw. Another configuration considered was the T-tail. This is very similar to the conventional style tail, except in this case, the horizontal surface is mounted on top of the vertical stabilizer. This is more aerodynamically efficient since the elevator is in a region of air which is undisturbed by the fuselage. Unfortunately, it requires more structure. Other members suggested that the tail be composed of two booms, with a small vertical stabilizer extending from the end of each boom, and being connected by one horizontal surface. This idea was best suited if we were to go with a pusher/puller motor setup. This idea wasn't officially discarded until we had decided to go with only one front mounted motor. The final tail configuration discussed was a V-tail. In this case, there are two stabilizing surfaces mounted in a V shape. Typically, the surfaces are angled 30 to 40 degrees from the horizontal. This is preferred because it can often be build lighter then any other type of tail, and would also significantly decrease the rated aircraft cost. Unfortunately, it would require extra structure and be more difficult to build. It was also uncertain whether it would be able to provide enough control as well as pitch and yaw stability. The team decided to go with this configuration because it had the potential to lower the cost of the aircraft.

3.5 Power Plant Configuration:

The University of Arizona team learned in the 2000-year competition that none of the other design features mattered at all if the power was insufficient. Over the past two years, much time was spent perfecting the propulsion section of this aircraft. The power plant is composed of the motor, battery pack, and propeller. These three separate but equally important parts must be optimized to best perform the mission at hand. For example, a motor and propeller combination can be made to produce anywhere from a lot of torque with little speed, to a lot of speed with little torque. The more torque produced, the more the aircraft will be able to lift. Without the necessary speed, however, the aircraft won't produce enough lift to get into the air within the two hundred feet take off requirement. Since the propellers vary in diameter and pitch, the number of combinations is virtually endless. The battery pack utilized can also affect the motor's pulling power and duration. Unless the pack provides enough voltage, the motor may not be able to draw enough power to obtain the thrust and speed needed for take off. Equally important, the pack needs to be composed of cells that are able to run the motor for the duration of the ten-minute flight. Because of these issues, a pack of 36, 2400 mAh cells were chosen for their lightweight and duration/power balance.

3.6 Landing Gear Configuration:

The landing gear may not seem as influential as other components, such as the wing or fuselage, but they still make a significant difference in the aircraft's performance in the air as well as on the ground. A tricycle style gear consists of a nose wheel mounted forward of the center of gravity (CG) and a main gear mounted right on, or near to the CG. In a tail dragger configuration, the main gear is mounted just forward of the CG, and a small wheel or skid is mounted right underneath the tail. This configuration has far less drag in the air although is the least stabile on the ground. The tail draggers also have a tendency to "whether vain" or rotate about the main gear during a crosswind. Another way to decrease drag is to make the gear retractable. Both of the above configurations can be made so that the wheels fold up into the fuselage or wing after take off. This is preferred because the decrease in drag is significant. When this is performed, the aircraft can often fly longer, further, and faster. Unfortunately, it often requires an increase in weight and complexity of design. All of the above configurations were considered and the team decided on a tail dragger configuration with a retractable main gear. This was favored for the decrease in drag and increase in performance. Difficulties building the retractable gear into the prototype much later in the design process limited the design to a fixed main gear, however.

3.7 Figures of Merit:

For each of the design alternatives discussed during the conceptual design phase, an aircraft concept was created with each different combination of component. Then, for each different aircraft, the rated aircraft cost was calculated. These different quantitative approximations of performance are referred to as Figures of Merit. By comparing the figures of merit for the different designs, the best configuration could be determined. These values can be viewed in Figure 3.1.

A spreadsheet was created that contains several options for the aircraft conceptual design. For each design, the components varied in retractable to non retractable gear, tricycle or tail dragger, tapered or rectangular wing, seven or eight foot span, and conventional or V-Tail empennage. The rated aircraft cost was calculated for every combination of the aforementioned properties. For the rated aircraft cost, the aircraft was assumed to weight fifteen pounds, have one motor and five pounds of batteries, a wing with a maximum chord of 1 foot, and two control surfaces. The aircraft was also assumed to have a fuselage length of 2 feet and utilize a single motor and a single propeller. It didn't matter if the assumptions were wrong, just as long as they were consistent for each concept. This would yield an even comparison. In order to estimate the best possible score, the report was assumed to score one hundred percent, and the aircraft was assumed to fly for ten minutes with a load of 24 softballs. The number of laps was estimated and varied to account for components that had more or less drag. This was done in a consistent manner so that the approximation would yield a good comparison.

From the spreadsheet on the following page, it can be observed that the top five designs all incorporate V-Tails and a seven-foot wingspan. In addition, three out of the five top scores had a tapered wing and a tail dragger configuration. It can also be noted that four out of the five to best scores incorporated retractable gear. Due to the results produced from the Figures of Merit spreadsheet, the

conceptual design included a wing configuration consisting of a tapered, seven-foot wing span, an empennage configuration consisting of a V-Tail, and a landing gear consisting of retractable main gear in a tail dragger configuration.

Figure 3.1
Figures of Merit

Aircraft	Wing Planform		Landing Gear		Empennage	Figures of Merit		
Concept	Rectangular/Tapered	Span (7/8 ft)	Retractable	Tricycle/Dragger	Conventional/Vtail	RAC	# Laps Flown	Best Total Score
1	R	7	Y	T	C	11.8	8	8.745762712
2	R	8	Y	T	C	11.96	8	8.628762542
3	***R***	***7***	***Y***	***T***	***V***	***11.7***	***8***	***8.820512821***
4	R	8	Y	T	V	11.86	8	8.701517707
5	R	7	Y	D	C	11.8	9	8.754237288
6	R	8	Y	D	C	11.96	9	8.637123746
7	***R***	***7***	***Y***	***D***	***V***	***11.7***	***9***	***8.829059829***
8	R	8	Y	D	V	11.86	9	8.70994941
9	R	7	N	T	C	11.8	6	8.728813559
10	R	8	N	T	C	11.96	6	8.612040134
11	R	7	N	T	V	11.7	6	8.803418803
12	R	8	N	T	V	11.86	6	8.6846543
13	R	7	N	D	C	11.8	7	8.737288136
14	R	8	N	D	C	11.96	7	8.620401338
15	R	7	N	D	V	11.7	7	8.811965812
16	R	8	N	D	V	11.86	7	8.693086003
17	T	7	Y	T	C	11.8	9	8.754237288
18	T	8	Y	T	C	11.96	9	8.637123746
19	***T***	***7***	***Y***	***T***	***V***	***11.7***	***9***	***8.829059829***
20	T	8	Y	T	V	11.86	9	8.70994941
21	T	7	Y	D	C	11.8	10	8.762711864
22	T	8	Y	D	C	11.96	10	8.64548495
23	***T***	***7***	***Y***	***D***	***V***	***11.7***	***10***	***8.837606838***
24	T	8	Y	D	V	11.86	10	8.718381113
25	T	7	N	T	C	11.8	7	8.737288136
26	T	8	N	T	C	11.96	7	8.620401338
27	T	7	N	T	V	11.7	7	8.811965812
28	T	8	N	T	V	11.86	7	8.693086003
29	T	7	N	D	C	11.8	8	8.745762712
30	T	8	N	D	C	11.96	8	8.628762542
31	***T***	***7***	***N***	***D***	***V***	***11.7***	***8***	***8.820512821***
32	T	8	N	D	V	11.86	8	8.701517707

4. Preliminary Design:

During the conceptual design phase, many aircraft configurations were discussed and explored in an effort to find the best overall configuration for the aircraft. Having discussed many concepts, the figures of merit analysis aided in the finalization of the concept design. Once the general components of the plane had been determined, it was time to further specify each component group and complete the sizing of the prototype aircraft. For this stage of the design process, it was the goal of the design team to create and flying model of the aircraft so that further design refinement could be completed.

4.1 Wing Sizing:

One of the very first constraints that were defined on the wing design was the wing loading. Wing loading, calculated as the reverence area of the wing divided by the aircraft weight, greatly affects the aircraft's overall performance. If an aircraft has a low wing loading, such as ten or fifteen ounce per square foot for example, the wings will support much of the aircraft's weight, and the aircraft will behave as a glider or trainer. If an aircraft has a high wing loading, such as twenty to thirty ounces per square foot, the aircraft will less stable, more maneuverable, and faster. To best obtain the teams goals defined in the conceptual design stage the wing loading was specified to be within twenty to thirty ounces per square foot.

From the figure of merit analysis during the conceptual design phase, it was determined that the wing would have a taper and a span of seven feet. In order to obtain the desired wing loading, the root chord was required to be seventeen inches and the tip chord was required to be twelve inches. These dimensions were set for ease of fabrication. At this point, the only variable left was the airfoil. Because the wing area is limited, it was necessary to choose an airfoil that would provide enough lift at low speeds, with out too much drag. For this reason, many airfoils were considered. Those airfoils include the Eppler 192, the Eppler 176, and the Selig S6061. The Eppler 192 was finally chosen for its balance of lift verses drag at low speeds.

The wing spar is another critical property of the wing because it supports all the major loading of the wing during every aspect of flight. The wing spar is often times a wooden or metal beam that extends the length of the span within the wing. Much time was spent experimenting with many spar designs. It was important to make the spar lightweight since it represents a significant amount of the aircraft weight. Similarly, it was also important to make the spar strong since it is important for the wings to remain attached to the fuselage and not fold during flight. Many different materials were examined from solid wooden beams of various dimensions, to built up wing box designs. The design that seemed to have best balance of weight to strength was a hollow tube made out of one-sixteenth inch aluminum. Since rest of the wing was to be made out of white foam, this spar would make the wing strong enough so that the entire loaded weight of the plane could be supported by just the wing tip. This is equivalent to a design safety factor of two.

4.2 Fuselage Sizing:

It was determined in the conceptual design phase that the fuselage would be the shape of a symmetric airfoil, and the twenty-four softballs would be carried internal to the fuselage in a configuration of six rows of four columns. For this configuration, it was found that the payload section would have to be a minimum of fourteen inches wide by twenty-one inches long. There was to be a channel through the center of the fuselage for the flight electronics and batteries so two inches needed to be added to the width. Finally an additional two inches was added to each side for structure. The resulting width was eighteen inches. The length of the fuselage was determined by the airfoil profile. Many different airfoils were considered, such as the NACA 0012, the Selig SD8020, and the NACA 64008A. These were chosen for their excellent lift to drag coefficient ratios. Ultimately, a modified NACA 0012 with seventeen-percent thickness was chosen because it was the only one of the three that would be thick enough to contain the softballs. In order to do this, its chord was set to 42 inches.

With the geometry of the fuselage set, its materials and structure could be determined. In order to keep the aircraft as light as possible, the team agreed to use a balsa and plywood frame to support the majority of the loads. White foam cut to shape and epoxied wherever necessary to increase the rigidity of the frame. It was decided that the fastest way to load and unload the softballs would be through a top mounted, removable, foam hatch. Once completed, the fuselage was incredibly light since it had only the minimum structure to support it during flight. This was of some concern but with the performance gained, it was concluded that the increase in performance was substantial.

4.3 Empennage Sizing:

The tail was the one of last major obstacles of the design. Although it had been decided that the aircraft would have a V-Tail, there was still much discussion on how to mount that to the fuselage. The length from the center of gravity of the tail to the center of gravity of the wing was calculated to be three feet four inches. This was obtained by multiplying two and a half by the mean aerodynamic chord. The wing spar was placed at about the quarter chord of the fuselage airfoil, so the tail needed to be placed 18 inches aft of the fuselage trailing edge. In an effort to keep the aircraft light and relatively low drag, the team concluded that a boom would be used to extend the V-Tail to the desired length.

The material of the boom was of significant concern. As with every component of the aircraft, a perfect balance between weight and structure needed to be found. Many different composite, plastic, metal, and wooden boom designs were considered. It was important that the boom be rigid enough so the tail would not be allowed to deflect significantly during flight. Although many of the tested booms could provide the needed strength, most were too heavy to be considered. A one-inch diameter, one-sixteenth inch thickness aluminum tube was found which seemed to provide the best strength to weight ratio. This was selected for the prototype while many composite booms were still being tested. It was determined that if a better material was found, the boom could be replaced quite easily.

4.4 Power Plant Finalization:

As mentioned previously, the University of Arizona team has stressed much importance in the power plant configuration. Much research and experimentation has been completed in this area to assure that the best motor, battery, and propeller combination could be arranged. The competition rules state that either Graupner or AstroFlight families of brushed electric motors must be used. All motors from each family were analyzed and those considered were the AstroFlight Cobalt 90 and 60. The University of Arizona has flown a Cobalt 90 in the past and has been able to achieve great thrust performance. Both motors purchased were with a gear reduction and direct drive configurations for testing. The results of these tests are tabulated in Table 4.1. From the experimental data in the table, it can be observed that the Cobalt 90 produces the greatest amount of thrust. Once completed, the prototype was flown with both the direct drive and gear reduction configurations. Due to availability, thrust data was only obtained for the geared motors. The direct drive motors have been tested on the ground, but no actual data is available at this time. It was observed, however, that the direct drive motors have slower acceleration, but a much higher top end speed. If the aircraft can take off with in the required two hundred feet when being powered by a direct drive motor, that may be the best configuration. For the time being, that the best overall flight performance, in terms of available thrust and flight duration comes from the Cobalt 90 with the geared reduction box.

The battery technology hasn't changed in the last two years. For that reason, the same battery pack flown in the 2001 DBF competition was determined to be flown in this year's aircraft. The Sanyo Cadnica 2400 mAh cells are the lightest and most powerful nickel cadmium cells that can be purchased. Because of their lightweight, 36 cells are able to be grouped together and still be below the five-pound weight limit. Since this pack performed flawlessly at the competition, and had always supplied the motor with enough power to fly the full ten minutes, there was no need to change the pack configuration.

Table 4.1: Thrust Testing

	Cobalt 60 Geared	**Cobalt 90 Geared**
Propeller	**Thrust**	**Thrust**
Mejzlik (20 x 8)	7.5	9.5
Mejzlik (22 x 8)	9.5	12.5
Menz (21 x 10)	9.5	12.5
MasterAirscrew (20 x 10)	6	8.2
MasterAirscrew (20 x 8)	8	9.8
MasterAirscrew (20 x 6)	6.5	9

4.5 Landing Gear Sizing:

The landing gear configuration selected during the conceptual design phase is significantly different then the team has ever made in the past. Because a tail dragger was selected, there were some tail-wheel issues to be decided. Firstly, it needed to be decided whether the team wanted to go with a small wheel or a skid. It was determined that a skid would be more aerodynamic in the air, and would help the plane slow quickly on the ground, but would provide poor ground handling and unnecessary friction during take off. A wheel, on the other hand, would provide slightly less aerodynamic performance in the air, but would significantly increase the ground handling characteristics. Both types were attempted and it was obvious that a small wheel was preferred. The next issue was whether the tail wheel would be able to steer. By adding a servo to control the tail wheel, extra weight and cost would be factored into the design. This was definitely not favored but it was concluded that the aircraft would need to be able to steer to fight the whether veining of the aircraft during takeoff and landing. Fortunately, one of the members proposed a design where the tail wheel could be attached to the V-Tail control surfaces via small springs, so that the wheel could steer without the need of adding another servo and control linkages.

The main gear was to be made retractable but, unfortunately, the wing did not provide the necessary thickness to support the gear. When the main gear was mounted in place, the gear seemed weak and the aircraft was generally unstable. It was quickly asserted that a main gear would have to be fixed in place. It was decided, at that point, to make the main out of carbon graphite. In order to do this, a template was cut out of wood and the thirty layers of carbon fiber and fiberglass were laid on top and vacuum bagged. Once the gear had cured completely, the graphite was broken from the mold and cut into shape. This technique had been perfected over the past couple years and stress testing has indicated that the gear was able to support twice the aircraft weight and was still flexible enough to dampen hard shock during landing. Furthermore, the weight of this gear is half of what can be purchased or assembled out of different materials. To provide the propeller ground clearance, the gear was made eight inches tall. With three-inch diameter wheels this would allow the aircraft to spin a 28-inch propeller. The main gear was mounted just forward of the center of gravity as to provide sufficient stability. It was also determined that should a design be developed, another attempt would be made to incorporate retractable gear.

4.6 Figures of Merit:

One of the tools that was used to help make the final decisions during the preliminary design phase was another figure of merit analysis of the remaining aircraft variables. To do this, various weighting factors were assigned to each component being analyzed. Each weighting factor was a function of production difficulty, drag, strength, and weight. Based on how much the component contributed to each category, the component received a value between 0 and 3 if 0 is a low value and 3 is a high value. The weighting factor could then be calculated by summing up all the individual factor scores. For example, retractable landing gear is difficult to produce (+2), has very low drag (+0), has no

added strength (+0), but adds some weight (+1). For this, the weighting factor would equal three. Smaller the weighting factors yield higher the scores. These values can be observed in the figure of merit graph on the next page.

The figure of merit analysis shows two very interesting features. As indicated by the weighting factors, the carbon tail boom and the carbon wing spar are preferred to the aluminum. The aluminum was chosen because it cost less, and was readily accessible. The carbon rods had to be researched and ordered. Plus, the cost was found to be expensive. Carbon rods were ordered for testing but the results of those tests were not available at the time this report was written. The option to switching to carbon graphite components was never thrown out. The figure of merit analysis also shows that retractable gear and tail dragging configuration were still the preferred landing gear configurations.

4.7 Prototype Fabrication

With all of the sizing complete, a prototype aircraft was created and its first flight was on November 11, 2001, thus ending the preliminary design stage. Based on the aircraft's flight performance, a few changes needed to be made to the overall design. These refinements, all future flight tests, and any further analysis of the aircraft was done as part of the detailed design stage.

Figure 4.1
Figures of Merit

	Component	Production Difficulty	Drag	Strength	Weight	Weighting Factor	
Landing Gear	Retractable	2	0	0	1	3	<-----
	Fixed	0	3	1	1	5	
	Tricycle	0	2	1	0	3	
	Tail Dragger	0	1	0	0	1	<-----
Wing Structure	foam w/ spar	0	NA	1	1	2	<-----
	composite/foam	3	NA	0	0	3	
	composite/spar	2	NA	3	2	7	
	built up	1	NA	3	3	7	
Spar	aluminum tube	0	NA	1	1	2	
	composite tube	0	NA	1	0	1	<-----
Tail boom	aluminum tube	0	1	3	1	5	
	wood bar	0	1	2	3	6	
	composite tube	0	1	3	0	4	<-----

5. Detailed Design Stage:

Upon completion of the first test flight, the team moved directly into the detailed design phase. As soon as it was obvious that the aircraft was generally good, much effort went in to refining the design of the plane. The important areas for refinement were aircraft stability, general aerodynamic optimization, and general structural optimization.

5.1 Wind Tunnel Test Results:

A 29% scale model of the aircraft was created for general wind tunnel testing. The model was placed into the University of Arizona Low Speed Wind Tunnel (LSWT) for testing over the months of November and December. The first tests were run in an effort to determine the stall point of the aircraft without jeopardizing the prototype. In addition to stall data, the tests returned lift coefficient, drag coefficient, and pitching moment coefficient over a range of common flight angles of attack. By plotting the data in different ways, many aerodynamic properties of the aircraft can be determined.

Aerodynamic coefficients are values that aerodynamicists use to model all of the complex dependencies of shape, inclination, and some flow conditions on a particular property. For example, the lift coefficient is equal to the lift divided by the quantity of the density times half of the velocity squared times the wing area. These dimensionless coefficients are useful because while lift is a function of velocity, the lift coefficient is not. There for, for set atmospheric conditions, it varies only with angle of attack. Three common aerodynamic coefficients that were quantified during the wind tunnel tests where the lift coefficient (Cl), the drag coefficient (Cd), and the pitching moment coefficient (Cmp). These values are plotted verses angle of attack in figure 5.2. Unfortunately, do to limitations with the model and the wind tunnel mount, the model was only able to be tested over the range of angles of attack of negative eight to positive eighteen. Fortunately, this is enough of a range to see general trends in the data.

On figure 5.2, it can be noticed that as the angle of attack increased, the lift coefficient increased linearly. This will continue for a point until the coefficient reaches it maximum. Past this point, the coefficient will drop as the aircraft stalls. To the top of the lift coefficient data, a slight decrease in slope can be observed. It is believed that this is the very first sign of stall. Because this lift coefficient data matches the theoretical data so closely, it can be said with confidence that the maximum lift coefficient of the aircraft is about 1.2. This will be verified with future tests.

Also plotted on the same graph is the drag coefficient and pitching moment coefficient. These plots are significant because their values are so low. The drag coefficient ranges in value from –0.204 to 0.0913. The pitching moment coefficient is also very low as it ranges in value from –0.0424 to –0.0004. These values are quite low and this very sufficient. Figure 5.3 is what is called a polar curve because it plots the lift coefficient versus the drag coefficient. Neither the lift coefficient nor the drag coefficients are very useful by themselves because the aircraft's aerodynamic performance is a function of the two. By comparing how the lift and drag coefficients vary together, many flight characteristics can be analyzed. For example, from the graph, it can be observed that the regions of highest drag are when the aircraft the lift coefficient are either very large or very small. This best represents takeoff and landing. Fortunately,

the graph also indicates that the region of lowest drag is when the lift coefficient is also small. This best represents the aircraft in cruise.

The wind tunnel data obtained was used to analyze the aircrafts performance abilities as well as to help solve performance problems. As mentioned in the following section, review the wind tunnel data often aided the team in finding solutions to various flight problems. This data played a very significant role in the detailed design phase, because it allowed the team to successfully optimize the aircrafts aerodynamic performance characteristics.

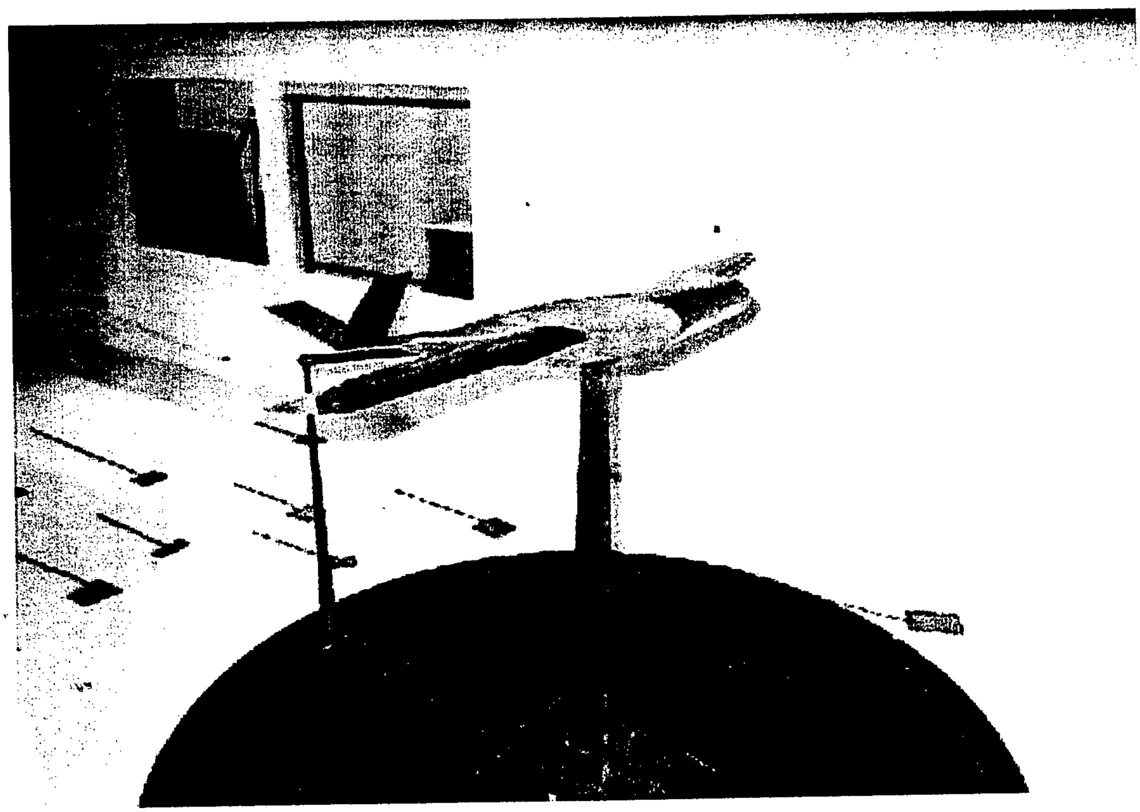


Figure 5.1: 0.29 Scale Model Wind Tunnel Test

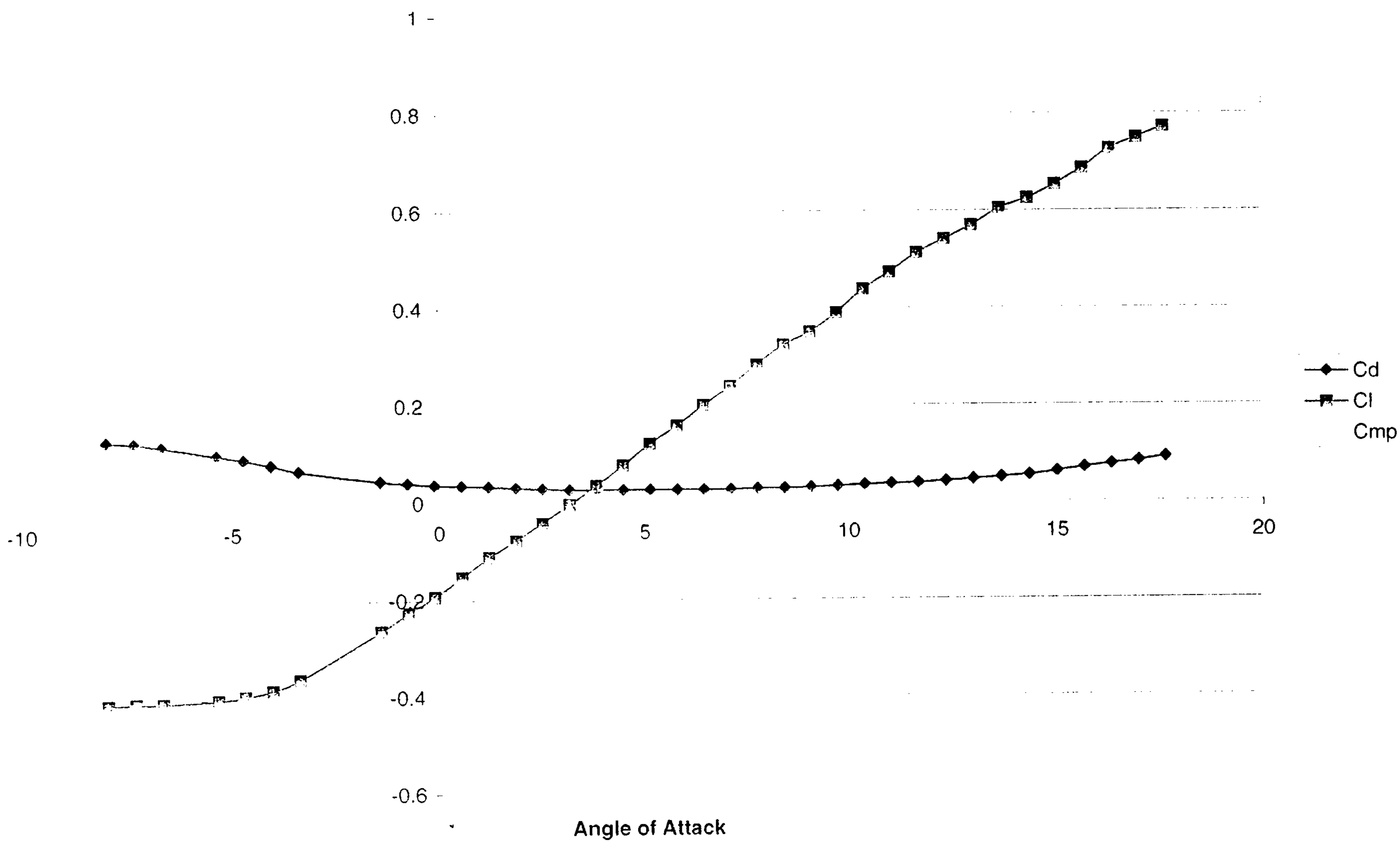
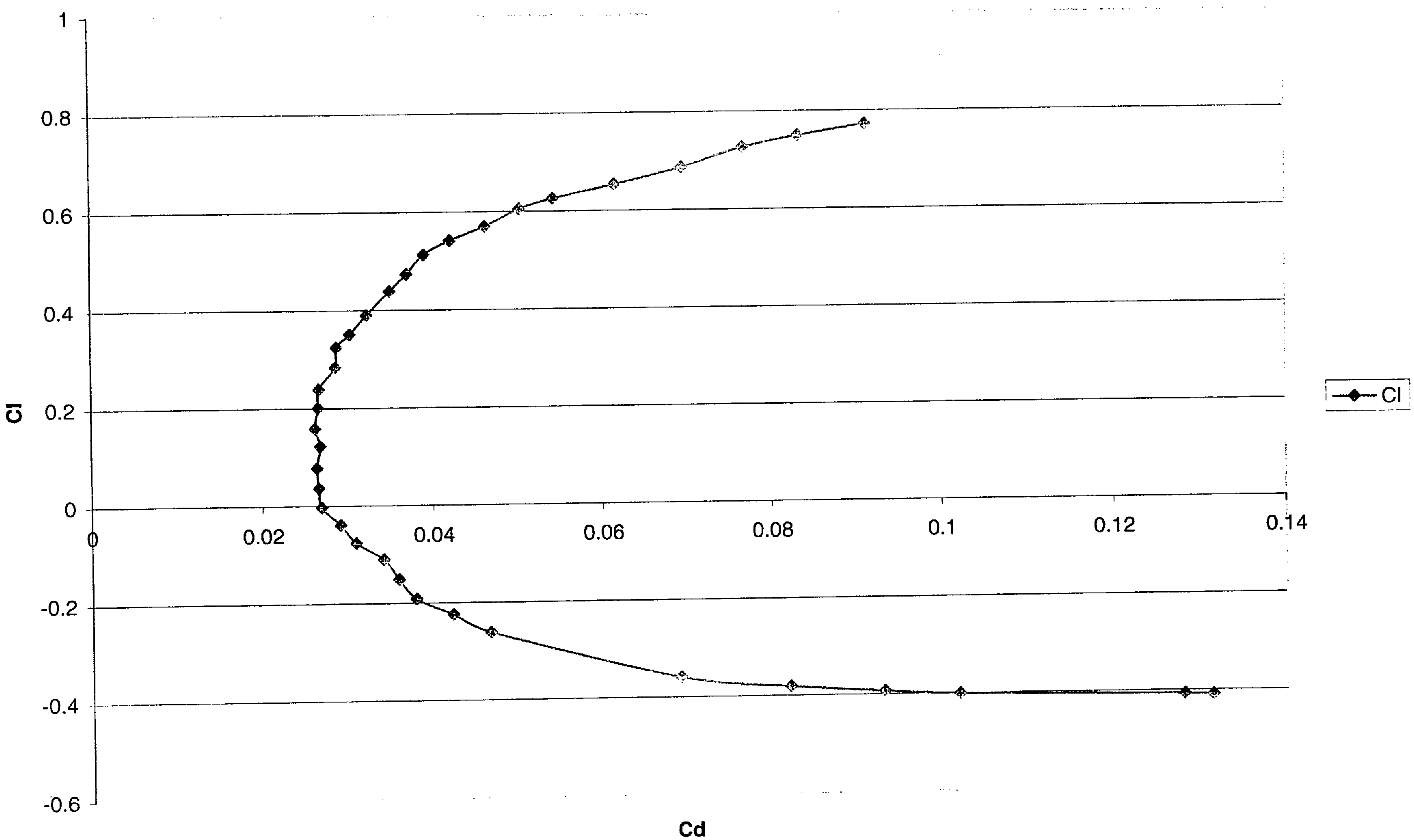


Figure 5.2: Cl vs. Cd



5.2 Flight Test Results:

Flight testing of the prototype is very critical because it yields a direct representation of how the current design of the aircraft performs in the air. Over the course of several weeks, the prototype was flown and varied slightly. After the team would watch the aircraft in the air, and listen to the pilot's comments regarding the aircrafts flight performance, the team would meet and discuss how to attack the issues and/or augment the aircraft's overall performance.

The very first flight of the aircraft was successful yet many issues regarding the aircraft's stability arouse. The aircraft had no problem taking off within the two hundred foot requirement and quickly climbed into the air. The pilot felt that the aircraft had more then sufficient power and that it was capable of flying quite quickly. The aircraft seemed very responsive in roll and didn't have any tendencies to stall at reasonable angles of attack. The pilot did have issues with the aircraft's pitch stability. The aircraft would not fly hands free because the aircraft would suddenly begin to climb or dive. This became an issue, as the pilot could not predict when either of these events would occur. The aircraft seemed generally controllable, but a few modifications had to be made.

At first it was believed that the pitch instability was coming from extra lift originating from the fuselage. With the fuselage creating lift at its center of pressure, and the wing creating lift at it's center of pressure, which was further forward the that of the fuselage, a moment would be created causing the aircraft to climb. If the prototype did not have these lift vectors originating from the same point, it would be difficult to keep the aircraft flying level. This suggestion was not thrown out, but it was set aside since the wind tunnel tests revealed that the pitching moment coefficient remained nearly zero over a wind range of angles of attack. It was believed that a significant problem would have been visible in the data.

Another solution was to simply move the center of gravity further forward. If the center of gravity was too far back, this could cause the aircraft to be very maneuverable and not too controllable. As the center of gravity moves forward, the aircraft would tend to be more stable. During the first test flight, the center of gravity was placed just forward of the wing spar. For the following test flights, the center of gravity was placed at least a couple inches in front of the spar. When this was done, handling characteristics were greatly improved.

Finally, it was of some concern that the boom tail was being allowed to flex too much during flight. It was felt that either the aluminum boom, or the V-Tail's attachment to the boom wasn't rigid enough. If the boom tail had been oscillating, or simply allowed to deflect, it would then change the angle of attack of the tail. If that were the case, a moment would be created that would change the angle of attack of the entire aircraft. It was felt that this was a viable solution to the problem so both a more rigid boom and better V-Tail mounting procedures were researched. None of these modifications were performed on account that they would add weight, and that the aircraft's in flight stability problems were solved once the CG was moved forward.

At this point, the aircraft completed several additional test flights, which tested its aerodynamic limitations. After each flight, the limitations were pushed back as the aircraft completed increasingly

difficult maneuvers. During one of these test flights, the aircraft successfully completed three consecutive aileron rolls. Once it was determined that the aircraft had more then sufficient aerodynamic stability, the team switched it's focus to finding sources of ground instability and structural weakness.

During several of the test flights, it was noticed that the aircraft had significantly poor ground handling characteristics. Although these were often sufficiently combated by the pilot, it was decided that changes would need to be implemented. Observations showed that the tail wheel would flutter and track off course while the aircraft was accelerating. This often caused the pilot to abort a take off or take off abruptly and uncomfortably. In order to save weight, the design called for the tail wheel to be controlled by two springs attached to the control surfaces of the V-tail. This allowed the tail to steer without addition of another servo. Unfortunately, it was the springs allowed for the flutter. Many alternatives were considered but the team finally decided to re-configure the aircraft to accommodate a tricycle gear. This was mostly in part to the fact that there was much concern on how the typical Wichita wind would affect a tail dragger. The tricycle gear proved to be the most stabile and would yield the best ground handling. For these reasons, the changes were made.

Several structural weaknesses were found in the design as well. Although the aircraft was very light, and although no one wanted to add more weight, it was decided that some parts of the internal structure were redesigned with more support. As stated before, the walls of the fuselage was constructed with thin balsa wood sheeting. Because of this, the walls were flimsy and the entire fuselage could twist and skew if the proper forces were applied. Although we knew the aircraft was strong enough for aerodynamic loading, it was believed that during loading, unloading, service, and travel, the aircraft could break and jeopardize the structure. In an effort to strengthen the aircraft without adding any extra weight, the walls were lined with a think plywood truss, and a keel of fiberglass and kevlar honeycomb was added along the center of the fuselage. After these modifications the prototype aircraft was re-built, and the University of Arizona's "Evolution" was ready for competition.

5.3 General Aircraft Properties:

Rated Aircraft Cost = 12.3367

Table 5.1: Wing Characteristics

Wing Span	7 ft
Camber	3.5 %
Thickness	12 %
Root Chord	17 in
Tip Chord	12 in
Maximum Lift Coefficient	1.2
Maximum Drag Coefficient	.04
Wing Area	9.3 ft^2
Wing Loading (empty)	25.8 oz/ft^2
Wing Loading (loaded)	37.8 oz/ft^2

Table 5.2: Power Plant Specifications

Motor Type	AstroFlight Cobalt 90
Gearbox Type	AstroFlight Super-box 2.75:1
Battery/Cell Type	Sanyo 2400 mAh Nickel Cadmium
Number of Cells	36
Propeller Type	Mejlik carbon fiber
Propeller Dimensions	28 X 12
Maximum Power Output	1600 Watts at 80% efficiency
Power Input	20 Amps
Propeller RPM	2700
Brake Horse Power	1.4
Speed Controller	AstroFlight 204D (60 Amp)

5.4 Stability Analysis:

Aside from flight testing and mere observations, a few students of the team, lead by Aerodynamics Senior Raymond Spall, set out to write a program which would calculate many key stability derivatives. From this analysis, the overall stability of the aircraft could be verified. The program was written in Matlab for ease of operation and plotting. The code is attached to this report, and it's major calculations are tabulated below.

Table 5.3: Stability Derivatives

$C_{x,u}$	-0.2	$C_{w,0}$	0.17313
$C_{x,\alpha}$	0.081	X_u	-0.19895
$C_{z,\alpha}$	-3.7	X_w	0.080573
$C_{z,q}$	-0.39908	X_q	0
$C_{m,\alpha}$	-0.608	Z_u	-0.34444
$C_{m,\dot{\alpha}}$	-0.12256	Z_w	-3.6805
$C_{m,q}$	-0.80786	Z_q	-0.27789
$X_{\dot{w}}$	0	M_u	0
$Z_{\dot{w}}$	-0.00046842	M_w	-0.84672
$M_{\dot{w}}$	-0.0013275	M_q	-0.78753

```
%AIAA (lead by Raymond Spall)
%Stability Derivative Calculation Program
%November 8, 2001

clear all
tic
go_plot = 1;

%Airplane Constants.

w        = 15.5;      %lbf Flying weight
g        = 32.2;      %ft/s2 Gravitational acceleration
S        = 9.3;       %ft2 Wing Area
S_t      = 0.4167;    %ft2 Tail Area
b        = 7;         %ft Wing span
c_bar    = 1.4;       %ft Mean chord
c_t      = 1.0;       %ft Tip Chord
c_0      = 1.4167;    %ft Root Chord
Lambda   = 30;        % Wing leading edge sweep
u_0      = 60;        %ft/s Reference Velocity
theta_0  = 0;         % Reference angle
sm       = 0.16;      % Static Margin
a_t      = 2.2;       % Tail lift curve slope
a        = 3.8;       % Wing Lift curve slope
C_L_0    = 0.281;     % Reference Coefficient of Lift
C_D_0    = 0.1;       % Reference Coefficient of Drag
C_D_alpha = 0.2;      % Overall drag curve slope
rho_ssl  = 0.0023769; %slug/ft3 Air density @ ssl
l_t_bar  = 2.834;     %ft Horz Dist between wing mac and tail mac
h_H      = 0;         %ft Vert Dist between wing mac and tail mac

%Calculated Parameters

m = w/g;                              %slugs Mass
lambda = c_t/c_0;                     % Taper Ratio
A = (2*b)/(c_0*(1+lambda));           %Aspect Ratio
Lambda_c4 = atan(tan(Lambda)-((4*(1-lambda))/(A*(1+lambda))));
K_A = 1/A - 1/(1+A^1.7);              % Wing Aspect Ratio Factor
K_lambda = (10-3*lambda)/7;           % Wing Taper Ratio Factor
K_H = (1-abs(h_H/b))/(2*l_t_bar/b)^(1/3); % Wing Horiz tail factor
de_da = 4.44*(K_A*K_lambda*K_H*(cos(Lambda_c4)^0.5))^1.19; %
V_H = (l_t_bar*S_t)/(c_bar*S);        % Horizontal Tail Volume
i_yy = 0.059;                         %slugs/ft2

%Stability Derivatives (Not u_0 dependent)

C_x_u = -2*C_D_0;
C_x_alpha = C_L_0 - C_D_alpha;
C_x_alpha_dot = 0;
C_x_q = 0;
C_z_u = 0;
C_z_alpha = -(a - C_D_0);
C_z_alpha_dot = -2*a_t*V_H*de_da;
C_z_q = -2*a_t*V_H;
C_m_u = 0;
C_m_alpha = a*(-sm);
```

```
C_m_alpha_dot = -2*a_t*V_H*de_da*(l_t_bar/c_bar);
C_m_q = -2*a_t*V_H*(l_t_bar/c_bar);
X_w_dot = 0.25*rho_ssl*c_bar*S*C_x_alpha_dot;
Z_w_dot = 0.25*rho_ssl*c_bar*S*C_z_alpha_dot;
M_w_dot = 0.25*rho_ssl*(c_bar)^2*S*C_m_alpha_dot;

%Stability Derivatives (u_0 dependent)

k = 1;
u = [30:1:90];  %ft/s  Velocity Vector for Root Locus
len_u = length(u);
for ii = 1:len_u
  C_w_0 = (m*g)/(0.5*rho_ssl*u(ii)^2*S); % Reference Weight coef.
  X_u = 0.5 *rho_ssl*u(ii)*S*(2*C_w_0*sin(theta_0) + C_x_u);
  X_w = 0.5 *rho_ssl*u(ii)*S*C_x_alpha;
  X_q = 0.25*rho_ssl*u(ii)*c_bar*S*C_x_q;
  Z_u = 0.5 *rho_ssl*u(ii)*S*(-2*C_w_0*cos(theta_0) + C_z_u);
  Z_w = 0.5 *rho_ssl*u(ii)*S*C_z_alpha;
  Z_q = 0.25*rho_ssl*u(ii)*c_bar*S*C_z_q;
  M_u = 0.5 *rho_ssl*u(ii)*c_bar*S*C_m_u;
  M_w = 0.5 *rho_ssl*u(ii)*c_bar*S*C_m_alpha;
  M_q = 0.25*rho_ssl*u(ii)*(c_bar)^2*S*C_m_q;

  %System Matrix Definition

  MAT{ii} = [X_u/m                              , X_w/m                              , 0
, -g*cos(theta_0)                   ;
        Z_u/(m-Z_w_dot)                        , Z_w/(m-Z_w_dot)                     , (Z_q + m*u(ii))/(m-
Z_w_dot)                 , -m*g*sin(theta_0)/(m-Z_w_dot)              ;
        (1/i_yy)*(M_u + (M_w_dot*Z_u)/(m-Z_w_dot)), (1/i_yy)*(M_w + (M_w_dot*Z_w)/(m-Z_w_dot)),
(1/i_yy)*(M_q + (M_w_dot*(Z_q+m*u(ii)))/(m-Z_w_dot)), -M_w_dot*m*g*sin(theta_0)/(i_yy*(m-Z_w_dot));
        0                              , 0                              , 1                              , 0
];
  [eig_vec{ii},eig_val{ii}]  = eig(MAT{ii});
  for j = 1:4
    X(k) = real(eig_val{ii}(j,j));
    Y(k) = imag(eig_val{ii}(j,j));
    k = k+1;
  end
end
omega_phu = abs(imag(eig_val{31}(1,1)));
omega_sp= abs(imag(eig_val{31}(3,3)));

t = toc;

%Plotting Routine
if go_plot == 1
for l = 1:4
  h1 = plot(X(l),Y(l));
  set(h1,'LineStyle','none','Marker','o','MarkerEdgeColor','k',...
       'MarkerFaceColor','k');
  hold on
end
h1 = plot(X,Y);
set(h1,'LineStyle','none','Marker','.','MarkerEdgeColor','k',...
  'MarkerFaceColor','k','MarkerSize',6);
```

```
hold on
for l = k-5:k-1
  h1 = plot(X(l),Y(l));
  set(h1,'LineStyle','none','Marker','s','MarkerEdgeColor','k',...
    'MarkerFaceColor','k');
  hold on
end

hold off
grid on
title('Root Locus Plot');
xlabel('Real');
ylabel('Imaginary');
end
```

6. Manufacturing Plan:

Manufacturing ability is very important to consider when designing an aircraft. Unless built properly, even the best design will fall apart in the air. In many cases, the same design will be out performed by a worse design built right. For this matter, it was necessary to completely weigh all the options, and assure that the aircraft was built properly. From the first days of the conceptual design, the aircraft-manufacturing plan was developed. Manufacturing was kept in mind during the entire design process so that the aircraft would be easy to produce. This was lesson learned the hard way during the 2000 DBF competition when our aircraft design called for too many exotic manufacturing practices. Although the team already had a plan for manufacturing the aircraft based on previous experience, a figure of merit analysis was completed anyway.

For the figure of merit determination, five different material properties were considered; availability, skill level required, time required, reliability and cost. Each material received a score between zero and three depending on its fulfillment of the property. The intent of the figure of merit analysis was to see which materials had the highest total. For this reason, if a material had a high availability, it was giving a score of three. Conversely, if a material didn't require a much skill to assemble it, it scored a three. Similarly, if a material didn't require much time to work with, it scored a three.

The results of the figure of merit analysis indicated that the best material to use for each aircraft component. For the wing, the material that scored the highest was foam with a spar. For the spar, aluminum scored the highest. This is primarily due to the fact that both aluminum and foam are readily accessible, cheap, and easy to work with. Composites may increase performance, but they are more expensive, require more time to lay up, require more experience since they are harder to work with, and cost significantly more. For this aircraft design, the team opted to cut the wings out of white foam, run an aluminum spar all the way to the wing tips, and bolt into place.

For the fuselage and empennage, a wooden built up composition scored the highest. This is primarily for the same reasons above. Just as for the wings, the team agreed to build the fuselage as the figure of merit analysis suggested. In order to keep the aircraft as light as possible, the team agreed to use a balsa and plywood frame to support the majority of the loads. White foam cut to shape and epoxied wherever necessary to increase the rigidity of the frame. As a result, the fuselage was lightweight and just strong enough to perform its mission. The fuselage was also fabricated with a fiberglass and kevlar honeycomb. It was found that this material was very strong and quite light. It was added along the length of the fuselage to provide a keel that would keep the fuselage rigid and vibration free. It also replaced heavy balsa wood components like the engine mount and landing gear block. Incorporating additional composite materials are still being considered. An alternative aircraft made almost completely of carbon fiber is being produced at this point. It will be tested and compared to the current design. Depending on which plane will provide the highest score, the competition plane will be chosen accordingly.

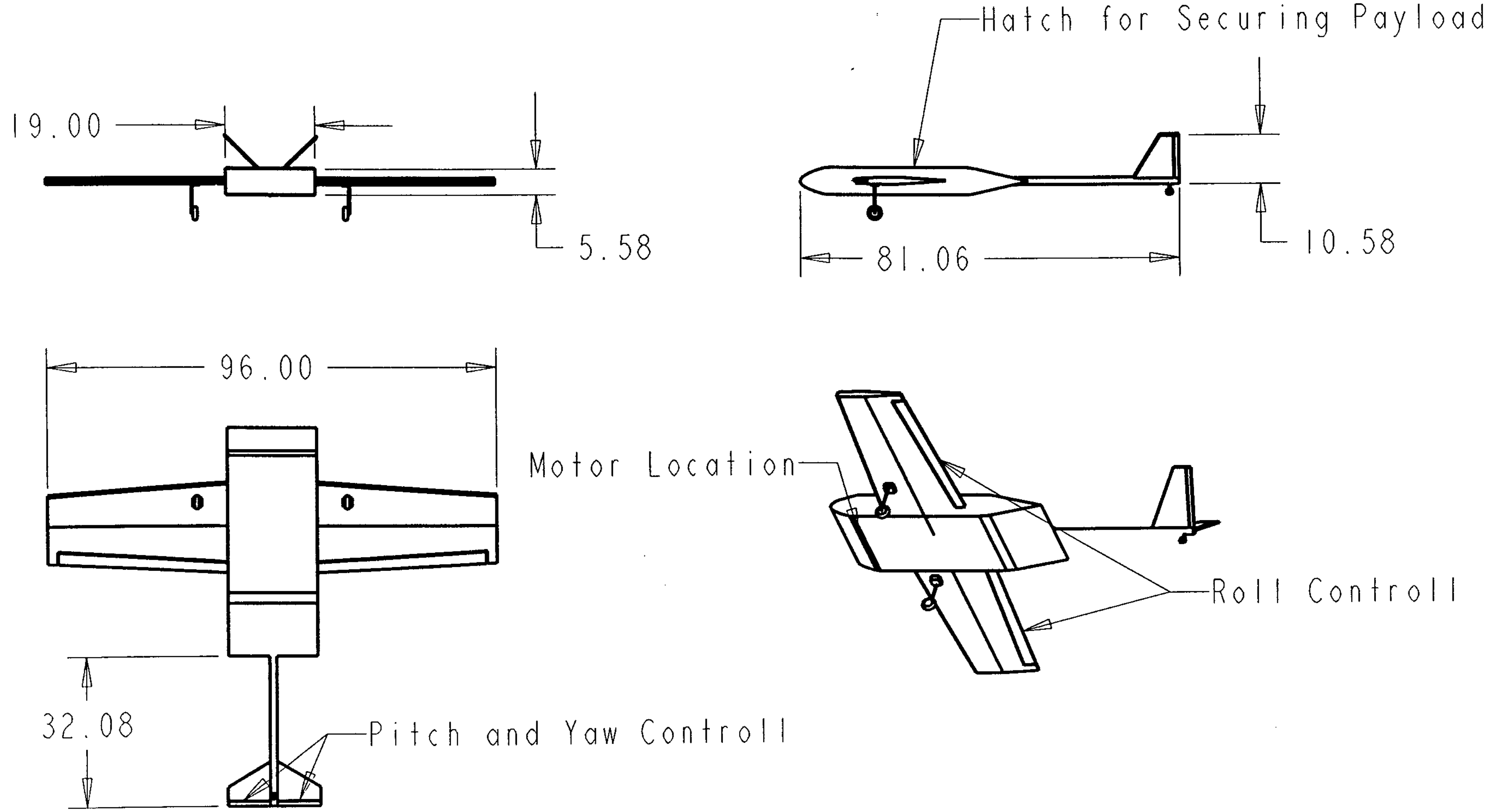
The landing gear was the only acceptation the team made when deciding how best to manufacture a component. Even though the aluminum scored higher, the University of Arizona team decided that a carbon graphite component would perform better. In the past competitions, this team had had much success with the landing gear design. The team used 30 layers of carbon graphite and fiberglass to make a lightweight, strong, yet springy landing gear. Although the carbon is more expensive and more difficult to use, the team decided that the finished carbon product would be better.

Figure 6.1
Figures of Merit

		Availability	Required Skill Level	Time Required	Reliability	Cost	Total
Wings	Foam w/ Composite	0	2	2	2	1	**6**
	Composite	0	1	1	2	1	**4**
	Foam w/ Spar	2	3	3	3	3	**11**
Wing Spar	Composite	1	1	1	2		**5**
	Wood	3	3	3	1		**10**
	aluminum	2	3	3	3		**11**
Tail boom	Composite	1	2	2	3	1	**9**
	Aluminum	3	2	2	3	2	**12**
Landing Gear	Composite	1	2	2	3	1	**9**
	aluminum	3	2	2	3	2	**12**
Fuselage	Composite	1	1	1	1	1	**5**
	Wooden built up	3	2	2	2	3	**12**
	Foam and Composite	2	1	1	2	3	**9**
	Foam	1	2	3	1	1	**8**
Empenage	Wood	3	3	2	3	3	**14**
	Composite	1	1	1	3	1	**7**
	Foam	3	2	3	1	2	**11**

Figure 6.2
Manufacturing Gant Chart

Component Manufactuered	Start	Finish	September	October	November
Wing					
Cut Wings and Insert Spar	9/1/01	9/8/01			
Cover and insert electronics	9/8/01	9/15/01			
Finish and Cover	9/15/01	9/22/01			
Fuselage					
Build frame	9/29/01	10/6/01			
Insert foam and spar mount tube	9/29/01	10/6/01			
Install motor and gear mounts	10/13/01	11/3/01			
Create tail mount	10/13/01	10/20/01			
Create Hatch	10/13/01	10/20/01			
Finish and Cover	11/3/01	11/10/01			
Tail Boom					
Cut and cover tail	10/20/01	10/27/01			
Mount tail to boom	10/20/01	10/27/01			
Install boom into aircraft	10/20/01	10/27/01			
Landing Gear					
Cut out composite strips	10/27/01	11/3/01			
Composite layup	11/3/01	11/3/01			
Remove from mold and finish	11/3/01	11/3/01			
Final Preparation					
Install electronics	11/3/01	11/10/01			
Cover and Finish	11/3/01	11/10/01			



University of Arizona

2002 AIAA DBF Entry- Evolution

Units in Inches

AIAA Design Build Fly Competition

2002

CLARKSON
UNIVERSITY

Knight Hawk

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1. Executive Summary

1.1 Summary of Development

The Knight Hawk project for the Design, Build, Fly Team of 2001-2002 started with the fall 2001 semester at Clarkson University. The project is an extracurricular activity at Clarkson University, comprised of undergraduate students at all levels of education. The team's development of the aircraft was divided into two stages, which include the design and analysis phase, and the manufacturing stage. The conceptual design phase determined the aircrafts baseline configuration by weighing benefits versus costs for several proposed configurations. The team reduced the configuration possibilities to a totally new concept compared to the previous two year's canard designs. The new conventional designs investigated provided a rather easily constructed and stable design. After a cost/benefit review of four configurations, a somewhat conventional design with a V-tail was chosen. After selection of the aircraft type, the team moved into the preliminary design phase. The majority of the fall semester was used to perform analysis on the potential design and obtain the optimum configuration for the mission specifications.

With the completion of the detailed design phase late in the fall semester, construction began before the end of the semester. Initial layouts for the tooling to make the wing and fuselage were created and the materials were ordered. With the start of the spring 2001 semester construction resumed. The team continued to make modifications to the aircraft and solve problems encountered when moving from the design phase to the manufacturing phase. The semester was also used to prepare the necessary reports for the competition and test the aircraft's actual performance and stability characteristics in flight.

1.2 Alternative Designs

The production of adequate lift to minimize takeoff distance and maximize payload capacity was the major driving design parameter. A secondary design parameter that wasn't deemed crucial but still important was the ability for the aircraft to fly fast since time was now a factor in scoring. To produce an aircraft that has a limited amount of power available, lifting surface configurations took the highest priority in the design research. Larger wing area would reduce the wing loading, thereby decreasing the stall speed. Since a lower stall speed decreases the takeoff speed, less power is required during the takeoff run. Takeoff is the active constraint when sizing motor power so a smaller motor can be used. The decrease in weight results in less required lift.

Several configurations were chosen for evaluation including a twin fuselage, conventional high wing, conventional low wing, twin engine and a ducted fan. Primary concerns were lifting area, flight stability, airframe strength, ease of construction, aircraft speed and time of building. Other lesser concerns included maneuverability and drag. The team selected the V-tail conventional configuration during the conceptual design. It was believed that a stable, controllable aircraft could be made with the V-tail design.

During the preliminary design phase, the airfoil was chosen for the wing, the fuselage layout was constructed and motor research was conducted. During the latter part of the preliminary design phase, a new fuselage concept was suggested. After a short time of weighing the advantages and disadvantages of the new configuration, it was decided that fiberglass fuselage was a more light weight design and it was adopted.

Detailed design included structural layout of the aircraft, final motor and battery selection and final sizing of control surfaces. It was decided that a balsa wood built-up structure would be the best manufacturing method and would produce quality wings. Several aspects of the mission performance were looked into greater detail, as well as the stability of the aircraft. Also weight estimates were refined and the structural loading of the aircraft were looked into. Finally, the rated cost of the aircraft was tweaked and finalized.

1.3 Design Tools

Several computer-based design tools were employed to aid in the different stages of the aircraft development. Among them were:

- Microsoft Excel
- Computervision DesignView
- Autocad 2000
- XFOIL by Mark Drela
- JAVAFOIL
- Laminar Research X-Plane

Microsoft Excel was used to compute lift and drag data, set up center of gravity calculations, and size the tail fins and control surfaces.

DesignView is a parametric 2-dimensional drawing program that was used for wing and fuselage layout. Its programming features allowed for the resizing of the airframe by a simple change of one dimension. It was also used to calculate the neutral point for the aircraft.

Autocad was used to provide a 3-dimensional view of parts to be constructed. It aided in the construction process by providing a real-life simulation of part fitting.

XFOIL is a Unix-Windows based 2-dimensional airfoil analysis software. It provided a numerical estimation for various proposed airfoils. It was used to compute lift, drag and pitching moment in both viscous and non-viscous flow modes.

Similarly, JAVAFOIL based in the Java programming language was used to verify the results of XFOIL. The program also was used to edit existing airfoils to optimize them for the task at hand.

Laminar Research X-Plane is a software package that provides a design environment to create an aircraft and a flight simulator to test the aircraft. The software allows for the entry of drag data; airfoils; wing, fuselage and empennage geometry; center of gravity and propulsion placement and size. Using

blade element theory the code breaks the aircraft down, performing aerodynamic calculations 15 times per second. The code also includes data output that allows the designer to see the motion of the aircraft. Several variables can be output to determine the stability and performance of the design. Also, many different aircraft views are available to allow the pilot to fly the aircraft from the exterior. The program was used to train students in the operation of model aircraft.

2. Management Summary

2.1 Architecture of the Design Team

This year the team is comprised of several members with varying abilities and commitment. Well most of the team are returning members that have a pretty good idea of the way that the team has worked in the past we do have a few new freshman that are on the team. This year the team has decided to function as a single group instead of breaking up into small sub groups for the design phase. This way the newer people on the team can get a taste for the project as a whole instead of showing them a narrow part of the process. During the preliminary and conceptual design phases the team leaders acted as project managers and the other team members would volunteer for different areas to research. After researching their part it would then be that team member's job to lead the next meeting to teach to group and lead discussion. This would allow everyone to actively participate in the design process. It was then the team leader's reasonability to lead the group as we assembled a final design. The Major contributors are as follows:

Jason Kuntz, a senior Aeronautical Engineer, was given the role of engineering leader and shared team leadership with Russ Zea. Jason was in this position because of his past experience with both the DBF competition and leadership on campus. Jason was involved heavily in the design and building process, along with helping the younger team members stay involved with the design process.

Russ Zea, a senior Interdisciplinary Engineering and Management major, was Given the role of project manager along with sharing the team leadership role. Russ plays a large role in making sure the team stays within their given budget along with keeping the team close to the timeline that was the team set up.

Aroosh Nagvi, a sophomore Aeronautical Engineer, plays a big part in the manufacturing process of the plane. Aroosh has brought years of model aircraft experience to the team which allows us to construct the aircraft to maximize the strength of the aircraft well minimizing weight. Aroosh is also the team's pilot.

Chuck Traill, David Young, Jason Camps, Matt Feldman, and Matt Bennington contributed in the analysis and construction of the aircraft. These members were greatly involved in the project, spending many hours to ensure the completion of the aircraft.

Dr. Ken Visser, a professor in the Mechanical and Aeronautical Engineering Department at Clarkson University and graduate student Matt Duquette served as advisors for the Knight Hawk team.

2.2 Assignment Areas

This year's team had five main areas of concern that team members were assigned to. These areas included stability and control, aerodynamics, power system, wing configuration, and fuselage structure. The stability and control area along with the aerodynamics areas were assigned to Jason Camps. Jason used the program X-Plane to generate a complex model of the plane for the detailed

design. The X-plane software allowed Jason to actually fly the plane and provided several modes of analysis and output for the aircraft. Jason also investigated potential airfoils and the different aerodynamic advantage of each potential design. He used the program X-Foil to provide a numerical estimation for various proposed airfoils. The program was used to compute lift, drag and pitching moment in both viscous and non-viscous flow modes.

The power system area was taken by Russ Zea. He coordinated the research and gather of both the propulsion and power supply for the aircraft. These members also were vital in the integration of both the engine and battery packs into the desired design.

The wing configuration area was assigned to Aroosh Nagvi. Since Aroosh had the most experience with building wings for model aircrafts he felt that he would be the best one to lead a group in laying out the design for the internal structure of the wing.

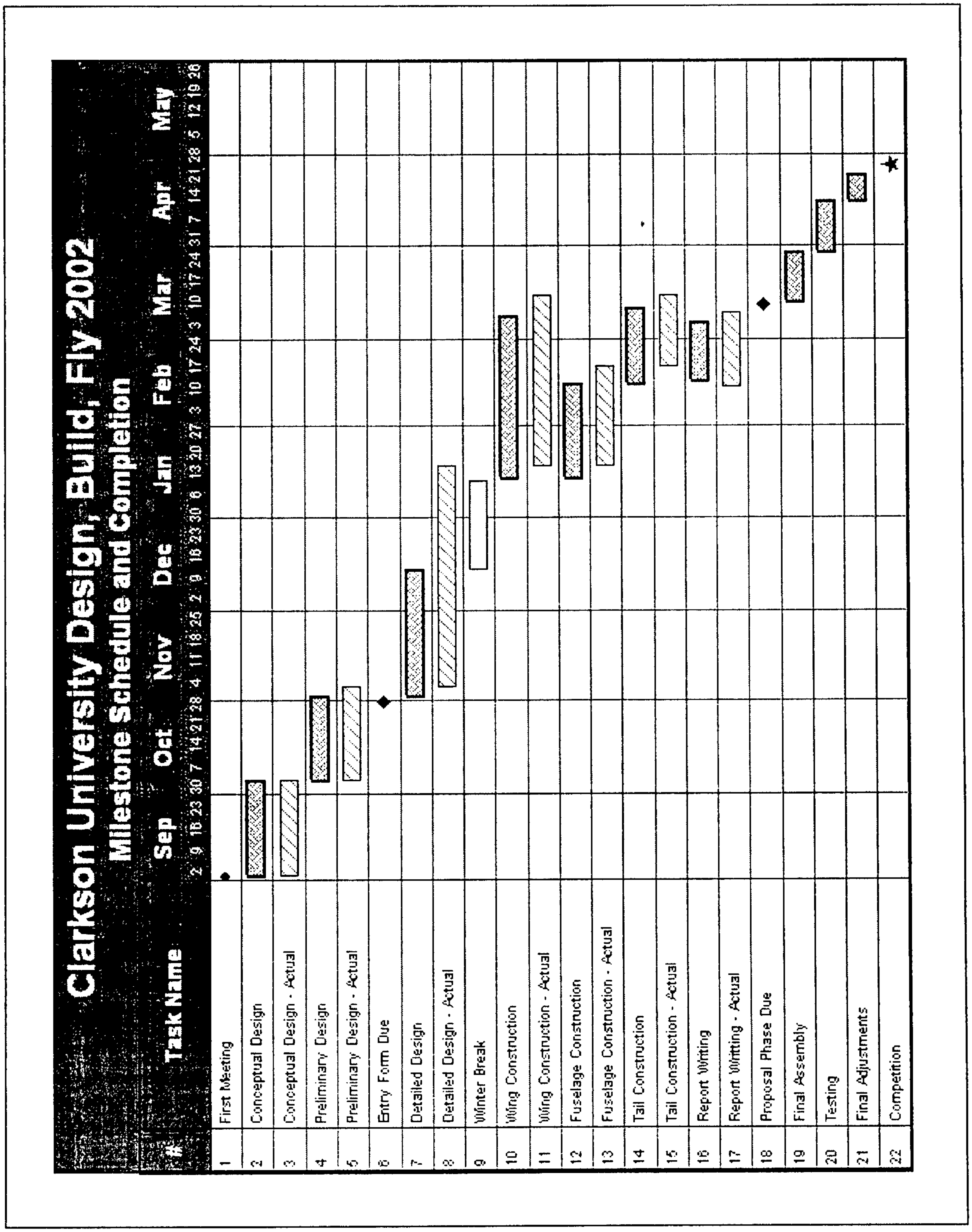
The fuselage structure area was assigned to Chuck Traill. This year we were looking into building the plain out of a lightweight material like fiberglass. To be able to do this Chuck lead a small group of people in researching how we could use fiberglass in the construction of the fuselage.

Though these areas were assigned to certain team personnel, the team as a whole actively communicated with each other providing and included every member in each area of assignment.

2.3 Milestone Chart

The Milestone Schedule and Completion Chart in Figure 2.1 shows the timeline that was established at the beginning of the year and how it compares to the actual time that we worked on this project.

Figure 2.1 Milestone Chart



3. Conceptual Design

A large part of the design process is taking past experiences, both good and bad, and applying them to the problem at hand. Before designing this year's plane the team reviewed the difficulties and downfalls of last year's entry. The team decided that speed, controllability, and a more conventional design were necessary parameters. Speed is a major component of a high score. After last year's plane's sluggish performance, the team decided that this year's plane had to be faster. By designing something lighter the speed problem would be easily remedied. The lack of handling and control of last year's plane caused major concern for the pilot. To overcome this obstacle this year's team brainstormed a more conventional approach with wider control surfaces instead of long, narrow ones.

3.1 Figures of Merit

3.1.1 Figures of Merit Factors

Rated Aircraft Cost - This is the most highly weighted factor on the FOM table. The team weighted it so highly because of its significance in the rating system of the competition and its implications on our budget.

Ease of Construction - Construction is one of the main components of the competition. Therefore it had to be a major consideration of the team. This way members with limited experience could participate in construction.

Man Hours Required - The hours required to build a working design has a low weight due to it being spread out over a large span of time. If the construction is scheduled properly, this should not be a very large factor, although there must be some restrictions on time.

Speed of Aircraft - As stated above, lack of speed was a major downfall in last year's design. Therefore, this year it was rated higher so that those issues would be directly addressed and failures could be avoided.

Controllability - Another problem with last year's plane was the inability to maneuver well. To alleviate this problem, controllability was a major issue the team discussed while brainstorming alternatives.

Table 3.1: Figures Of Merit

Objectives	Weight	V-tail Score	Utility	Twin Fuselage Score	Utility	Dual Prop Score	Utility	Ducted Fan Score	Utility
Rated Aircraft Cost	25	10	250	6	150	7	175	8	200
Ease Of Construction	20	9	180	7	140	7	140	6	120
Man Hours Required	15	8	120	7	105	8	120	7	105
Speed of Aircraft	20	7	140	8	160	8	160	9	180
Controllability	20	9	180	7	140	6	120	8	160
Total			870		695		715		765

V-tail has 1 prop
Twin Fuse. Has 1 prop
Ducted fan has 1 prop

3.2 Alternative Design Overview

V-tail - The v-tail design was a departure from a conventional aircraft in order to address the problems of maneuverability. The design is comprised of a single fuselage with a single prop located in the nose of the plane. It has a high-aspect wing ratio and a high wing.

Twin Fuselage - The twin fuselage was considered for its interesting configuration and it was based off a working design from a previous year. In this design the cargo could be set on either side of the engine, which is mounted on a central body.

Dual Prop - The dual prop design was just that. A conventional aircraft with 2 engine mounted props instead of the basic one. It was designed for its speed and power. It was modeled after the configuration of World War 2 bombers and how they held their payloads.

Ducted Fan - The ducted fan configuration was chosen to follow up on the uniqueness of Clarkson aircraft. This design was chosen because of its speed and maneuverability. It would have one engine with air ducts running down the fuselage on either side.

3.3 Evaluating Alternative Designs

The v-tail was found to have the highest total utility therefore making it the most feasible design. It had a rated aircraft cost that was lower than all other designs. The construction of the aircraft would be straightforward and easily manageable. With the construction period strewn out over a lengthy period of time, it would be easy to finish and test within the time constraints. It takes into account the need for speed and would be easily controllable.

The twin fuselage was found to have the lowest total utility. It had an extremely high rated aircraft cost in comparison to all other designs. It would be difficult to build and be sluggish in controllability. The

plane would be as fast as the v-tail but would take longer to construct because of the difficulty in designing.

The dual prop design had a moderate rated aircraft cost. However, it too would not be very easily controlled and would be sluggish in maneuverability. The power and speed would be increased with the addition of a second prop, but that also makes the design harder to construct. Therefore that would also increase the time needed to build it.

The ducted fan also had a moderate rated aircraft cost. It would be faster than all the other designs because of the ducted fan. The ductwork would make this plane much more difficult to build and therefore greatly increase the man-hours needed to assemble it. The plane would be as easily controlled.

3.4 Rated Aircraft Cost

Since rated aircraft cost is a big part of the scoring system, it was desired that the chosen configuration have the lowest possible cost. Both the V-tail and the Ducted Fan concepts were comparable on cost, however, since the Ducted Fan did not offer that much of an advantage on the V-tail, the V-tail was chosen.

Table 3.2: Rated Aircraft Cost

Description	Cost	V-tail Amt.	Val.	TF Amt.	Val.	DP Amt.	Val.	DF Amt.	Val.
Wing	8hours/ft.	9 ft.	72 hours	9 ft.	72 hours	9 ft.	72 hours	9 ft.	72 hours
Chord	8hours/ft.	1.5 ft	12 hours	1.5 ft.	12 hours	1.5 ft.	12 hours	1.5 ft.	12 hours
Cont. Surf.	3hours/surf.	2 surf	6 hours	2 surf.	6 hours	2 surf.	6 hours	2 surf.	6 hours
Fuselage	10hours/ft.	5.5 ft.	55 hours	11 ft.	110 hours	5.5 ft.	55 hours	5.5 ft.	55 hours
Hor. Fins	5hours/fin	0	0	1	10 hours	1	10 hours	1	10 hours
Vert. Fins	5hours/fin	0	0	2	20 hours	1	10 hours	1	10 hours
V-tail	15hours/v-tail	1	15 hours	0	0	0	0	0	0
Servos	5hours/servo	5	25 hours	5	25 hours	5	25 hours	5	25 hours
Engines	5hours/eng.	1	5 hours	1	5 hours	2	10 hours	1	5 hours
Propellers	5hours/prop	1	5 hours	1	5 hours	2	10 hours	1	5 hours
Total Hours			195 hour		255 hour		210 hour		200 hour
Total Cost			13.9		15.1		16.1		14.0

4. Preliminary Design

4.1 Design Parameters

Several design parameters were considered in the preliminary design of the aircraft. These included:

- Weight Estimation
- Airfoil Selection
- Wing Planform
- Propulsion System
- Fuselage Design
- Tail Sizing

4.2 Weight Estimation

The first crucial parameter investigated in the preliminary design phase was the initial weight estimation. Based on last year's entry, estimates were made for the unknown components of this year's design, and these were taken along with the known components to get the initial weight estimate. Since the design was to carry the maximum amount of softballs at 24, the payload weight was specified as approximately 10 pounds. Next, the design was to use the full amount of batteries so the "fuel" weight was 5 pounds. Using the same motor from last year resulted in an additional 3 pounds. Taking into account the very heavy wing from last year, it was estimated that the remaining weight would give a total airframe weight of approximately 30 pounds. This number was increased to 35 pounds to take into account any components that may not have been thought of and to ensure that the wing designed would be able to lift the aircraft.

4.3 Airfoil Selection

The airfoil selection process began with choosing several airfoils to benchmark, with the best one being taken and tweaked to be the most efficient airfoil possible. These were chosen based on the following figures of merit; lift to drag ratio, maximum lift coefficient, and stall characteristics. Upon completion of this process, the NACA 6412 was chosen as the best model to base the new airfoil on. Using the program JAVAFOIL in conjunction with XFOIL, the NACA 6412 was modified to improve its characteristics at low speeds. Shown in figures 4-1 and 4-2 are both the aerodynamic efficiency and lift/drag polar plots from both programs. Though the XFOIL numbers seem to be slightly off, it was determined that since the program was developed for higher Reynolds number flows, the low Reynolds number used could be making the difference.

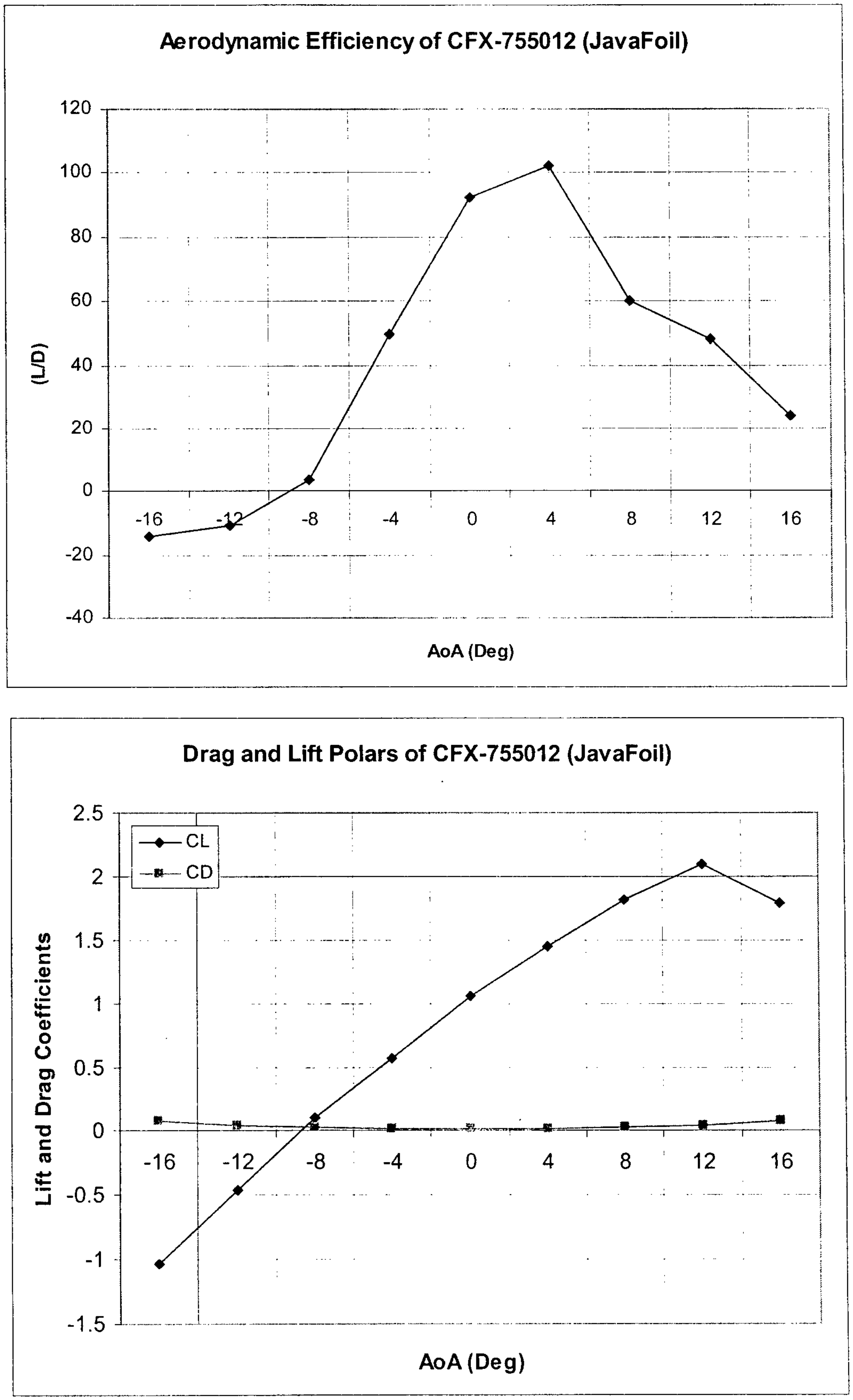
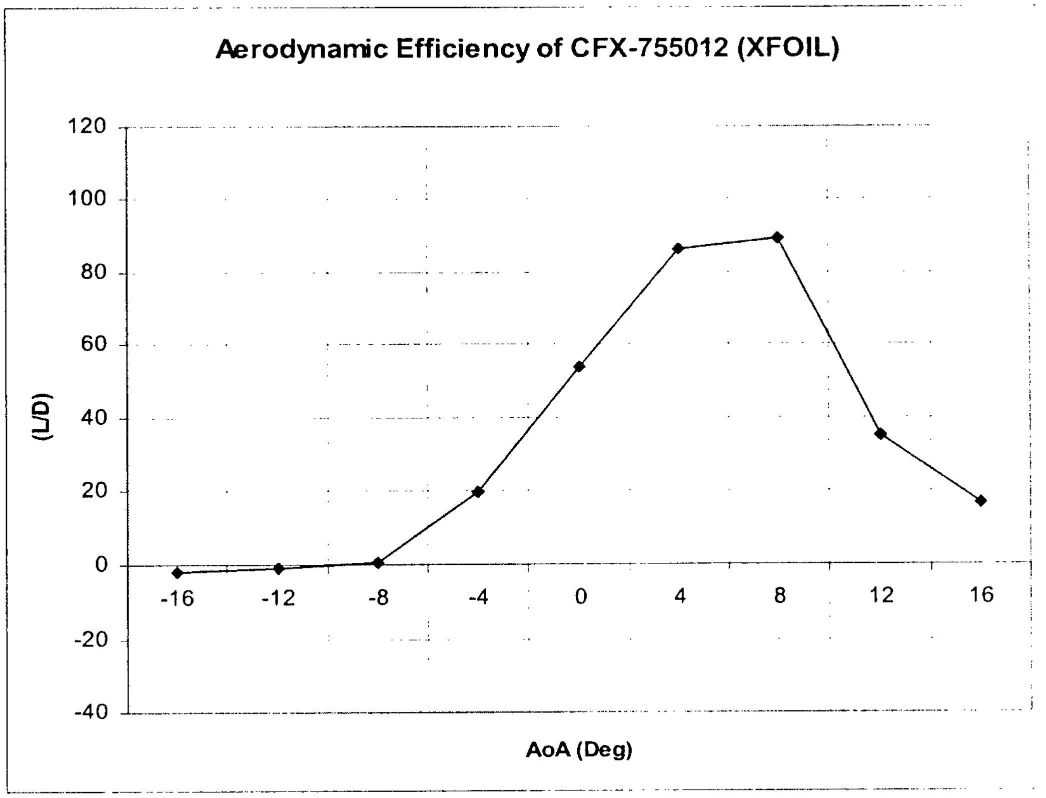


Figure 4-1: JAVAFOIL Airfoil Characteristics



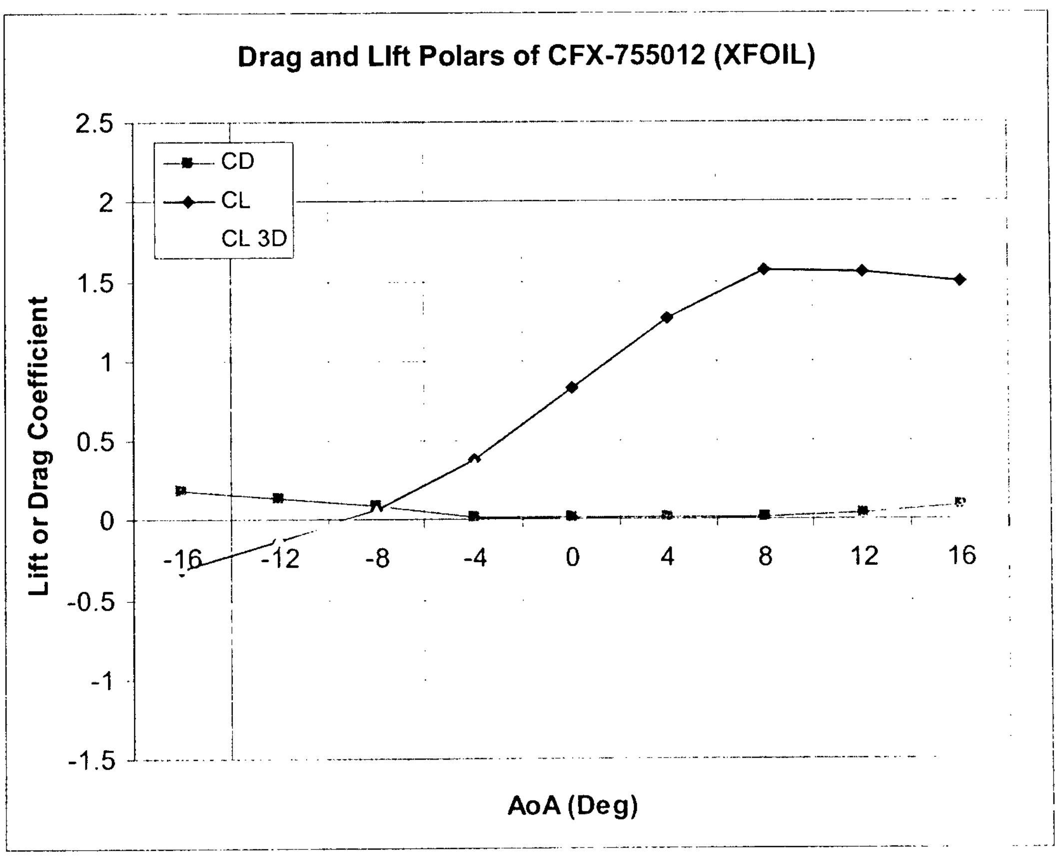


Figure 4-2: XFOIL Airfoil Characteristics

A similar analysis was done for the NACA 6412 using both programs. The lift curve in this case from XFOIL was not linear from an angle of attack from 0 to 10. Based on this, it was said that the JAVAFOIL numbers at this Reynolds number were more reliable and that the design would be based on them.

4.4 Wing Planform

In designing the wing planform, three main factors needed to be determined. These included the taper ratio, wing area and wing chord. The span in an attempt to keep the cost of the aircraft down was reduced from last year's entry to 9 ft.

4.4.1 Taper Ratio

A taper ratio of approximately 0.6 was initially considered in an effort to keep drag down and try for an efficient elliptical lift distribution. However this was determined to increase the time to build the wing and that there would not be that big of a drag increase by keeping the wing rectangular. Thus the taper ratio was chosen as 1.

4.4.2 Wing Area

The wing area was determined using the equation for lift coefficient shown below in equation 4.1.

$$C_{L\min} = \frac{2L}{\rho S_w V_{\max}^{2}} \qquad (4.1)$$

For straight and level flight, the lift is assumed to be equal to the weight. In conjunction with the lift coefficients calculated from the airfoil work, a wing area matrix was put together to calculate the necessary wing area based on the flight speed and weight of the aircraft. Based on last year's entry, a cruise flight speed for this year's aircraft was assumed to be on the order of 25 knots. From the lift curve created by JAVAFOIL, a lift coefficient at 0 degrees angle of attack was found to be around 1.1. Matching these values up on the wing area graph gave an area of approximately 13 sq. ft. The wing area graph is shown on the following page as figure 4-3.

4.4.3 Wing Chord

Once wing area was determined, and already having the span specified, the chord could be determined. It was calculated as 1.5 ft. for the necessary wing area. Inadvertently, this resulted in an aspect ratio of 6. Although a higher aspect ratio would have been desired to decrease induced drag and help in the glide characteristics of the aircraft, it was not a driving parameter of the design.

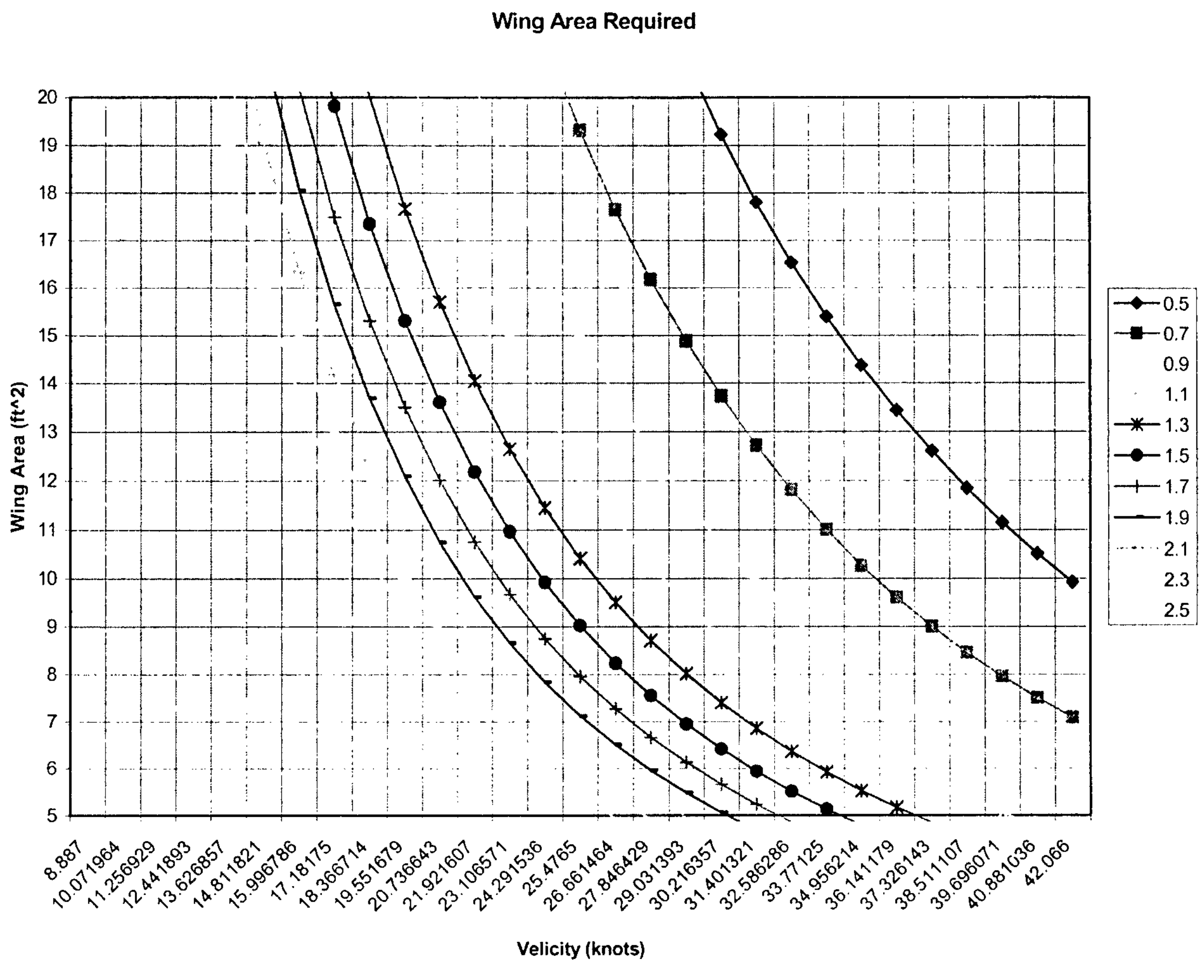


Figure 4-3: Wing Area Required Plot

4.5 Propulsion System

4.5.1 Motor

Due to the competitions continued requirement for Astroflight or Graupner motors, the team decided to try and keep the motor used last year. Having more than enough thrust for the heavy aircraft of last year's entry, and since this year the plane would be much lighter, it was determined that the motor from last year would be sufficient to power this years entry.

4.5.2 Batteries

To increase the endurance of the aircraft there either needs to be an increase in current to the motors or and increase in the number of cells in the battery packs. Since current is set by the rules at 40 amps, the only way to maximize endurance would be to maximize the number of cells. To meet the maximum battery weight criteria of 5 pounds, there can be a maximum of 36-40 cells depending on the weight of each cell. Inspecting numerous battery types resulted in a set having a power density of 3000 mAh along with a total number of cells of 38.

4.5.3 Propeller

To determine the necessary propeller speed, the stall speed is required. A rough calculation given by equation 4-2, shows that the stall speed is approximately 24 mph. We know propeller speed should be about 3 times the stall speed so the propeller speed is then determined to be approximately 72 mph. Looking up the motor specs, it is listed that the Astroflight motor to be used is capable of spinning the propeller well above the necessary speed. Based on this speed then, a required propeller diameter can be found. In an effort to reduce the propeller diameter and keep the landing gear relatively short, three blade propellers were investigated. A propeller of 18 inches diameter was chosen for this application. The thrust provided by this propeller was then found to be approximately 12 pounds.

4.6 Fuselage Design

The driving idea behind the fuselage was to minimize drag as much as possible. Therefore, the main figure of merit most looked at was the fineness ratio of the fuselage, that is, the ratio of the height to the length. Other minor figures of merit that were taken into consideration included; the ease of construction, structural stability, and also the amount of wasted space was to be kept a minimum. The fuselage was then sized by the required area necessary for the number of softballs to be carried. It was decided that the rows should be kept short so to make the shift in center of gravity less subtle when the balls are moved. Also, to minimize cost in the wing span of a blended fuselage configuration, the maximum 18 inch wide fuselage wanted to be utilized. To achieve this and leave room for necessary equipment, a ball configuration of 3 columns by 8 rows was used. This gave a required cargo area of 1536 sq. in. Since staggering balls and stacking them more than 2 high was against the rules, a round

fuselage was not chosen and a flat fuselage was decided to be used instead. It was hoped that if shaped properly, some additional lift may be generated by the fuselage during rotation, aiding in takeoffs and landings. This would be the baseline configuration for the fuselage. To support the radio gear and motor, a tapered front would be added to the original box to contain the softballs in the desired configuration. To provide the required tail arm, two different concepts were investigated. The first involved extending the fuselage aft the required distance and have the V-tail join at the base of the tails into the one fuselage. The second concept had the V-tails joining at their tops, and then having the bases join to two booms that would extend from the fuselage aft. Because two booms would be less material, it would be less weight and result in an overall lighter aircraft. Also having the added space in between the booms would reduce the amount of skin drag with less wetted area. Since an inverted V-tail works with equal effectiveness as the normal V-tail, this was the design chosen.

4.7 Tail Sizing

The main figure of merit taken into consideration for the tail design was stability and controllability. It was desired that there would be enough control authority for the pilot to be able to correct any disturbances during the flight that might occur. To calculate the necessary volume for each part of the V-tail, the following relation for converting horizontal and vertical tails to V-tails was used.

$$A\_horizontal = A\_vtail * [\cos(angle)]^2 \quad (4.2)$$

For the preliminary design, the common approach that the horizontal tail area is approximately 25% of the wing area gave a required horizontal tail area of 3.4 sq. ft. Since the desired angle for the V-tail was an even 45 degrees, the equation above was used to find the area of the V-tail as 6.8 sq. ft. for both surfaces. Knowing that the pitching moment coefficient of the wing airfoil is -0.107 and that the pitching moment of the tail has to be greater, we can find the required tail arm. This gave a tail arm of 3.85 ft.

5. Detail Design

With all of the initial numbers calculated in the preliminary design, an X-Plane model of the aircraft was created to get a feel of how the plane would react under flight conditions as well as what would be the major design features that would need revising. Upon completion of the virtual test flight the only real problem realized was the pitch control and longitudinal stability. This problem was solved by increasing the tail arm and will be discussed later in the detailed design.

A detailed performance analysis is presented in this section. Analyzed are the drag estimation of the aircraft, the takeoff distance and the handling or stability qualities of the aircraft. From this the mission performance is estimated and the aircraft weights are also finalized. Critical components of the structure are documented with the systems architecture. Also, the rated aircraft cost is refined and finalized. Detailed drawings of the proposed design are available at the end of the section.

5.1 Aircraft Drag Estimation

Initial drag calculations were performed in order to predict power requirements in all phases of flight, and takeoff distance. The drag analysis can be seen in its tabulated form in Table 5.1.

Table 5.1

Drag Analysis			
	C_{do}	Area (ft^2)	Drag Area
Fuselage	0.3000	4.50	0.15000
Wing	0.0294	13.5	0.58800
Tail	0.0200	0.07756	0.00039
Gear	0.2000	0.04167	0.08333
Sub Total		18.11923	0.93932
Interference	10%		
Total		Drag Area ->	1.03325

Wing drag was based on 2-dimensional wing theory as determined by Calcfoil, corrected for 3-dimensional effects. An interference drag addition of 10% was added to the overall drag estimation. The estimated drag in cruise flight at a zero angle of attack is approximately 2.9 lbs at 35 mph. The drag was cross-checked using Motocalc's built-in drag estimation feature. A model was configured as a monoplane with an equivalent horizontal and vertical tail to replace the V-tail used in the real design. Despite the difference, Motocalc results showed drag within 10% of the analytic values at stall speed. Fuselage drag data is based off a report published by Hewitt Phillips and Bill Tyler, "Cutting Down the Drag." A series of tests in an MIT wind tunnel provided data for several model aircraft shapes. An interpolation between the different shapes was used to estimate the drag coefficient for the fuselages of the aircraft. Landing gear was assumed to be a flat plate. The vertical tail was considered a NACA 0008 airfoil due the leading and trailing edge taper.

5.2 Stall Speed

The stall speed was determined using the equation for coefficient of lift and the data from the airfoil analysis. From the airfoil analysis, the maximum lift coefficient seems to be around 2.1. With the full payload a stall speed of 24 mph was calculated. This value seemed slightly high but it was deemed acceptable.

5.3 Takeoff Performance

A takeoff analysis, based on Anderson's method, shows ground roll for the aircraft to be estimated as the following equation:

$$Sg = 1.21(W/S)/g\rho Cl(T/W)$$

Equation 5-1

From the propeller selection in the preliminary design, the thrust is specified as approximately 12 lbs along with the weight and wing area to get a result for takeoff roll of approximately 108.53 ft. Assumptions include straight and level flight since the aircraft can be said to be moving along the runway at flight conditions. There also assumes no headwind or ground effects during takeoff. This would be acceptable as the takeoff distance is within the 200 ft. allowed.

5.4 Turning Analysis

Using turn analysis from Anderson, the turn radius at a load factor of 1.2 is 64 feet. For a load factor of 1.5 is 47 feet. Since stall speed increases by the square root of load factor, the stall speed in a fully loaded configuration increases to 25 mph at a load factor of 1.2 and to 28 mph at a load factor of 1.5. The bank angle of a 1.2 g turn is 34 degrees. That increases to 48 degrees at a load factor of 1.5. This analysis was used in the flight pattern analysis described below. A summary of the turning Analysis can be seen in Table 5.2.

Table 5.2

Turning Analysis			
n	ϕ (Rad)	ϕ (deg)	V_{stall} Increase (%)
1	0.000	0	0.0
1.1	0.430	25	4.9
1.2	0.586	34	9.5
1.3	0.693	40	14.0
1.4	0.775	44	18.3
1.5	0.841	48	22.5
1.6	0.896	51	26.5
1.7	0.942	54	30.4
1.8	0.982	56	34.2
1.9	1.017	58	37.8
2	1.047	60	41.4

5.5 Endurance

A power schedule was defined so as to maximize flight endurance with installed battery pack. The analysis was based on the published flight course, the turning analysis performed above and values that are obtained from the program Motocalc. Amperage ratings were used to determine total battery draw to maintain both level flight speed and turning radius. The course has an initial full throttle period of 20 seconds to allow for take-off roll and climb-out, which completes the upwind leg of the flight. Once at a safe altitude, a turn to downwind is performed at 75% power and 1.2 g. Power is kept at 75% to build airspeed for the 360-degree turn scheduled half down this leg of the flight. The downwind 360 degree turn is performed at 85% power and 1.5 g. The final downwind section of the flight will be flown at a maximum of 75% power, with reduced power depending on wind conditions. The final turn or base leg is performed at 25% power for 15 seconds and 1.2 g. The remainder of the flight is completed at minimum required gliding power while on approach for landing. The total power use during a single sortie is 26 Amp-minutes. The full capacity of the 3000 mAh batteries is 108 Amp-minutes. The analysis is performed for a fully loaded aircraft (24 softballs). Twenty-four percent of the battery pack capacity is used in a single fully loaded leg of the sortie. The total time aloft is estimated to be 1 minutes, 15 seconds. The estimated distance is 2700 feet. The course is flown at 30 mph except the takeoff run and the gliding phase to landing. Since six laps or three legs are needed for the entire sortie, the total distance traveled is 8100 feet. Three minutes, forty five seconds are needed to complete the full sortie. The energy used is approximately 78 Amp-minutes. This is 72 % of the battery pack capacity. Hence, the there should be enough battery power to fly the entire mission as well as allowing for any unexpected deviations that might occur during the flight.

5.6 Control Surface Sizing

Control surface sizes were determined using historical values for the rudder, elevator, and aileron sizing. The control surfaces were estimated at 25 % of the chord of the respective surface to which the control was mounted.

5.6.1 Aileron Sizing

The ailerons were sized in conjunction with the historical convention of near 50% semi span and 25% chord. This resulted in 22x5 in. ailerons. This provides a control surface area of 110 sq. in. The ends of the ailerons were kept enclosed and not extended to the tip of the wing so that the tip vortices would not over stress the ailerons or the servos necessary to power them.

5.6.2 Elevator Sizing

Since the tail is a V-tail based on horizontal tail sizing, the control surfaces would be based on the required area for an elevator on the calculated horizontal tail. Using the historical convention that the elevator should be 30-40 percent of the horizontal tail, the elevator for the V-tail was calculated as 40 percent of its area. This resulted in 1.18 sq. ft. Since the most control possible was desired, this was

based upon the upper limit of 40 percent for the elevator. Also since the control surface is to double as both the rudder and elevator, it was necessary to make sure there would be enough area to perform both functions.

5.7 Stability Analysis

Due to lack of necessary numbers for a dynamic analysis, only a static stability analysis was conducted. Since this aircraft is not performing acrobatic stunts and is only flying in a straight and level flight anyway, static stability is all that matters. The velocity of the aircraft used for the stability calculations was the estimated cruise velocity of 50 ft/s. All calculations made were based on equations from Dynamics of Flight by Etkin and Reid and Flight Stability and Automatic Control by Nelson.

5.7.1 Longitudinal Stability

The lift-curve slope of the wing and horizontal tail were determined in 2-D then corrected for the 3-D values. Using C_f/C which is .25 for the Knight Hawk as stated above the values of (C_{ld})theory from the figure was found to be 3.7 per rad for the 3-D corrected case. The value for the 3-D elevator effectiveness was then computed to be .058 per rad.

The aircrafts neutral point and longitudinal stability was computed by ignoring body effects. The calculated neutral point for this aircraft is .88 ft from the leading edge. The longitudinal stability $C_{m\alpha}$ was calculated using the following equation.

$$C_{m\alpha} = (C_{mcg} - C_{mo})/\alpha$$

Equation 5-2

Using C_{mcg} and C_{mo} for the airfoil used, the longitudinal stability ($C_{M\alpha}$) for the wing was found to be 0.0058 and -.0097 for the tail both in per degrees. Thus the total aircraft longitudinal stability is -0.0039. Since $C_{M\alpha}$ for the entire aircraft is negative, we can say that the aircraft will be longitidiuinally stable, however just barely. This confirms the unsteadiness seen in the X-Plane model. During construction, the tail arm will be increased to account for the lack of a static margin.

The pitching moment coefficient was computed as a function of angle of attack for three different elevator deflections –10 deg, 0 deg, and 10 deg respectively. Figure 5.1 displays the result of the function with the three different elevator deflections.

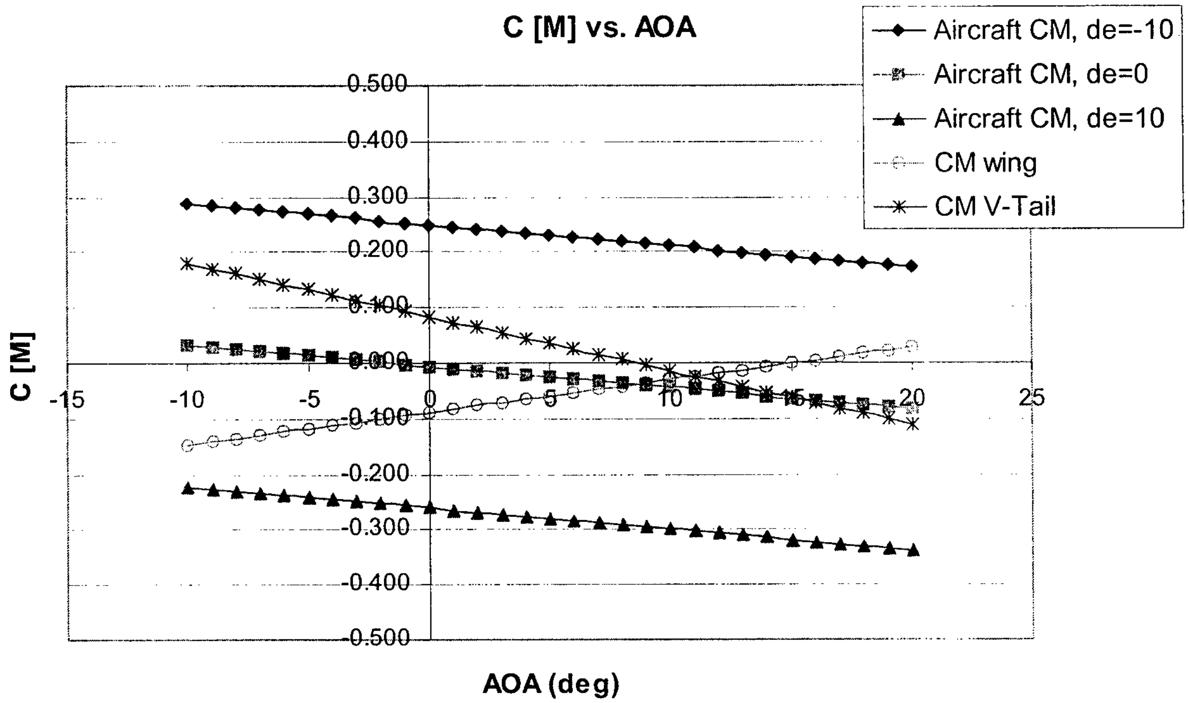


Figure 5.1: Cm vs. AOA

The coefficient of pitching moment decreases with an increase in angle of angle of attack. The coefficient of pitching moment shifts upwards with a positive deflection of the elevator. This trend is conducive to a stable aircraft.

The deflection trim was calculated as a function of angle of attack. This trimmed deflection is when the pitching moment is equal to zero. The trimmed deflection is negatively increasing as the angle of attack increases.

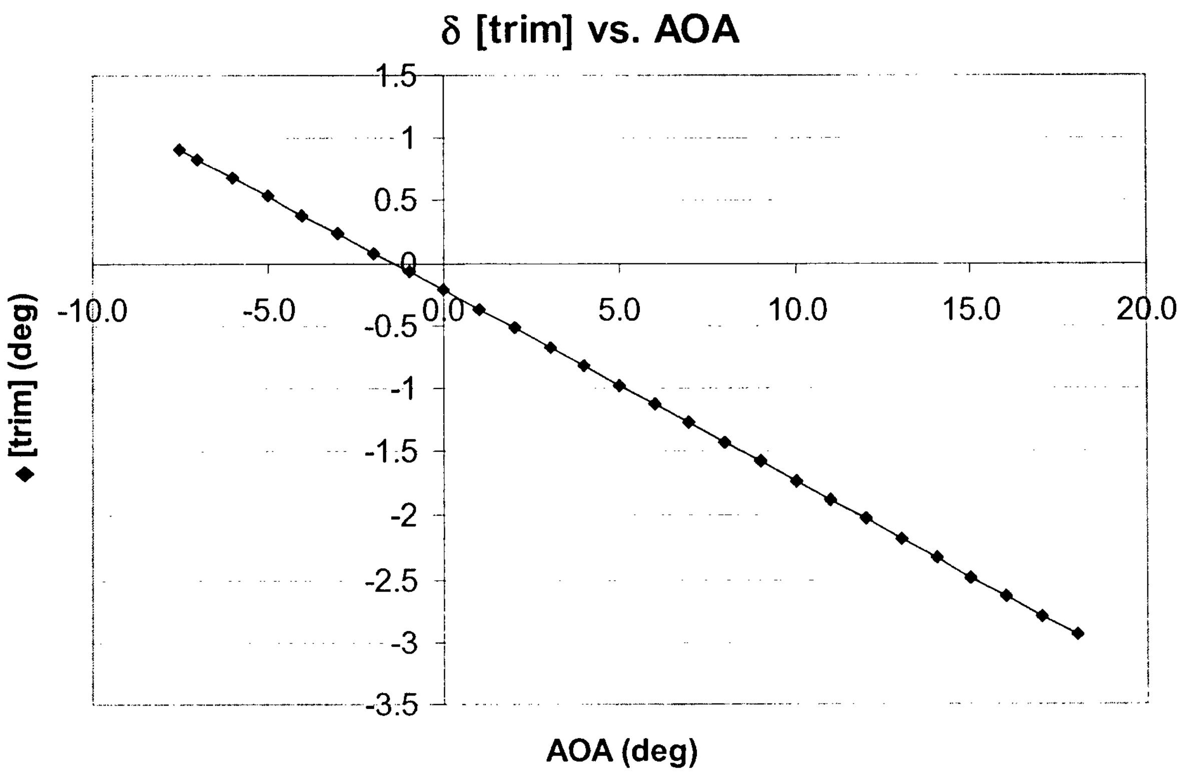


Figure 5.2: dtrim vs. Angle of Attack

The coefficient of lift with respects to the trimmed condition was computed as a function of the angle of attack from 0 to 12 degrees. Where the trimmed deflection for each angle of attack was found in the previous calculation and plotted in Figure 5.2. Figure 5.3 plots the relationship above for angle of attack from 0 to 12 degrees. The coefficient of lift for the trimmed condition increased with an increase in angle of attack. This graph displays the coefficients of lift for the aircraft when there is no pitching moment on the airplane.

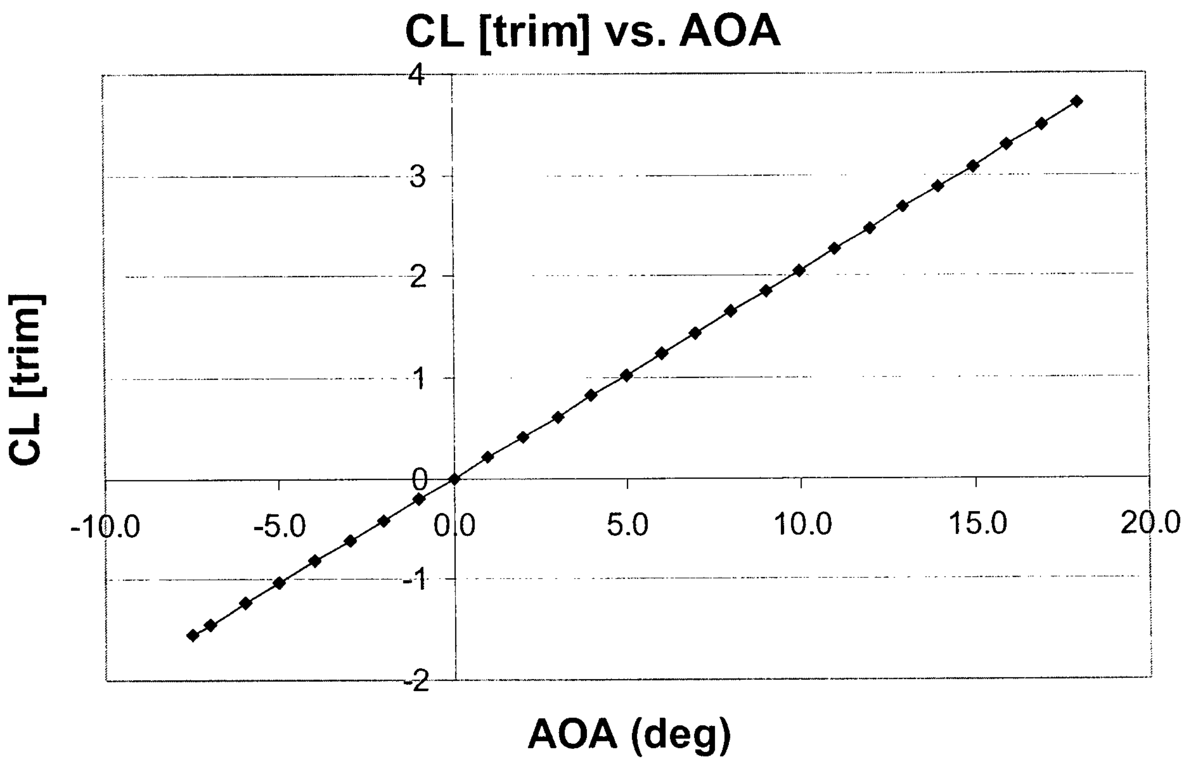


Figure 5.3: Cltrim vs. Angle of Attack

5.7.2 Lateral-Direction Stability

The fuselage-wing contribution to directional stability was computed. The value obtained for C_{nbf} is -.0003 for the fuselage-wing contribution.

The vertical tail contribution was obtained through historical data assumptions obtained form Datacom. However these approximations were complied from historical data of swept wing aircraft with a horizontal tail. $A_{c/4}$ is the quarter chord sweep, which is equal to zero for the Knight Hawk because the wing is rectangular. The vertical tail contribution for the Knight Hawk is .0046. In addition the rudder effectiveness, C_{ndr} was calculated to be -0.018.

The coefficient of the yawing moment was plotted as a function of the sideslip angle β□ and separately as a function of the rudder deflection. These were plotted in Figure 5.4 and 5.5. The range of sideslip angles ranges from –10 deg to 10 deg and the rudder deflections range from –20 deg to 20 deg.

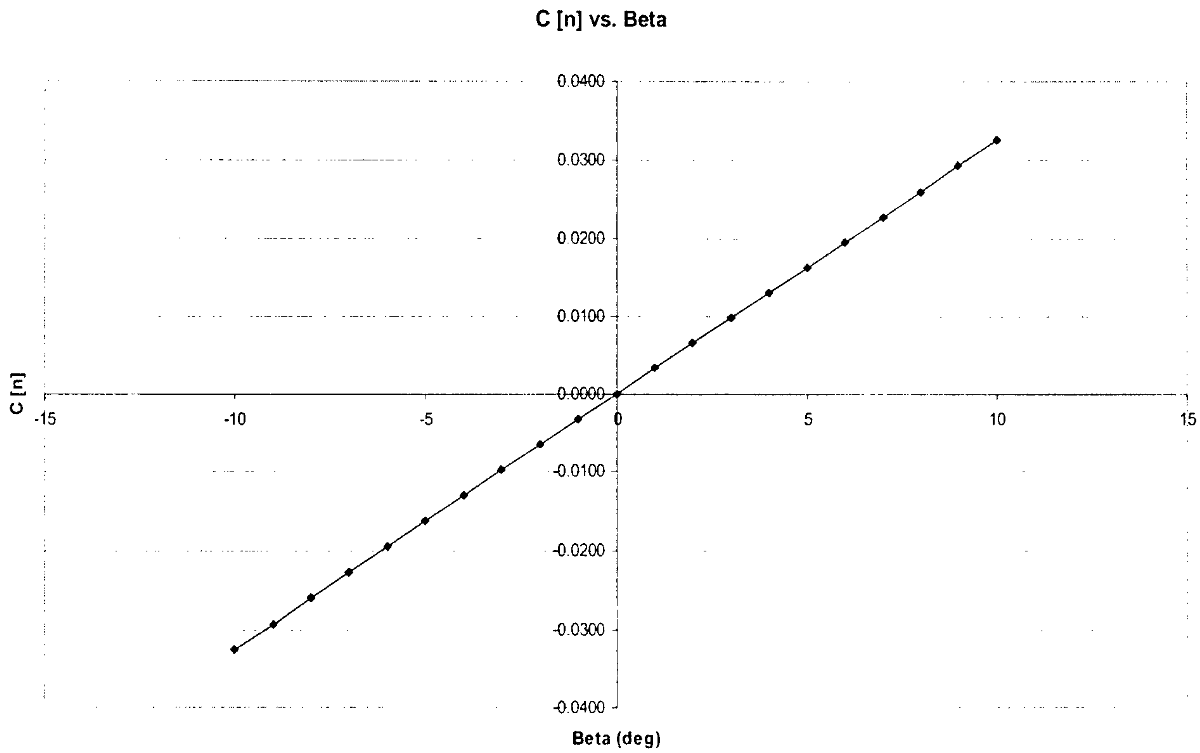


Figure 5.4: Cn vs. B

With the increase in the sideslip angle the yawing moment coefficient increases both in the positive and negative regions. Since the directional stability derivative $Cn\beta$ is shown as positive, we can say that this aircraft is stable. In this case the horizontal tail has a small effect, while the rudder has the most stabilizing effect. The fuselage in this case is destabilizing.

The maximum sideslip angle was determined for a steady sideslipe with a maximum rudder deflection of 20 deg. The maximum sideslip angle for the condition stated above was 7.87 rad.

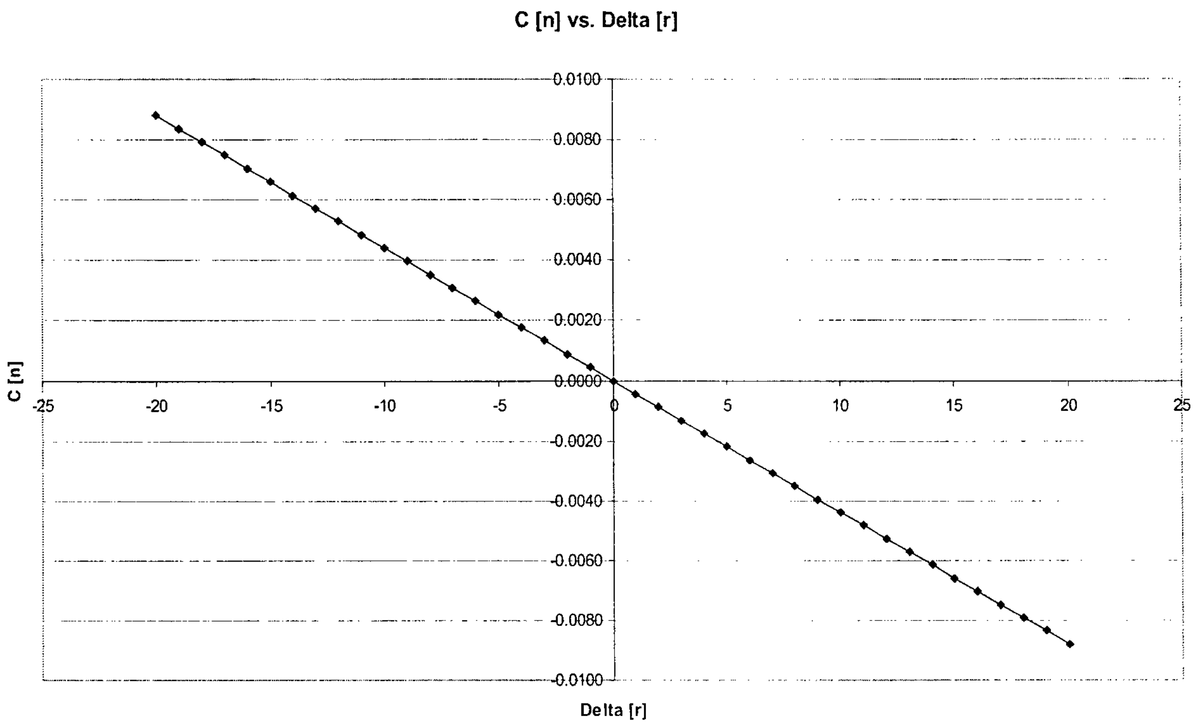


Figure 5.5: Cn vs. dr

5.7.3 Roll Stability

Wing dihedral and sweep contributions to an aircraft are significant in the roll stability of an aircraft however this aircraft does not have wing dihedral or sweep due to other manufacturing and structural constraints. Therefore the wing was made with no dihedral and no sweep. Thus calculations for these effects were not calculated.

The vertical tail contribution to the roll stability and control was calculated using the equation below. C_{lb} for the vertical tail obtained from the equation below is –0.0046.

$$C_{l_{betav}} = -\frac{S_v z_v \eta_v \left(1 + \frac{\sigma}{\beta}\right) C_{L_{\alpha_v}}}{S\, b}$$

Equation 5-3

The aileron effectiveness was computed using strip theory however the equation simplifies because the aircraft has a rectangular wing. The equation yields a value of -0.018 for the aileron effectiveness.

$$C_{l_{\delta a}} = -2 \frac{C_{L_{\alpha_w}} \tau \int c(y)\, y\, dy}{S\, b}$$

The coefficient of the rolling moment was plotted as a function of the sideslip angle and the aileron deflection.

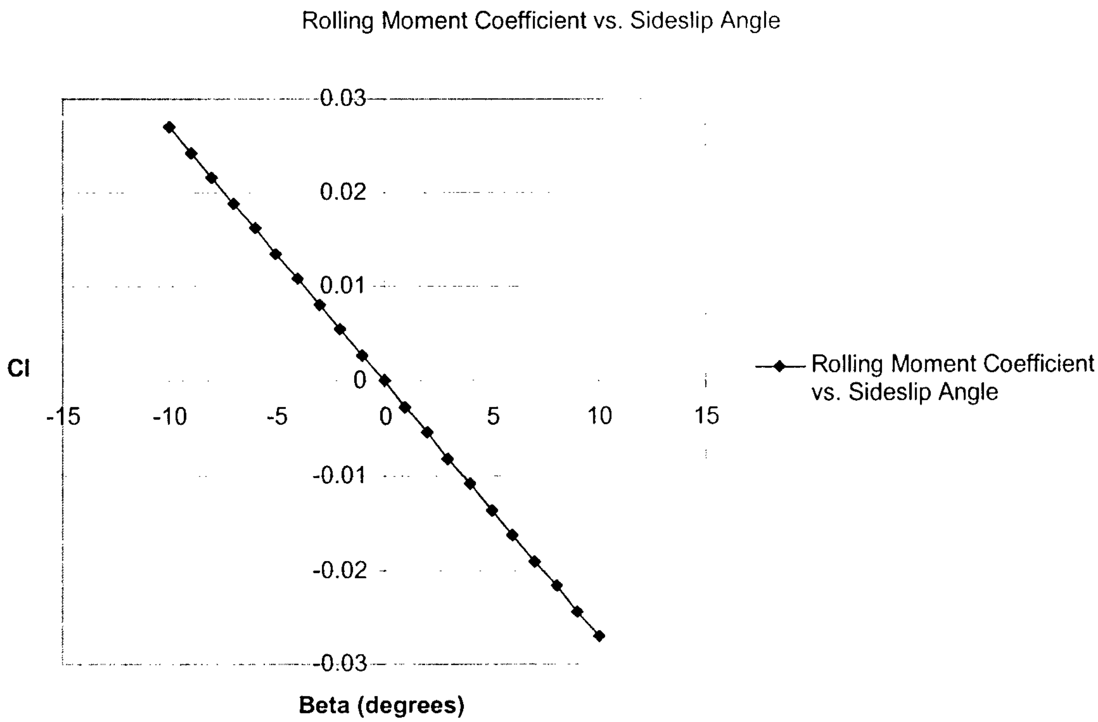


Figure 5.6: Cl vs. B

The rolling moment coefficient was calculated for a range of sideslip angles from –10 deg to 10 deg. Figure 5.6 is the plot of the rolling moment coefficient as a function of the sideslip angle. From the plot the rolling moment coefficient decreases as the sideslip angle increase. Since the stability derivative of $Cl\beta$ is negative, this trend shows a stable aircraft. Figure 5.7 plots the rolling moment coefficient as a function of the aileron deflection from a –20 deg deflection to a 20 deg deflection. As the aileron deflection is increased the rolling moment decreases. C_l is positive for negative aileron deflection and negative for positive aileron deflection.

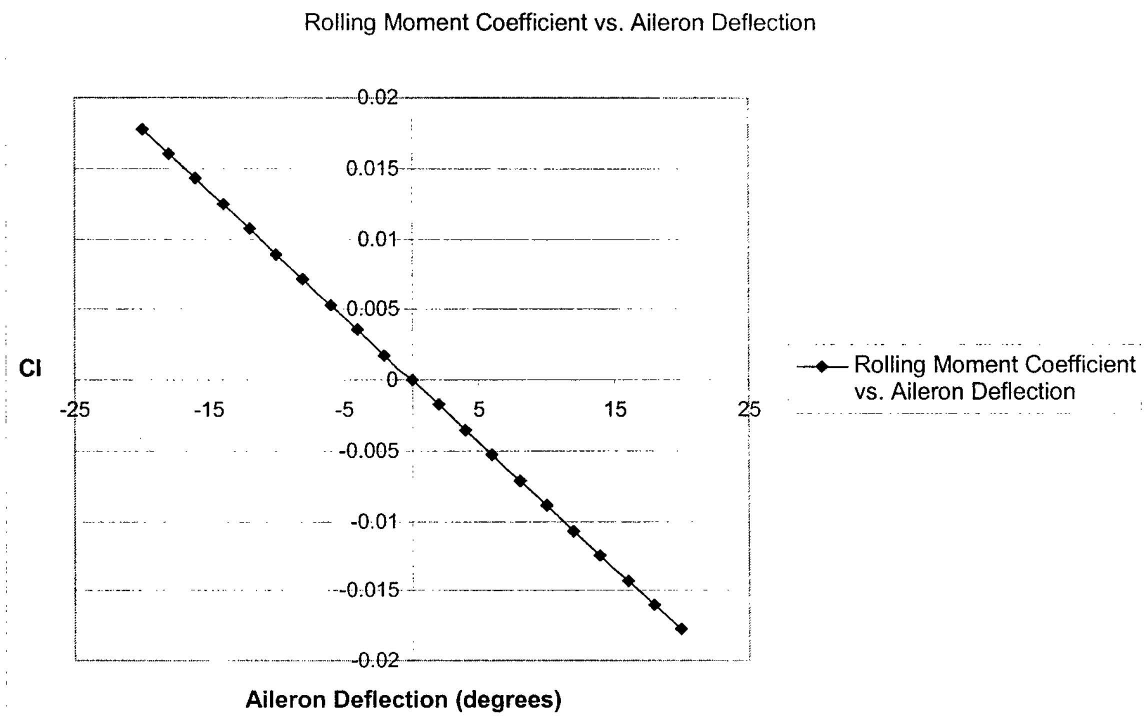


Figure 5.7: Cl vs. da

5.8 Weight Analysis

For the final design phase, estimates for the weight became more detailed as more components were known and better approximations could be made. All approximations were sure to be made over approximations so that there would not be an aircraft built that could not bear its own weight. Below is a final tabulation of the weight estimates.

Table 5.3 Weight Estimation Table

Component	Weight (lbs)	Quantity	Subtotal (lbs)
Softball	0.361	24	8.7
Battery Pack	5.00	1	5.00
Receiver	0.166	1	0.166
Servos	0.166	5	0.833
Landing Gear	1.00	1	1.00
Wheels	0.250	3	0.750
Wing	3.75	1	3.75
Tail	1.00	1	1.00
Fuselage	4.50	1	4.50
Motor	2.50	1	2.50
Speed Controller	0.075	1	0.075
Gear Box	0.100	1	0.100
Receiver Battery	0.183	1	0.183
Hardware	0.500	1	0.500
Prop	0.150	1	0.150
Gross TO Wt.	29.21		
Operational Empty Wt.	20.51		
Airframe Wt.	15.51		

5.9 Component Selection

5.9.1 Radio Gear

A Futaba 6XAS 6-channel radio controls the airplane. Each control surface has its own servo which is programmed through the radio mixing capabilities. This allows for the V-tail to be used as well as the possibility of flaps. Also controlled through the radio is the motor speed with help from the motor speed controller. The receiver is powered by a 600 mAh battery which also provides power to the servos. This pack should last for several missions, but will be changed before each run.

One Astroflight 90 geared DC model aircraft motors provides propulsion for the aircraft. The motor has an Astroflight speed controller rated at a maximum current draw of 60amps. The motors are connected in series to a series-string of 38 Sanyo 3000 mAh RC type cells. Each cell is a Sub-C size. Total measured battery pack weight is 4.95 lbs. The batteries are in the cargo area at the bottom of the fuselage. This allows us to move the batteries around so as to obtain a precise predetermined Center of Gravity. Possible propellers range from 20 X 8 to a 26 X 9. Final propeller selection will be determined from static thrust testing. Analysis shows a 20 X 8 propeller is optimal.

5.9.2 Servo Selection

Heavy duty servos are used for all control surfaces. Each servo produces 44 ounce-inches of torque. Flight-testing will determine if stronger servos are necessary for some or all control surfaces. An R148DP PCM receiver that provides failsafe features on all channels is used. Four channels of the six

available are used. Ailerons, elevators, and rudders provide control. The rudder and elevators are both run by one servo each and ailerons are controlled by one servo per surface as well. Nose-wheel steering is controlled by another servo, but is attached to the same channel as the rudder servo.

5.10 Landing Gear Analysis

A solid spring landing gear was chosen for the aircraft due to its simplicity of design. This landing gear was also analyzed using MARC to ensure that it could withstand the weight of the aircraft sitting on the ground and also during landing. A preliminary analysis showed that the height of the main gear needed to be 13 inches in order to allow adequate clearance for the propeller on rotation. The gear was then designed to extend 12 inches down from the fuselage (assuming 2" diameter tires) and 9" out. As with the spar, both steel and aluminum were analyzed.

Using MARC, it was found that the maximum displacement occurred at the bottom of the landing gear, with the gear being pushed out and up. The deflections are as follows (positive x is away from the aircraft centerline, positive y is up)

Table 5.4

Material	Aluminum	Carbon Steel
x-deflection	0.59"	0.21"
y-deflection	0.55"	0.19"

Since during rotation, the weight will be transferred from the gear to the wings, it was determined that the aluminum would provide enough strength while saving weight over using steel. In order to ensure that the landing gear can handle a "bad" landing, a transient analysis of the landing gear was performed. The load was quickly increased and unloaded to simulate an initial bounce. This was then followed by a second, softer landing a short time later. This simulated a hard original touch down with the aircraft bouncing back up into the air and then settling back down on the runway. The landing weight of the aircraft is still 30lbs. As the yellow (y deflection) and red (x deflection) lines show in the figure below, the gear is first pushed out (negative x direction) and up (positive y). It then continues out for a short time even after the gear has been unloaded, when it then springs back down and right of its neutral (unloaded) position. When it loads again, we see that it again deflects up and out. The green line, representing the total displacement, shows the gear returning to it's steady state position after the aircraft finally settles on the ground. Since during landing the aircraft will most likely not see as high an angle of attack as it does during rotation, clearance was not a major concern. As it turned out, the maximum compression of the gear was only about 0.14" more than the static case.

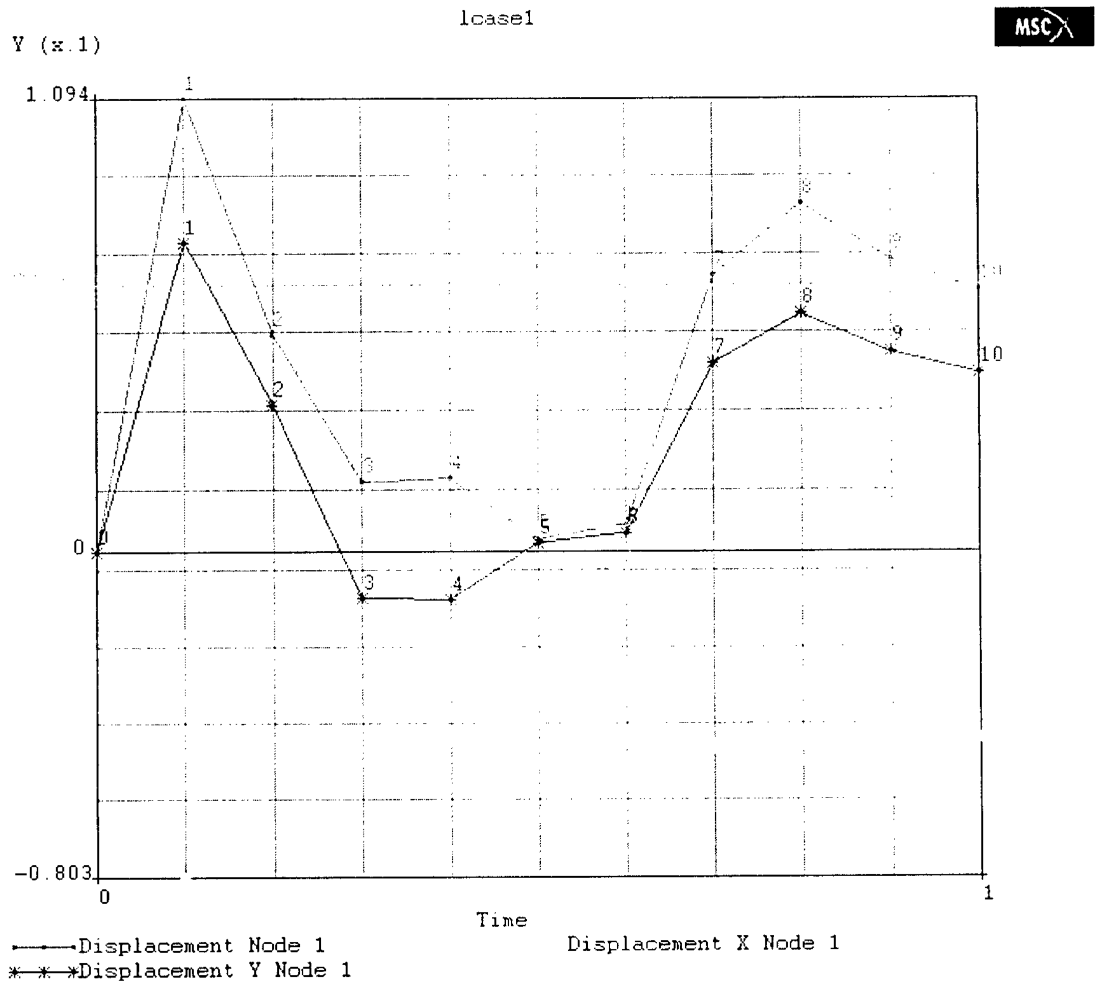


Figure 5.8: Landing Gear Displacement

5.11 Final Aircraft Configuration and RAC

The final aircraft configuration and systems layout are shown in the attached drawing package. The final aircraft configuration is a conventional design with a straight wing mounted towards the front of the fuselage, a motor in the nose with a puller propeller, and a V-tail in the rear. The fuselage consists of a 18 inch wide flat panel, thick enough to enclose the softballs, batteries and engine. The landing gear consists of a main gear mounted just behind the center of gravity, and a standard steer-able nose wheel underneath the nose. The main gear is placed far enough behind the center of gravity so the aircraft does not tip, but close enough to it to minimize rotation problems. The aircraft has a wingspan of 9 feet, and an overall length of 6 feet. Estimated aircraft empty weight is 19.54 lbs. Estimated take-off weight is 35.25 lbs. (fully loaded)

The Rated Aircraft Cost for the final configuration was now updated from the preliminary design. This cost now included a better estimate of the weight as well as updated numbers for sizes surfaces.

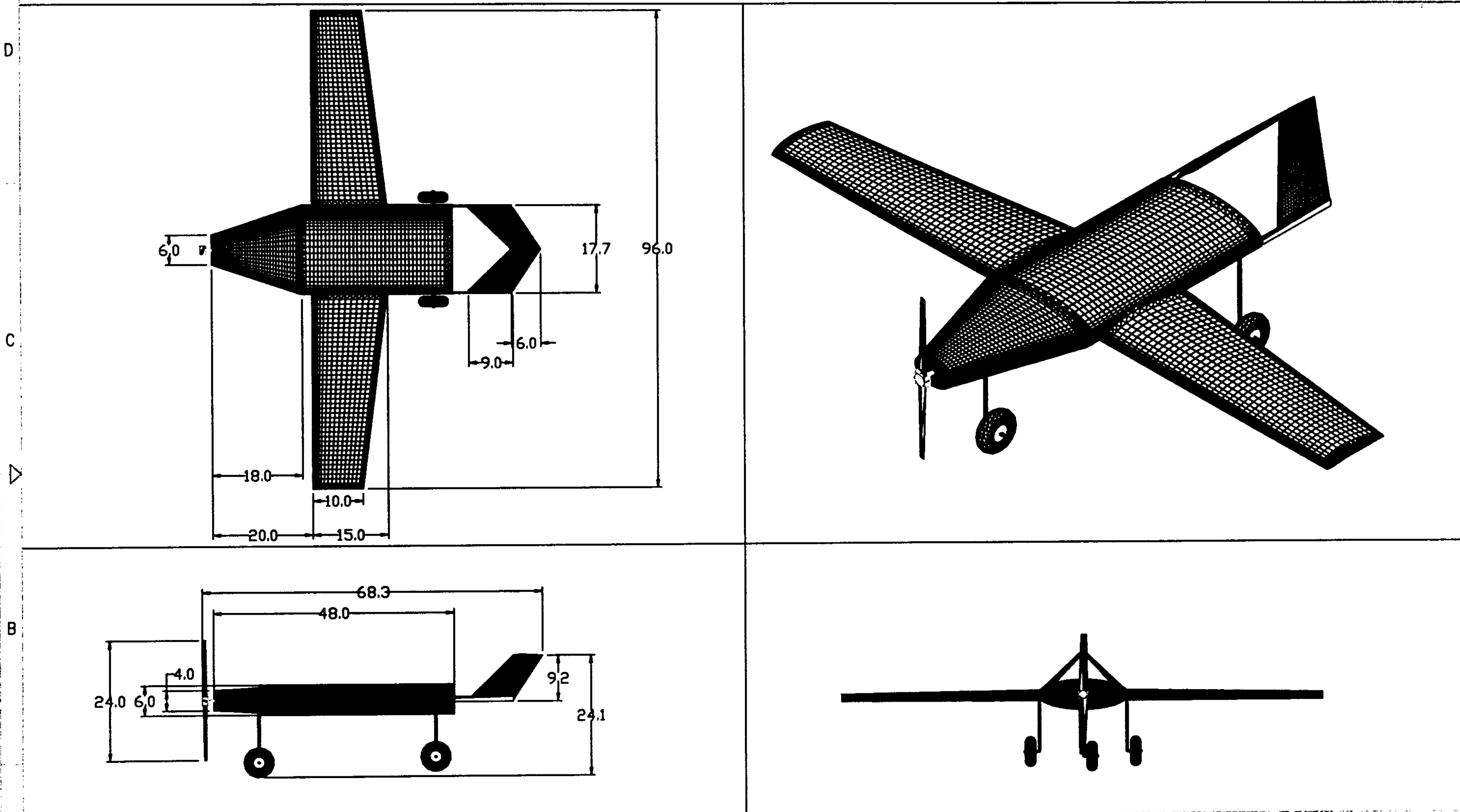
Table 5.5

Description	Cost	V-tail Amt.	Val.
Wing	8hours/ft.	9 ft.	72 hours
Chord	8hours/ft.	1.5 ft	12 hours
Cont. Surf.	3hours/surf.	2 surf	6 hours
Fuselage	10hours/ft.	6 ft.	60 hours
Hor. Fins	5hours/fin	0	0
Vert. Fins	5hours/fin	0	0
V-tail	15hours/v-tail	1	15 hours
Servos	5hours/servo	5	25 hours
Engines	5hours/eng.	1	5 hours
Propellers	5hours/prop	1	5 hours
Total Hours	$20/hr		200 hour
MEW	$100/lb		20.5 lb
REP	$1500		5
Total Cost			13.55

As a result of the updated numbers and more refined weight estimate, it was found that our aircraft cost has decreased considerably by almost $400 dollars.

5.12 Drawing Package

On the following pages is the drawing package. The first plot shows a three view of the aircraft with an isometric view. This plot is dimensioned. The remainder of the images show the four views in a more detailed sense and without dimensions to give a clean view of the aircraft.



REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED

Notes:
1. Dimensions are in Inches.

Aircraft Assembly 2002-100

PARTS LIST

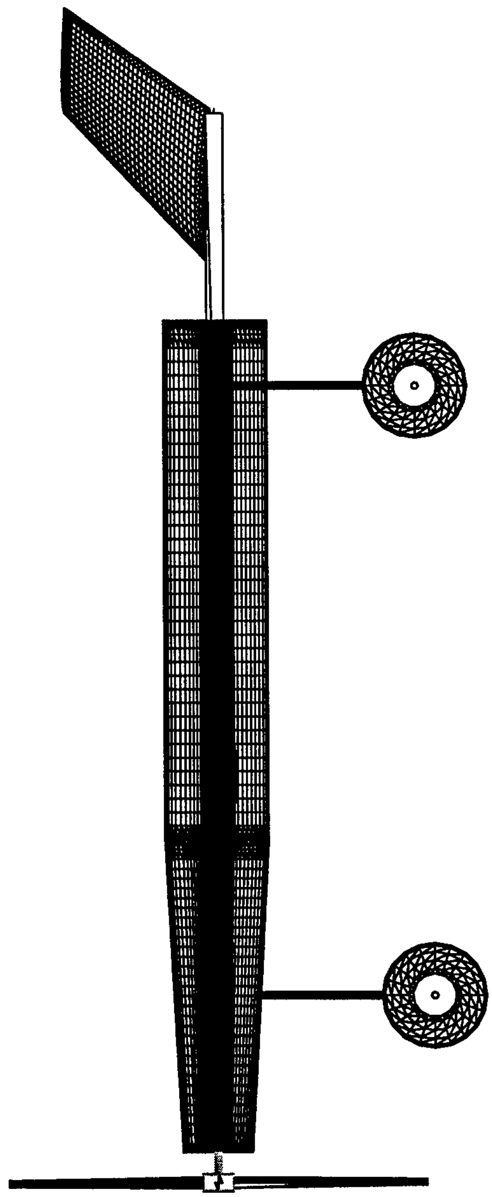
Clarkson University

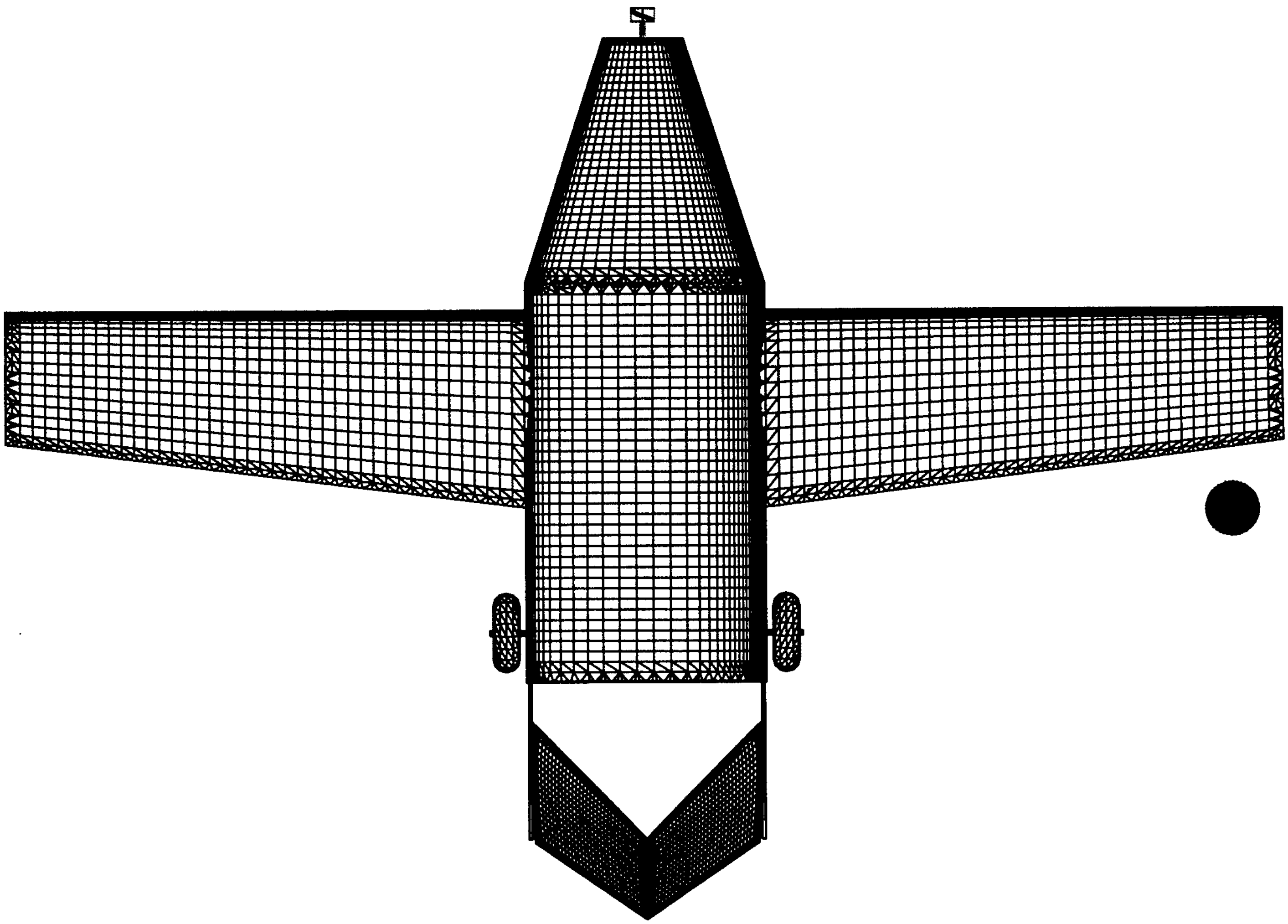
DBF
KnightHawk 2
ASSEMBLY

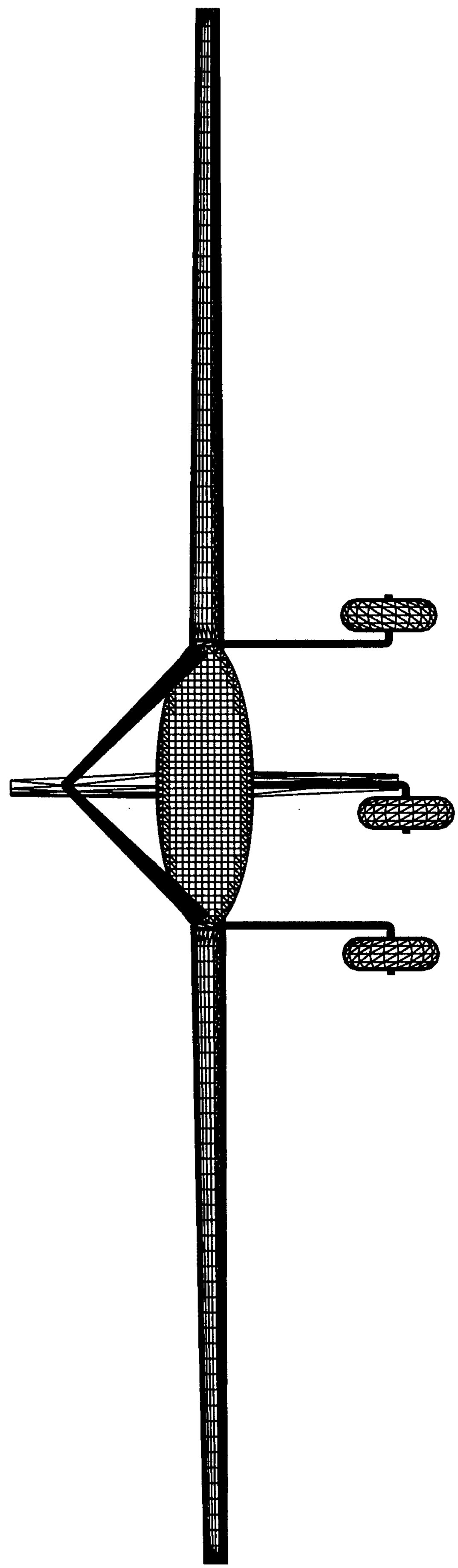
A 00101 2002-100 B

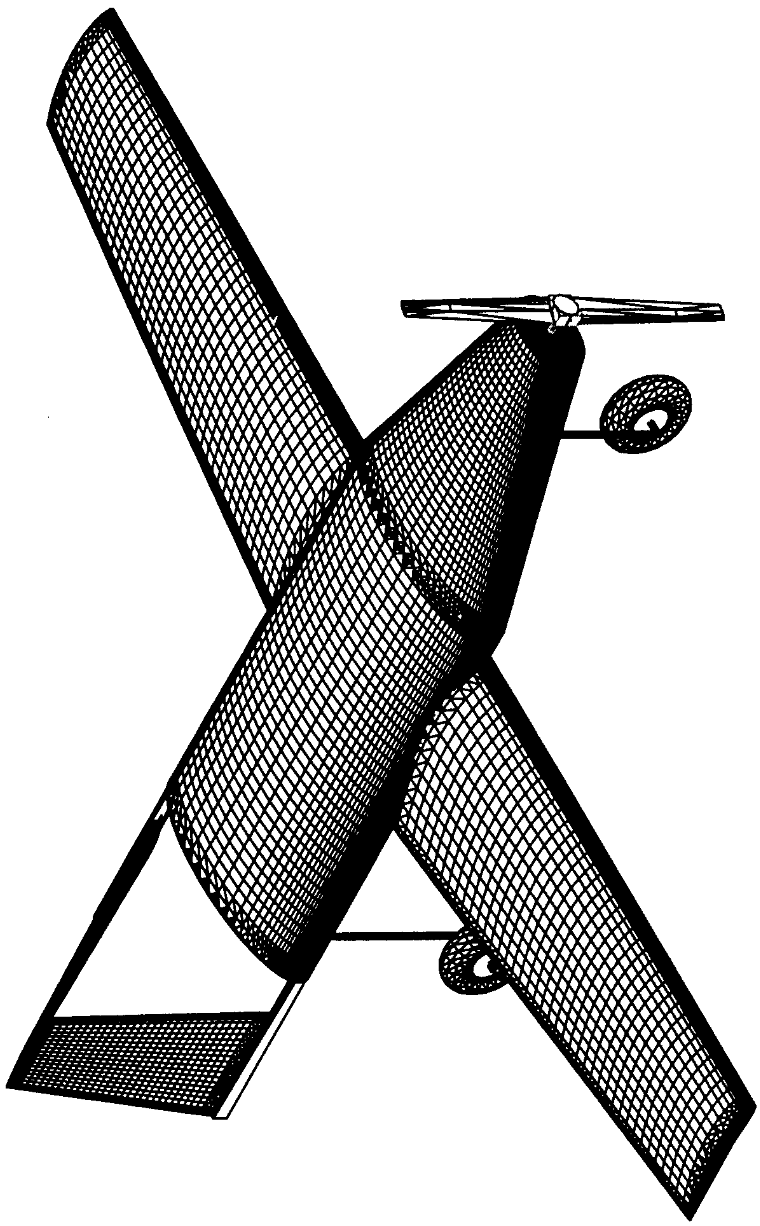
NEXT ASSY. USED ON
APPLICATION

7220-000-600









6. Manufacturing Plan

Once the detailed design had been completed, the manufacturing processes for each individual component of the aircraft were investigated. The aircraft was divided into three components of importance: the main wing, the empennage, and the fuselage. Various building materials for each component were explored. Figures of Merit that each technique would be evaluated by would include; strength, time of construction, skill, and lightweight.

6.1 Main Wing Construction

Three specific building techniques were evaluated for the main wing. These included a built-up construction, a solid foam construction and a composite method. Each approach was evaluated based on the desired qualities they would give; ability to build, lightweight, and also strong enough to endure the various loads experienced during flight.

6.1.1 Built-up Technique

In the case of a built-up wing construction, ribs would be made in the shape of the desired airfoil out of balsa wood, and then joined together in an array via a main and secondary spar. These spars would be located at the quarter chord and the trailing edge of the wing respectively. Thin sheets of balsa are laid along the first ten to twenty percent of the airfoil to ensure a smooth surface for the flow to first encounter. When the initial "skeleton" of the wing is completed, monokote is applied to the exterior to form the skin of the wing. This form of construction, if done properly, can provide a significantly light as well as strong wing. The one disadvantage is that replicating the numerous ribs and allowing for drying of the glue can become time consuming.

6.1.2 Solid Foam Technique

By using a hot wire cutting system similar to what was used with the 2001 Clarkson entry, a wing could be constructed out of simple housing foam. By placing airfoil templates at the ends of a piece of foam and tracing around these templates with a hot wire, the entire wing can be cut out in one pass. This results in a very strong wing using an appropriate spar. On small scale applications like with the canard from last years entry, this proved an excellent technique. However for larger scale applications such as a nine foot wing, the weight can escalate quite quickly.

6.1.3 Composite Technique

Composite construction undoubtedly provides the most efficient construction technique. This method provides an excellent weight to strength ratio as well as an extremely well finished surface for airflow to follow. However the materials and the equipment for this process are not easily available. The molds necessary for the shape you wish to make can become almost as time consuming as a built up structure.

6.2 Empennage Construction

Two construction processes were examined for the empennage. These were again a built-up construction and a foam construction. A composite construction was not considered because of the size of the empennage, where the weight savings would not be marginally better than the other techniques. Also, as stated before, the expense to use this technique would not be justified with the other means available.

6.2.1 Built-up Construction

A built-up construction of the empennage would be achieved by creating a shape of the surface with balsa strips, and then covering both sides of the outline with aircraft plywood. This shape could be covered with monokote to give the skin, however this would not be structurally sufficient and so aircraft ply would be used. Also control surfaces on the empennage would not be able to be supported.

6.2.2 Foam Construction

Construction of the empennage out of foam would be similar to the way the wing would be constructed. A template shape would be made from which the empennage could be cut out of the foam using the hot wire system. The problem with this construction is that the empennage tends to become very thin. The foam would then need to be reinforced to be able to remain rigid under normal flight conditions as well as support control surfaces.

6.3 Fuselage Construction

Realizing the need to seriously investigate different fuselage construction techniques after last years entry, three new methods were inspected. These included a composite construction, a fiberglass built-up construction and a balsa built-up construction.

6.3.1 Composite Construction

Creating the entire fuselage out of composites requires extensive materials and ability, but the results can be well worth the effort. An extremely light and strong structure is very beneficial to the goal at hand. For this method, a mold of either the entire fuselage or parts of it are made up over which the composite structure is laid. This structure then needs to be baked at a high temperature for a time until the resin cures. It has been estimated that this could take up to ten hours per part. There is also the problem then of joining the separate parts, as well as how to repair any damage that may happen in a crash.

6.3.2 Fiberglass Construction

In this method, a convex mold of the fuselage would be made over which the fiberglass would lay.

Once the part is cured, several wooden bulkheads would be placed along the inside of the shell to ensure structural stability. Two booms are then made from wood to give torsional strength in the fuselage as well as a moment arm for the empennage. This technique provides a compromise between the composite and built-up structures, giving good strength and decent weight properties.

6.3.3 Built-up Construction

In a built-up construction, the entire fuselage would be made of wood, using the conventional stringers and panels method used in many full size airplanes. The resulting structure would then be covered in monokote to provide a skin. This structure can be light if done right, but it also can become very fragile.

6.4 Construction Techniques Selected

6.4.1 Main Wing Construction

The wing construction technique that was chosen was the built-up structure. The composite technique would have been the unanimous choice of the team, however, the cost and inability to acquire the necessary equipment made this choice impossible. Learning lessons from last years entry, the foam wing, while very strong and quick to build, still was too heavy for what the design weight of this years aircraft. The built-up wing provided a strong and light wing that would serve the purpose needed. Also, there are members on the team with experience building this sort of wing, therefore, the time that it would take to build this structure would not be as long as if it were done by inexperienced people.

6.4.2 Empennage Construction

For the empennage construction, the built-up construction was chosen due to the strength it would offer over the foam construction. It was decided that any additional structural parts to the foam would negate the weight savings that it would provide. The time that it would take to build each was nearly the same as well was the skill necessary to build both.

6.4.3 Fuselage Construction

The technique chosen for the fuselage construction was the fiberglass structure. Though both the fiberglass and built-up structure offered a comparable weight, a built-up structure was found to be too fragile in a crash, where the fiberglass would be able to withstand more damage. The composite fuselage was thrown out due to the impracticality of the cost when taking into consideration our budget.

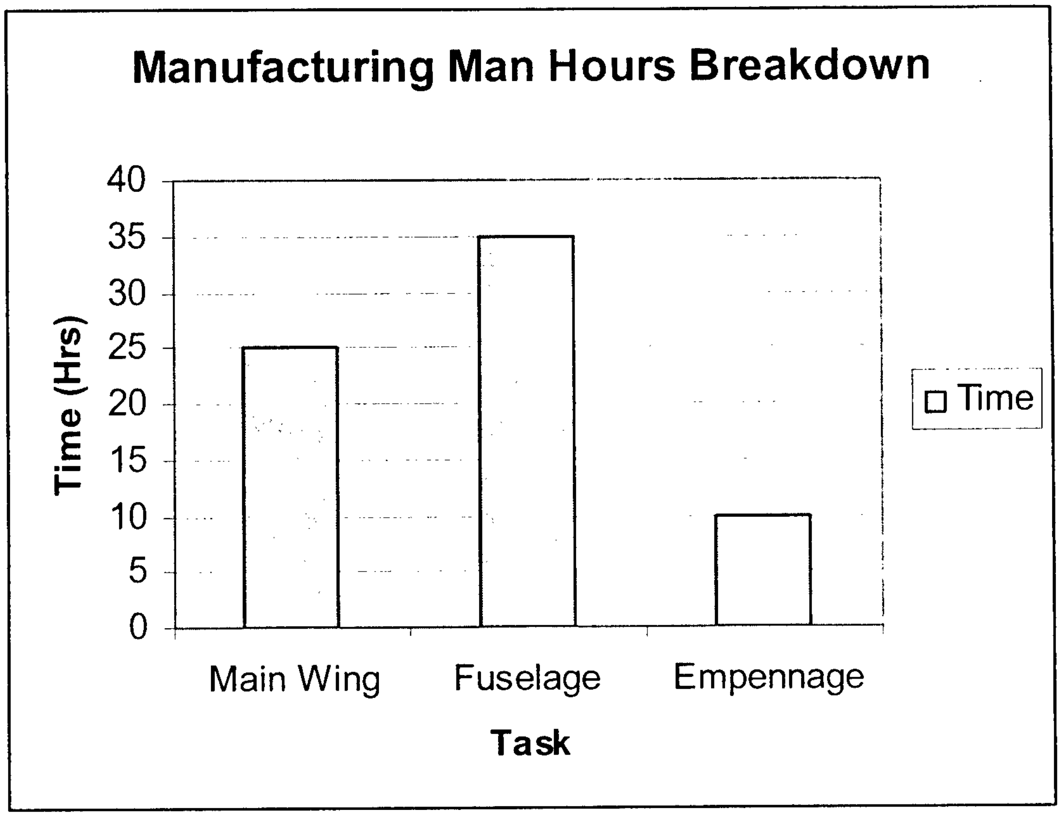


Figure 6.1: Manufacturing Man Hours Breakdown

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THE CITY COLLEGE OF NEW YORK

City University Of New York

New York, New York

F A L C O N

CESSNA ONR, DESIGN, BUILD & FLY COMPETITION

AMERICAN INSTITUTE OF AERONAUTICS AND ASTRONAUTICS

TABLE OF CONTENTS

EXECUTIVE SUMMARY

1. Development of Design:

From the beginning it was determined that the airplane would be a monoplane. All the other airplane configurations have more involved design and manufacturing requirements. The cross section of the fuselage was initially designed to be box-like. To improve the overall design, after aerodynamic considerations, as well as to take the challenge, the fuselage cross section was changed to be circular. After the required dimensions were determined, the cross section took the final shape of an ellipse.

A further development in the design was the choice of high wing or low wing. Last year the airplane built had a high wing configuration, and high wing airplanes are inherently more stable than low wing airplanes. Due to the experience carried over from last year as well as the stability and control advantages, a high wing was initially selected. After further deliberation, it was determined that a high wing airplane may have payload accessibility problems. This consideration was determined to be more important than the advantages of a high wing airplane. As a result the low wing configuration was selected over the high wing configuration.

For manufacturing purposes, the wingspan was limited at 10 feet. After further consideration of the bending moments a wingspan of 10 feet might place at the root chord of the wing, and also the general strength of the wing, the wingspan was changed to eight feet. This alteration of the wingspan had very important consequences for the airplane.

The limit of 8 feet placed on the wingspan, determined that twenty-four balls could not be carried. Instead twelve balls would have to be the maximum number of balls carried. A wingspan of eight feet also determined the airfoil to have high lift characteristics. The airfoil eventually selected, analytically proves to be a good match to eight feet of wingspan, and the airplane.

2. Development Highlights:

The determination of the lift coefficient desired at cruise was a highlight in the development of the airplane. The estimated lift coefficient determined using the lift force equation was 0.7. The NACA 6412 airfoil was selected due to its high-lift characteristics among other things. After correcting the lift coefficients for the airfoil to a wing of aspect ratio, it was determined that the lift coefficient the wing would give us at two degrees angle of attack is a little more than 0.7. What made this further exciting is that at two degrees angle of attack, is the angle for minimum profile drag. This was indeed a highlight in the development of the wing. The airfoil selected, the aspect ratio chosen, among other factors had resulted in a wing that would meet all the requirements.

3. Design Alternatives:

The design alternatives investigated range from aircraft configurations to individual components such as airfoils. In the conceptual design stage the alternative airplane configurations investigated include flying wing, canard, and bi-plane configurations. A reason these

configurations where rejected is lack of experience working with them. Alternative configurations investigated for airfoils, where those with less camber, and less thickness ratio. This is because camber and thickness ratios affect the capability of each airfoil. For wing plan form shapes, elliptical as well as tapered plan forms where considered. Elliptical plan forms are the most efficient, but like tapered plan forms they are difficult to construct.

4. Overview of Design Tools:

The design tools used in the conceptual design where books and software that are operable on the Internet. The books used in the conceptual design, where not very technical. The information contained in them where mainly descriptive. These books where useful in the conceptual design, because conceptual design is not technical in nature. The websites used in the conceptual design report where interactive. One of them plotted the points for the NACA airfoils, and another evaluated the airfoils. The tools used in the conceptual design where generic and mundane in nature.

For the preliminary and detail design report the tools used where more complex than those used in the conceptual design. "Fundamentals of Flight" written by Richard Shevell, was very useful in the preliminary design. All equations used for the aerodynamic design as well as part of the performance prediction where mainly obtained from this text. The remaining texts used served as supporting material. The online text from James F. Marchman of Virginia Tech was found very useful in determining and explaining the parabolic drag curve and corresponding power.

The variety of the tools used for all the design phases are indicative of the dedication and creativity of the team. Courses related to flight mechanics, airplane performance, and in general Aerospace Engineering courses geared towards aircraft design are not offered at CCNY. Only upper classmen have the opportunity to study aerodynamics in class. The knowledge necessary to actively participate in the competition this year has been acquired through self-study. Thus, the tools used for the design phases may seem creative and prolific.

MANAGEMENT SUMMARY.

1. Architecture of the Design Team:

The team this year was split into two separate groups, the design group and the manufacturing group. Two team leaders where selected, one for the design group and another for the manufacturing group. The design group leader is responsible for meeting the deadline for the report, and the manufacturing group leader is responsible for guaranteeing the airplane designed is built and ready to fly before the competition date. Figuratively speaking, the design group puts the airplane on paper, and the manufacturing group makes the designed plane tangible.

2. Personnel List and Assignment Area:

Table 1: Design group, names and assignment areas.

NAME	***ASSIGNMENT***
Enomen J. Okogun ME / So	Design Group leader, Aerodynamic Design, Performance Prediction, Design Report Editor
Sebastian Peter ME / Sr	Pro-Engineer CAD Drawings, Fuselage Design
Leonard Lluxama ME / Sr	Propulsion: Motor, Battery and Propeller Selection
Joo Silvia Cho ME / Jr	Performance Prediction
Sidy Ndao Me / Fr	Wing Planform, Preliminary Design Editor
Nancy ME / Soph	Administrative Tasks
Kirk Phillips ME / Sr	Stability and Control
Peter Chiu ME / Fr	Rated Aircraft Cost
Kenneth Wei ME / Jr	Stability and Control
Dean Sawyers ME / Fr	Conceptual Design Editor
Luca Chiarelli ME / So	Landing Gear Design

Table 2: Manufacturing group.

NAME	***ASSIGNMENT***
Cesar Gomez ME / Jr	Manufacturing Group Leader, Manufacturing Plan Co-Editor
Natalia Saldarriaga /ME/JR	Manufacturing Plan Co-Editor
Julian Canizales ME/SR	Group Member
Jason Morales ME/So	Group Member
Kirk Phillips ME / Sr	Group Member

Peter Chiu ME / Fr	Group Member

Prof. Ali Sadegh	Project Advisor, and Fundraising

3. **Management Structures**:

After the team was sub-divided into two groups, design and manufacturing, each group leader became responsible for making and meeting deadlines, and delegation of tasks. There was no elaborate process adopted for configuration control. From the outset of the project, wingspan and chord length were restricted. All other configuration alternatives were left open to the discretion of the group leaders. The both groups met separately on a weekly basis. Before each meeting an agenda was sent via email through the team "Yahoo groups" email address. Also, after each meeting the highlights of the meeting, and tasks that needed to be achieved by the following week were summarized in another email. At the design group meetings, alternative design consideration were discussed and decided upon mainly by common consensus. At times when agreement could not be reach, the group leader made the decisions. The milestones and schedule control for each group were the responsibility of the group leaders.

4. **Milestone Chart**:

Although active design and manufacturing did not begin until January, before this team members had been getting together to read about airplane design and flight mechanics. The main text used during the research phase was "Fundamentals of Flight" by Richard Shevell. The website for an airplane design course offered at Virginia Tech was very useful at this stage in outlining the role and tasks required to manage and achieve the project.

Week / Month	**Planned**	**Actual Timing**
Wk 1 / September	Virginia Tech, AOE 4065-4066 Design (Aircraft) Website	Virginia Tech, AOE 4065-4066 Design (Aircraft) Website
Wk 2 / Sept.	Virginia Tech, AOE 4065-4066 Design (Aircraft) Website	Virginia Tech, AOE 4065-4066 Design (Aircraft) Website
Wk 3 / Sept.	Ch 2, Shevell	Ch 2, Shevell
Wk 4 Sept.	Ch 3, Shevell	Ch 3, Shevell
Wk 1 / October	Ch 4, Shevell	Ch 4, Shevell
Wk 2 / Oct.	Ch. 5, Shevell	Ch. 5, Shevell
Wk 3 / Oct.	Ch. 6, Shevell	Ch. 6, Shevell
Wk 4 / Oct.	Ch. 8, Shevell	Ch. 8, Shevell
Wk 1 / November	Ch. 9, Shevell	Ch. 9, Shevell
Wk 2 / Nov.	Ch. 10, Shevell	Ch. 10, Shevell

Wk 3 / Nov.	Ch. 11, Shevell	Ch. 11, Shevell
Wk 4 / Nov.	Ch. 13, Shevell	Ch. 13, Shevell
Wk 1 / December	Ch. 16, Shevell	Exam Preparation, Research postponed
Wk 2 / Dec.	Wind Tunnel Test Proposal	Exams. Proposal Postponed till March
Wk 3 / Dec.	Wind Tunnel Test Preparation	Final Projects. Test Preparation Postponed till March
Wk 4 / Dec.	Christmas	Christmas
Wk 1 / January	Wind Tunnel Tests	Final Projects Completed. Tests Postponed till April
Wk 2 / Jan.	Team Selection	Team Convened and Group Selected
Wk 3 / Jan.	Airfoil Selection	Airfoil Design and Selection
Wk 4 / Jan.	Wings Fabricated. Conceptual Design Completed	Drag Prediction and Preliminary Design Commences
Wk 1 / February	Preliminary Design Commences	Conceptual Design Completed. Performance Prediction
Wk 2 / Feb.	Preliminary Design	Conceptual Design Report commenced. More Performance Prediction
Wk 3 / Feb.	Detail Design Commences	Fuselage Design. Preliminary Design Report Commenced.
Wk 4 / Feb.	All Design Completed And Finish Report.	WRAM static model show. Detail Design Report Commenced
Wk 1 / March	Send Finished Report	Report Finished and Sent.
Month of March	Airplane Fabrication and Completion	
Wk 1 – Wk 3 / April	Systems Integration, Flight Tests, and Design Alterations	
Wk 4 April	Competition	

CONCEPTUAL DESIGN REPORT

PART 1: ALTERNATIVE CONFIGURATION CONCEPTS

1. MONOPLANE VS. BIPLANE:

Figures of merit: drag, simplicity of design, ease of manufacture

The advantages of a biplane over a monoplane in the competition this year are few. Span, is no longer a limit this year. As a result, the wings for a monoplane can span however necessary to carry the weight of the aircraft. The main advantage of a biplane is that it gives a greater selection for the airfoil section. This is because any airfoil, from a low lifting airfoil, to a high lift airfoil, could be chosen.

A biplane has the ability to provide a larger wing area, less wing loading, and improved structural strength. The relationship between weights, wing loading, and power loading of a plane is the crucial in deciding between a biplane and monoplane in aircraft design. If a plane's horsepower is increased, its wing area can then be reduced. Consequently, the wing loading can also be increased.

Similar to some of the bombers designed early last century, we can assume that our aircraft may comprise of a wing loading less than 10 pounds per square feet, and that its power loading, may lie between 16 and 20 pounds per horsepower. A monoplane can achieve such characteristics with a reasonable wingspan and area. For instance, a thirty-pound plane, with a wing area of eight square feet, has a wing loading of approximately four pounds per square feet. And a fifty-pound airplane, with a wing area of eleven square feet, has a wing loading of less than five pounds per square feet. Likewise, a monoplane can achieve these wing loadings relatively easily.

On the contrary, the drag generated by a biplane, can very easily be greater than the drag generated by a monoplane. External bracing, and a higher exposed area cause this greater drag on the biplane. Since a monoplane has a higher span, it would also have a lower drag. Therefore, a biplane with its exposed surfaces that generate parasite drag, is less efficient than a monoplane.

Among the influential factors of our decision between a monoplane and a biplane, was the limited expertise of our members. So prediction of the plane's performance was imperative. We therefore had to select the plane that had ease of manufacturing and designing. After analyzing the advantages and disadvantages of both planes, including the time required for manufacturing, we chose the monoplane as the best with overall attributions.

2. UNCONVENTIONAL CONFIGURATIONS:

Figured of merit: ease of manufacture, simplicity of design

After choosing a monoplane over a biplane, it was wise to consider the unconventional forms of a monoplane.

2.1. Canard:

A canard is not just an aircraft with the horizontal stabilizer shifted ahead of the wings. It is in fact, a very useful design. For example, any lifting surface ahead of a plane's center of gravity (CG), results in a reduction of longitudinal stability. If any pitch disturbance is to be corrected in an aircraft, the aerodynamic restoring forces have to be behind the CG. Consequently, for a canard design, the larger wing actually becomes the horizontal stabilizer.

The Wright Flyer was a canard. In a successful canard, the net aft center of gravity tail volume is more than adequate for proper static longitudinal stability. The canard, therefore, benefits not only in dynamic pitch stability from the main surfaces, but also from the forward lifting surfaces. For a successful canard, the forward surface has to be able to stall before the wing. This is because at the instant the forward lifting surface (canard) stalls, the nose of the aircraft drops slightly and lift is restored. If the nose of the aircraft was held up at this stall region, the canard goes from stalling to stability; the aircraft nose then bobs up and down as the aircraft continues to fly stably. Such an attribute is beneficial than conventional configuration.

For a canard to be inherently stable it must meet the following criteria:

a) The rate of change of pitching moment with respect to the center of gravity must be higher for the main wing than for the canard.

b) The canard must stall before the wing reaches its maximum lift.

c) The canard and main wing must have a minimum incidence difference, that is, a zero lift direction difference that cannot be lowered by the canard elevator or any other flap.

A successful canard usually has the following characteristics:

A high aspect ratio canard, that has a greater lift slope than the main wings, and is set at a higher incidence angle than that of the main wings. This insures that the canard stalls first. A successful canard could also have a swept back main wing, with tip mounted vertical fins, and rudders, to insure sufficient directional stability. This sweep back supplies ample static margin for longitudinal stability, and it also provides balance for a rear-mounted engine. In addition, these vertical fins reduce wing tip vortices.

The vertical placement of the canard with respect to the wing is important. This is due to the possibility of interference of the stall characteristics, by the downwash and tip vortices from the canard. The swept back wings, of the design however, is less efficient than straight wings.

Also, tip mounted vertical fins can be replaced by a single rear mounted vertical surface, as long as the rear of the airplane can support it. The canard design supports a portion of the aircraft's weight, whereas, in a conventional aircraft, the rear tail produced excessive load. On the other hand, a canard prevents the main wing from attaining its maximum lift. As a result, aircrafts that instill a canard configuration usually have higher landing speeds, and require greater landing and takeoff distances.

In summary, a properly designed and built canard offers an aircraft that is impossible to stall, an induced drag reduction, and the capacity to carry 20% of the aircraft's weight. However, a

canard design is disadvantageous since it defies simplicity in design, and requires longer field lengths. Consideration of the canard's design difficulty was enough for the concept to be scrapped; it would take too long for an adequate and well design plane.

2.2. Flying Wing:

Firstly, a flying wing, if not well designed, is unsteady in yawing. Theoretically, however, flying wings can be more efficient than the more conventional designs. In fact, the total drag can be as little as 50%, resulting in a 25% power reduction compared to the more conventional designs. The maximum range of a flying wing is 43% greater than other planes at the same cruising speed; such a plane would set us ahead of all competitors. Furthermore, the amount of cells used could be reduced, thus, reducing the overall gross weight of the aircraft. A flying wing, with such unrivaled performance, could go around the course almost twice the number of times as a conventional design (if all power and weight parameters are equal for both).

Various techniques can be used to bring about proper stability in flying wings. Some of which include, sweep back, a reflexed airfoil, and a wing tip washout. An airfoil that is reflexed, has an upturned trailing edge which eliminates nose-down pitching moment. One way to achieve stability in a flying wing is to select an airfoil with a mild or zero pitching moment for longitudinal stability. For lateral stability, some dihedral is put on the outboard wing panels and sweep backs are added to the center to assist in directional stability. Vertical fins, may also be attached to the wing tips to aid in directional stability. Rudders may also be added aft of the fins, to control the wing. The wing tip rudders provide both yaw and glide path control. If one rudder is deflected it produces a yaw, if both rudders are deflected, more drag is created. This increases the angle of descent, and therefore provides a good degree of glide path control. Turning is coordinated opposite to conventional designs; the rudder leads the turn followed by the aileron.

An additional technique for stability developed by Don Mitchell, who once worked for Northrop, is to hang stabilators a small distance below the outboard trailing edges. A stabilitor is a small auxiliary wing that can be rotated simultaneously for elevator effect, and asymmetrically for aileron effect. Stabilitors are generally added a small distance below the trailing edge, since the flow underneath the wing, is always attached at positive angles of attack and possess energy. The stall characteristics of a flying wing thus configured with swept wings, would not affect the ailerons, which means the aircraft would still be controllable. In general, flying wings have faster pitch response because of there low longitudinal damping.

Flying wings may have advantages over conventional configurations in terms of reduces drag, increased thrust, less power consumption and increased range. The underlying disadvantage is that it is difficult and time consuming to design and build. Its complexity is no match for the expertise and resources the team possess at this time. This configuration also increases the probability of pilot error, because controlling it is not similar to conventional configurations. No wonder the experts at Northrop Grumman designed a fly-by-wire mechanism (technology, that is

currently out of our reach). Also, there is no guarantee that the pilot, graciously provided by the competitors, has adequate experience with flying wings. All things considered, this year's design will not be a flying wing.

3. HIGH WING VS. LOW WING:

Figures of Merit: stability, lift distribution, accessibility of payload.

3.1. *Stability*: A high wing is superior to a low wing, because in a high wing the center of gravity is below the center of pressure of the wing. This creates a pendulum effect, that stabilizes the aircraft when it is disturbed from trim flight. This could be a very useful characteristic in a location such as Kansas. Also, a well-mounted high wing would reduce side slipping in a bank more effectively than a low wing would. For a small aircraft, the best a low wing could be is neutrally stable. Dihedral, is added to low wing to have better lateral stability. Dihedral usually ranges between one and three degrees. A wing has to have the right dihedral for effective performance with the rudder in a coordinated turn. Adding dihedral to our wing would only be a luxury, such a device is only necessary for a low wing.

3.2. *Lift Distribution:* A high wing possesses a better lift distribution than a low wing. Unlike a low wing, a high wing does not have its top surface interrupted by the fuselage. The top surface, due to its higher curvature is more important in the generation of lift than the bottom surface. So a high wing in general generates more lift than a low wing of similar dimensions.

3.3. *Payload Accessibility:* Previously, it appeared that our aircraft would be a high wing. After accessing the competition rules, however, which stipulate that each softball be loaded, and unloaded one at a time, without the aid of a speed loader, we were forced to instill a low wing design, that allows for easy payload accessibility.

Payload accessibility was considered paramount over stability and lift distribution, by a significant portion of the team. Consequently, the figures of merit had to be supplanted by a vote. Majority carried the day for a low wing, with dihedral to supplement the lack of stability.

4. WING PLANFORM:

F.O.M.: Stall characteristics, efficiency, ease of manufacture

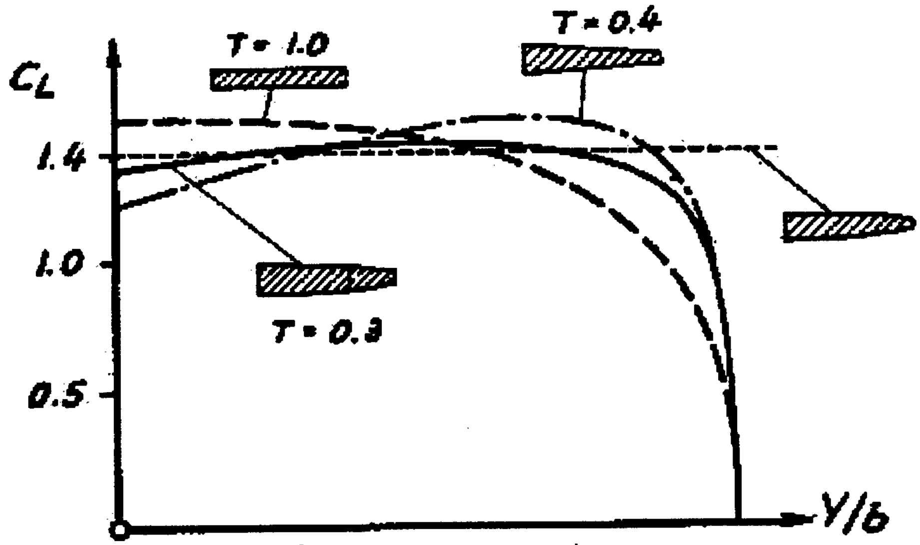
In the process of designing an efficient aircraft, the engineer has to decide which type of wing planform is most adequate. Since the length of the wingspan and the planform have significant effects on the aerodynamic characteristics of the airplane. Some of the aerodynamic characteristics, which are of interest when choosing the planform of a wing, are **efficiency, stall characteristics, the effective angle of attack, lift distribution and downwash. Ease of manufacturing is also important.**

In trying to optimize the efficiency of our airplane, we have considered three types of wing shapes, they are as follows: **Elliptical, Tapered and Rectangular**. Even though all these three wings are being used in current airplane design, they have several differences when it comes to

the aerodynamic characteristics mentioned above. For instance, downwash, a downward push to the air caused by the trailing vortices, tend to reduce the efficiency of the lift and retard the forward movement of the airplane. In essence, a rectangular wing shape, which has greater concentration of trailing of vortices, will generate more downwash than either an elliptical, or a tapered wing shape.

Since lift is proportional to the angle of attack, the effective angle of attack is also dependent to the planform shape of the wing. For example, a rectangular wing has its highest angle of attack at its roots; a tapered wing has its highest angle of attack approximately two-thirds of its semi span, while the elliptical wing has a uniform angle of attack along every section (figure 1), except at the latter. The rectangular wing, and the tapered wing, will not provide an even stalling sequence due to the different angles of attack along their respective span wise sections. The tapered wing should also start to stall at its outboard portion of the wing, considering, the location of its highest angle of attack. This situation is very unfavorable, because losing loft at this section of the span has a high potential for rolling. In addition to that, the surface for roll control (the aileron) is located at the stated section, so recovery of a roll, if needed, would be very unlikely. For that reason, it will not be easy to manage an airplane with tapered wing during the landing stage, where stalling is needed. Unlike tapered wing, the rectangular wing, tapered wing is efficient when it comes to stalling. The highest angle of attack on a rectangular wing is located at the root of the semi span, stalling would therefore start, at the inboard section, right where we want it, giving full control of the aileron to the pilot.

After examining the details of these three types of wing shapes, we decided to utilize the rectangular wing shape based on all its characteristics that we have mentioned above. An as a bonus to our plan, it is easy to manufacture, and provides the necessary lift for our design.



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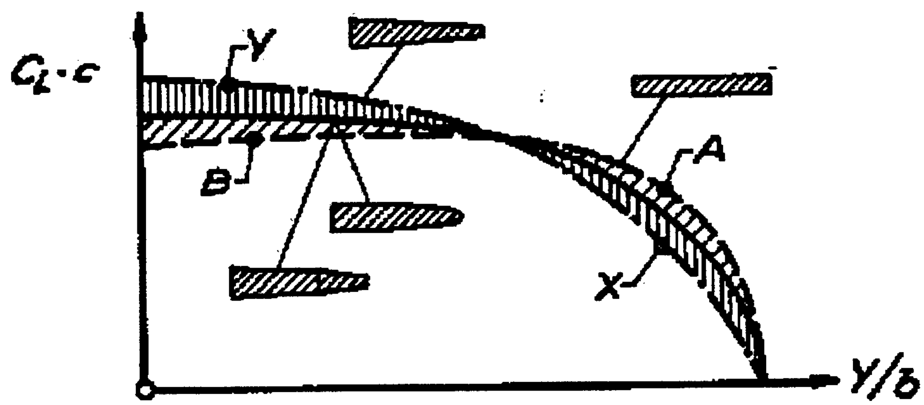

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FOR DIFFERENT WING PLANFORMS

FIGURE#1

5. DIRECTIONAL STABILITY AND CONTROL:

A conventional tail, with the horizontal stabilizer below the vertical stabilizer, was chosen for directional stability and control. The other configurations were considered and eliminated as follows:

5.1 V-Tails: A single V-tail could be incorporated upright or inverted. V-tails offer two control surfaces in place of three, this may reduce the drag generated by the empennage. V-tails are usually not subject to rotating slipstreams from the propeller. An inverted V-tail is better than an upright V-tail, because it offers favorable rolling moments with the rudder inputs, as does any

under-placed vertical tail. This is because a top mounted rudder produces a roll in the opposite direction of the intended yaw.

Some of the disadvantages of a V-tail are that they require complicated mixer mechanisms, which allow the moveable surfaces to act as both rudders and elevators. V-tails also respond to both vertical and horizontal gusts, this could lead to uncoordinated oscillation about the pitch and yaw axes that are difficult to correct. V-tails are thus complex in design, manufacture, and increases the probability of pilot error in application. Due to the complexity, the V-tai was eliminated.

5.5 <u>T-Tails:</u> T-tails require additional structuring, on the vertical tail, and aft fuselage. They generally have a more sophisticated control system. One possible advantage of a T-tail is that it can be placed away from the downwash of the wing, in a high configuration. This was not considered since our design, is a low wing configuration. The major disadvantage of the T-tail is that, if, it is placed improperly, it could lead to a deep stall. A deep stall is when the horizontal tail enters the wake of the stalled wing, and the unsteady vortices off the fuselage. IT is impossible to escape this kind of stall, this disastrous for any aircraft. With respect to our design, there are no advantages in using a T-tail; a T-tail is really extraneous, since our aircraft has a low wing.

6. <u>TRICYCLE VS. TAILDRAGGER</u>:

There is no question about the superiority of a wide tread tricycle arrangement during taxing. The tail dragger is simple unstable. With the CG behind the main wheels, any sideways motion of the aircraft will diverge it, inevitably leading the tail to swing around the nose of the aircraft. This is called a ground loop. On the other hand, the tricycle arrangement has the center of gravity positioned in front of the main wheels, which makes the aircraft straighten itself out. Another advantage of the tricycle landing gear is that it allows the wing to be at a no lift angle while on the ground. This will be useful in windy locations, such as Kansas, the site of this year's competition. Furthermore, this allows the airplane to take off only when the pilot wishes to do so. A tricycle landing gear, is thus more advantageous over a tail dragger, because of stability on the ground, and in keeping within the required field lengths.

7.<u>OTHER DESIGN CONSIDERATIONS:</u>

7.1 <u>Number of Engines: Single vs. Twin</u>

<u>Figures of Merit:</u> simplicity, ease of manufacture

The advantages of an aircraft with twin engines over an aircraft with single engines are as follows:

1) *Weight:* with twin engines, the aircraft can be guaranteed to carry twenty-four balls.

2) *Speed:* using twin engines, the aircraft could fly faster, because the performance relationship of thrust, divided by weight, could be greater than a single engine. A higher speed means a complete lap around the circuit in less time.

3) *Safety:* According FAA regulations, a twin-engine aircraft should be able to climb with one engine out. A properly designed twin engine6 aircraft that meets this specification may prove valuable in case of engine damage. If an engine quits in mid-flight, the aircraft could be returned to the ground with minimal damage. This guarantees that the aircraft stays in the competition.

The disadvantages:

a) *Complexity*: a twin-engine aircraft is more complicated than a single engine aircraft. The engines would have to be mounted on the wings. This is more complex than mounting an engine on the nose of the aircraft, as done on a single engine aircraft. In twin-engine concept, both engines and propellers must have similar performance; this could lead to intricate flight handling characteristics.

b) *Experience and Expertise:* The team has very little experience with single engine aircrafts, and no experience with twin-engine aircrafts. The expertise of the team is limited in the wiring of the engine and the support systems.

Considering the benefits and disadvantages of a twin-engine aircraft over a single engine aircraft, the disadvantages outweigh the advantages. A twin-engine aircraft is more complex than a single engine aircraft; this goes against both simplicity and ease of construction. A single engine aircraft is therefore the best option for this year's competition.

7.2 Tractor vs. Pusher

Figures of Merit: Experience and simplicity

Our team is more familiar with tractor configuration, than a pusher design. The only time a pusher was seriously considered was during the debate of a single engine over a twin-engine aircraft. In a twin engine-aircraft, a hybrid of a pusher or tractor, could be used, if, there was a malfunction with the engines. A pusher and a tractor configuration may also be used if a single-engine aircraft could not provide a high thrust to weight ratio. Using these mechanisms would eliminate the torque difference. Also, the pusher engine and propeller would be an auxiliary thrust generator that would increase the thrust to weight ratio. This concept with its advantages, again defies simplicity and ease of manufacture. A single tractor engine and a propeller is the best choice, considering simplicity and familiarity.

7.3 Fuselage Cross-section: Box vs. Circle

Figures of merit: drag, ease of manufacture and simplicity

A box fuselage is easier to manufacture than a circular fuselage, but a box fuselage can generate undesired aerodynamic drag during turns and climbs. A circular fuselage cross-section does not generate as much drag during turns and climbs. The drag generated while yawing is important; this year's competition requires the aircraft to perform many turns. On the other hand, a circular fuselage is more difficult to manufacture than a box fuselage. Both cross-sections are relatively simple to design. Consequent to the drag reduction in yawing, we chose to implement a circular fuselage.

7.4 **High Lift Systems, Flaps:**

The motivation behind the use of flaps arises because of the requirement, to fly loaded and unloaded sorties. An airplane outfitted to carry twenty-four balls, will have a ten-pound difference for each loaded and unloaded sortie. If the airplane is fitted with flaps, landing and take off qualities would be improved. This is because, for the loaded sortie, the aircraft could use its flap, while for unloaded sortie, it would not use the flaps. This could also make the field requirements uniformed, during the loaded, and unloaded sorties. An aircraft that is not outfitted for twenty-four balls may not need flaps.

Figures of Merit: Simplicity

The primary purpose of high-lift systems, such as flaps, is to obtain the highest Clmax. The secondary purpose is to obtain the highest L/D ratio, for the high CL used at take off. High lift systems consist of leading and trailing edge devices. A leading edge device is a slat, and a trailing edge device is a flap. The team focused on flaps as a possible high lift system. Our objective is to obtain high Clmax at the takeoff flaps deflection, but suffer the least possible drag at the lower CL, which the airplane normally flies at during the climb, after takeoff. A third requirement for high-lift systems is to develop reasonably high drag at the lift settings for landing. This would enable the aircraft to produce an adequate steep angle of decent in its final approach.

The high-lift system must meet these aerodynamic requirements, with the least possible structural weight. Flaps would require an extra servo, which means additional weight. The best starting point for a good high-lift system is an airfoil with the least possible structural weight. Flaps would require an extra servo, which means additional weight. The best starting point for a good high lift system is an airfoil with the highest clmax.

An airfoil with the highest clmax without flaps is a simpler and more attractive option. At this time while considering the simplicity of not including flaps, it is necessary to keep in mind the number of balls the aircraft is outfitted to carry. It becomes necessary to stipulate that if the airplane is outfitted to carry less than twenty-four balls, we shall not use flaps.

8. **Mission Feature of Figures of Merit:**

Simplicity: The figure of merit arose because of time constraint and the expertise of the team. A simple design will be make it easier to analyze during the aerodynamic design, performance prediction as well as draw. Moreover history teaches of the beauty and efficiency of simplicity. Some of the best aircraft designs that have stood the test of time have been simple designs. Simplicate, a word used by a famous aircraft designer and manufacturer will be incorporated into the design of this aircraft.

Ease of Manufacture: This figure of merit also arose as a result of time constraint. The team has considerable but limited experience in manufacturing of aircraft, especially in the handling of wood such as Balsa and Spruce wood. Due to the limited experience of the team it still remained

imperative for the aircraft to be easy to manufacture, especially given the time constraint. Ease of manufacturing is to manufacturing as simplicity is to design, both are very important.

Experience and expertise: These were implicitly reflected in the simplicity and ease of manufacturing figures of merit. In certain circumstances it became necessary to explicitly merit the experience and the expertise of the team with regard to specific configurations. This team is not very experienced and has little expertise in aircraft design and manufacturing. A configuration or concept may be simple to design, easy to manufacture, but if the team has no experience nor expertise with this concept it may be erroneous to further consider such a concept or configuration. An example is the pusher concept.

Other figures of merits: For example, drag in the case of fuselage cross-section, payload accessibility for the low wing, stall progression for the wing planform, and so on. These figures of merit served to improve the overall design.

9. CONFIGURATION SUMMARY:

The consideration of various aircraft configurations and components, has lead to the conclusion that the best configuration will be a conventional low wing monoplane with a rectangular planform, possessing a conventional empennage for directional stability, single engine, and using a tricycle landing gear.

PART 2: INITIAL AIRFOIL SELECTION

The airfoils considered are all NACA airfoils. This due to there popularity and this enables the analysis and information available on them to be more accessible as well as prolific. The goal of this initial airfoil analysis is to make a short list of high lift airfoils.

1. DESIGN PARAMETERS:

1.1. Thickness, t/c: Since the spar will occupy the point of maximum thickness, the wing has to be thick enough so that it can be supported internally. An airfoil with much less than 0.12 thickness could result in a heavy wing or one that cannot be supported internally. Also, in general higher thickness and camber leads to slower stall progression. As a result, 0.09 thickness is compared to 0.12 thickness airfoils. An airfoil of 0.11 thickness is included as a compromise between 0.09 and 0.12 thickness.

1.2. Camber: A cambered airfoil produces more lift than a non-cambered airfoil. Thus an airfoil of high camber produces more lift than an airfoil of less camber. Unfortunately the more camber an airfoil has the more drag it will produce. A trade off is required here. A good flap for a moderately cambered airfoil could compensate for lift during take off and landing, and thereby achieve moderate drag. Since flaps were eliminated during the first part of the conceptual design, it is imperative that the airfoil selected be a high lift airfoil. High lift airfoils generally have more camber that moderate lift and low lift airfoils. The drag and lift of an airfoil of high camber is beneficial during landing. An average airfoil has a 2% camber and may be considered a low lift airfoil. 4% camber is used more often in airfoils of light airplanes and may be considered

moderate airfoils. 6% camber airfoils are high-lift airfoils. Higher camber also increases the pitching moment of the wing. This will cause a need for larger stabilizing surfaces resulting in higher drag. The pitching moment was not considered at this stage of the airfoil selection.

1.3. **Angle of Zero Lift**: A less, i.e. more negative, angle of zero lift is beneficial during landing. It will enable the aircraft to establish adequately steep angles of descent during landing.

1.4. **Slope of Lift Curve**: The greater the slope of the lift curve, the greater the cl per alpha as well as clmax. This means an airfoil with a high slope will meet the criteria of being a high lift airfoil. The slope of the lift curve shall be implied in the lift coefficient at zero angle of attack, and angle of zero lift.

1.5. **Cl at Zero Angle of Attack**: This is a factor in determining the slope of the lift curve. Airfoils with higher lift slopes, have higher cl values at zero angle of attack. An airfoil with a high value at zero angle of attack, is a high lift airfoil.

1.6. **Stall Characteristics**: The stall characteristics of airfoils are important, because it determines the behavior of the aircraft when the wings stall. An airfoil with a sharp stall will be eliminated, because it means the aircraft will drop suddenly when the wings stall. This is an undesirable characteristic for most airplanes. An airfoil with a blunt stall, i.e. mush, will enable the airplane to slowly drop in altitude indicating to the pilot to reduce the angle of attack. This is a very desirable characteristic for airplanes.

1.7. **Coefficient of Drag**: The effects of drag were considered negligible at this stage in the selection process. The software used shows the drag coefficients unchanged at zero angle of attack. Even though this is questionable, we decided to focus on the high-lift characteristics and ignore the drag on the airfoils at this stage in the design.

Re = 300,000

Airfoil (NACA)	**Approx. angle of zero lift (degrees)**	**Angle at zero alpha**	**Good stall characteristics**	**Coefficient of drag at zero alpha**
2409	-2	0.2368	Bad – very sharp	0.0064
2412	-2	0.2422	Bad – very Sharp	0.0064
4409	-4	0.4729	Bad – marginally sharp	0.0064
4412	-4	0.4837	Decent – but still a little sharp	0.0064
5409	-5	0.5905	Not Available	0.0064
5411	-5	0.6039	Not Available	0.0064

Conclusion: The two airfoils that meet the criteria and so merit further analysis are the NACA 5411 and NACA 6412. Even though the stall characteristic of the 5411 airfoil could not be visually

determined, it can be approximated. This is because thicker airfoils tend to have good stall characteristic, and their lift curves tend to have a mush tendency at the stalled area. The main conclusion of this analysis is that both airfoils are high-lift airfoils with good stall characteristics.

CONCEPTUAL DESIGN REFERENCES:

1. http://backoff.pr.erau.edu/gallyt/Naca4.htm
2. http://naca.larc.nasa.gov/reports/1933/naca-report-460/start
3. http://w3.iac.net/~feguy/soaring_symposia/71-wvu.html
4. Smith, H.C. "skip" The illustrated Guide to Aerodynamics (1992). USA: TAB books.

PRELIMINARY DESIGN REFERENCES:

1. http://beadec1.ea.bs.dlr.de/airfoi /calcfoil_olde.htm
2. http://www.pagendarm.de/trapp/programming/java/profiles/NACA4.html
3. "An Introduction to the Flight Dynamics of Rigid Aeroplanes", G.J. Hancock, Ellis Horwood 1995
4. R.S. Shevell, Fundamentals of Flight, 2nd ed. Prentice Hall, (1988).
5. W. A.. Mair, Aircraft Performance, Cambridge University Press, (1992).
6. Simple Science of Flight, Henk Tennekes, The MIT Press, (1997).

http://courseware.vt.edu//users/M/marchman/Chpt4.pdf

8. http://www.aoe.vt.edu/~cliff/aoe3104/aoe3104.html

DETAIL DESIGN REFERENCES:

1. R.S. Shevell, Fundamentals of Flight, 2nd ed. Prentice Hall, (1988).
2. J.D. Anderson, Introduction to Flight, 3rd ed. MaGraw-Hill, New York, (1989).
3. Pilots Weight and Balance Handbook, U.S. Department of Transportation, Federal Aviation Administration
4. http://courseware.vt.edu//users/M/marchman/Chpt4.pdf
5. www.astroflight.com
6. Ultralight Flight, Micheal A. Markowski, Ultralight Publication, (1982)
7. Aerodynamics for Naval Aviators, H.H. Hurt, Jr., U.S. Navy, (1960)

MANUFACTURING PLAN REFERENCE:

http://www.brucescycleworks.com/tips/tip18.html

Rated aircraft cost:		Manufacturing Hours Required to build	Monoplane Dimensions	Costs	Flying Wing Dimensions	Costs	Canard Dimensions	Costs	V-Tail Dimensions	Costs
MEW	gross weight		25		25		25		25	
REP	Nr. Of engines		1		1		1		1	
	weight of batteries		5		5		5		5	
	Engine Power cost	(1 + (#engines - 1)) * 5		5		5		5		5
Manufacturing Man Hours										
Wings	Wingspan	8 hrs/ft	8	64	8	64	8	64	8	64
	Nr. Of control Surfaces	3 hrs/surface	2	6	4	12	2	6	2	6
Fuselage	Max body length	10hrs/ft	7	70	7	70	7	70	7	70
Empennage										
Vertical Surface	w/ active controls	10 hrs/vert. Surf.	1	10	0	0	1	10	0	0
	w/o active controls	5 hrs/vert. Surf	0	0	0	0	0	0	1	5
Horizontal Surface	w/ active controls	10 hrs/vert. Surf.	1	10	0	0		10	1	10
Flight Systems	Nr. Of servos or	5 hrs/servo	3	15	2	10	3	15	3	15
	Motor controllers	5 hrs/controller	1	5	1	5	1	5	1	5
Propulsion	Nr. Of engines	5 hrs/engine	1	5	1	5	1	5	1	5
	Nr. Of propellors	5 hrs/propeller	1	5	1	5	1	5	1	5
		Total Hours		195		176		195		190
		Rated aircraft costs	13.9		13.52		13.9		13.8	

FINAL RANKING CHART

Aircraft Configuration	Figures of Merits								
	Low drag	Simplicy of design	Efficiency	Ease of manufacture	Mission feature	Stability	Lift distribution	Stall characteristics	Accesibility of payload
Monoplane vs. Biplane									
Monoplane	1	1	N/A	1	N/A	N/A	N/A	N/A	N/A
Biplane	0	0	N/A	N/A	N/A	N/A	N/A	N/A	N/A
Unconventional configurations									
Canard	0	0		0		1	N/A	N/A	N/A
Flying Wing	1	0		0		0	N/A	N/A	N/A
High wing vs. Low wing									
High Wing	N/A	N/A	N/A	N/A	N/A	1	1	N/A	0
Low Wing	N/A	N/A	N/A	N/A	N/A	0	0	N/A	1
Wing Planform									
Rectangular wing shape	0	1	1	1	N/A	N/A		1	N/A
Elliptical wing shape	1	0	1	0			1	1	N/A
Tapered wing shape	1	0	1	0				0	
Directional Stability / control									
V-Tails	1	0		0	N/A		N/A	N/A	N/A
T-Tails		1	1	1	N/A		N/A	N/A	N/A
Tricycle vs. Taildragger									
Tricycle	N/A	N/A	N/A	N/A	N/A	1	N/A	N/A	N/A
Taildragger	N/A	N/A	N/A	N/A	N/A	0	N/A	N/A	N/A
Other design Considerations									
Single engine		1	1	1	N/A	N/A	N/A	N/A	N/A
Twin engine		0	1	0	N/A	N/A	N/A	N/A	N/A
Tractor	N/A	1	N/A	N/A	N/A	N/A	N/A	N/A	N/A
Pusher	N/A	0	N/A	N/A	N/A	N/A	N/A	N/A	N/A
High Lift Systems, Flaps			1		1		N/A		N/A
Fuselage Cross-section:									
*Box	0	1		1	N/A	N/A	N/A	N/A	N/A
*Circle	1	1	1	0	N/A	N/A	N/A	N/A	N/A

Preliminary Design Report

1. INTRODUCTION

From the beginning of this project certain restrictions were placed on the size of the wing. These restrictions arose as a result of some of the lessons learnt from last year's competition. The dimensions of the wings were estimated to be ten feet in span and a chord length of one foot. The chord length was fixed at one foot, because of ease of manufacture. A chord length of one foot is easier to accurately cut. After structural strength consideration and the bending loads that a long wingspan could create, a wingspan of eight feet was proposed parallel to the ten-foot wingspan. The tasks were then to determine the amount of softballs that each wing could carry and the necessary coefficient of lift at cruise. After merit consideration, performance and drag are predicted for the selected wing.

2. WEIGHT ESTIMATION:

The goal of the weight estimation is to determine the approximate weight of a fully loaded airplane carrying twenty-four balls as well as that of a fully loaded airplane carrying twelve balls. A payload of twenty-four softballs was selected to maximize the score and for a ten feet wingspan. A payload of twelve softballs was selected as a safe payload capacity for an eight feet wingspan. A twelve softball payload aircraft is a simpler aircraft to design, build and fly.

2.1 Twenty-four Softballs Payload: The approximate weight of an airplane that can carry the maximum number of balls was investigated. The weight estimate for the airplane to carry twenty-four balls is as follows:

i. 24 softballs: 24 * 0.41 pounds = 9.84 pounds
ii. 4 servos: 4 * 0.121 pounds = 0.5 pounds
iii. Motor + gearbox + electronic speed control + drive system = 6 pounds
iv. Batteries: 42 cells * 0.125 pounds = 5.25 pounds
v. Total: (9.84 + 0.5 + 6 + 5.25) = 21.59 pounds

The maximum number of softballs an airplane can carry is twenty-four balls. One softball weighs 0.41 pounds. It was determined that four servos will be adequate for the control and operation of the aircraft in flight. One servo for the ailerons, one servo for the rudder, one servo for the elevators and one servo to control the front landing gear. The servo used for the estimation is the Futaba servo FP-S9402, of speed 0.10 sec./ 60 degrees. The torque provided by the servo is 111 oz-in (8.0 kg-cm), at 6.0 volts. The size of each servo is 1.59 inches * 0.79 inches * 1.48 inches (40.5 mm * 20 mm * 37.5 mm), the weight of each is 1.94 ounces (55 g).

An adequate airplane structure including the landing gears and the propeller should be at least equal to the estimation done above. Multiplying 21.56 pounds by 2 gives approximately 43 Pounds. Forty-three pounds was increased to fifty pounds, to provide for some leeway. The result of the structural estimation including the landing gears and propeller is as follows:

vi. Structural weight + landing gears + propeller = 50 –22 = 28 pounds.

2.2. Twelve Softballs Payload: Using the same weight composition used above, the weight estimation of an airplane capable of carrying a maximum of twelve softballs is as follows:

i. 12 softballs: 12 * 0.41 pounds = 4.92 pounds

ii. 4 servos: 4 * 0.121 pounds = 0.5 pounds

iii. Motor + gearbox + electronic speed control + drive system = 6 pounds

iv. Batteries: 33 cells * 0.125 pounds = 4.125 pounds

v. Total = 5 + 6 + 0.5 + 4.92 = 15.5 pounds

The amount of structural weight, including the landing gears and propeller, is determined if a takeoff gross weight of the fully loaded airplane is determined to be 30 pounds.

vi. Structural weight, landing gear, and propeller = (30 – 15.5) pounds = 14.5 pounds.

2.3. Figures of Merit: score and simplicity.

2.3.1. Mission Feature

i. Score: a higher capacity airplane will attain a higher score if well designed.

ii. Simplicity: a higher capacity airplane will be harder to design than an airplane of less capacity. Twenty-four softballs aircraft may require a longer wing and two motors. A long wing requires more internal structure to prevent bucking due to the high bending loads at the root of the wing. Also a payload of twenty-four softballs may require two motors. Simplicity is not a characteristic of an airplane designed to carry twenty-four softballs.

Payload Capacity	Score	Simplicity	Total
Twenty-four Softballs	1	0	1
Twelve softballs	0	1	1

2.3.2. Conclusion: The use of weight estimation as a design parameter is inconclusive. The choice of payload capacity shall be determined by another design parameter.

3. CRUISE CONDITION; CL

With the aid of this design parameter we shall screen the maximum payload. The design parameters investigated here are cruise lift coefficients, cruise speed, and surface area of the wing. These design parameters are determined using the lift force equation. At cruise condition the lift force is equal to the weight. As a result, we can use these design parameters to screen the maximum payload. We investigate the cruise condition of both maximum payloads of twenty-four softballs and twelve softballs. From the conceptual design stage, the target lift coefficient at cruise is assumed to be about 0.7.

Birds that fly at speed less than 32 ft/s (10 m/s) find it difficult to return to their nests in a strong wind or storm. In consideration of this and the windy condition the airplane may experience at the competition site, we assume cruise speeds of above 62 ft/s (19 m/s). This will enable the aircraft to maintain its performance characteristics when flying under windy conditions. The elevation of Wichita, Kansas was determined to be at about one thousand feet above sea level.

As a result the density at 1000 feet above sea level is used during the analysis. The density at 1000 feet above sea level is 0.002308 slug/(feet)^3.

3.1. <u>Payload: Twenty-four Softballs</u>:

Takeoff gross weight, W = 50 pounds (22.65 kg)

Assumed cruise speed, V = 62 ft/s (19 m/s)

Surface area of wing, S = 10 ft^2 (0.93 m^2)

Density, ρ = 0.002308 slugs/ft^3 (1.2 kg/m^3)

The coefficient of lift required at cruise is given by the equation:

$$CL = W / (0.5 * \rho * V^2 * S) = 1.1$$

Comparing this value of the lift coefficient at cruise with the values of cl of the airfoils screened in the conceptual design stage, we found out that this cruise lift coefficient is too high. This is because the angle of attack necessary to achieve this value in straight and level flight will be too high and too close to the maximum lift coefficient attainable. The value of the lift coefficient has to be reduced.

To reduce this derived value of the lift coefficient at cruise condition, it is best to increase the speed. This is because the lift coefficient is inversely proportional to the square of the speed. If the speed is increased to 79 ft/s (24 m/s), we can attain the target lift coefficient of 0.7. The disadvantage is the high value of the speed. Decreasing the speed to 67 ft/s (20 m/s), and the wing area is increased to 14 ft^2 (1.2 m^2) will give us the target lift coefficient. The disadvantage that arises here is the large wing area, especially when the chord length is limited to one foot.

Two other set of values were considered. The first with the wing area equal to 12 (ft)^2 or 1.1 m^2 and the cruise velocity equal to 67 ft/s (20 m/s) gives a lift coefficient of 0.77. The lift coefficient obtained is still significantly above the targeted lift coefficient and the wingspan is still too long. Adjusting the values again, the wing area becomes 11 ft^2 (1 m^2) and the cruise speed is now equal to 69 ft/s (21 m/s), we attain our targeted lift coefficient at cruise condition.

In order to carry twenty-four softballs, the airplane should have a loaded gross weight of 50 pounds. The wingspan is 11 ft, the chord length is one foot, and the cruise speed should be at least 69 ft/s. It was assumed earlier that the wings should be ten feet, but a ten feet wing cannot attain the desired lift coefficient at cruise within reasons. As a result, it was necessary to compromise wingspan in order to attain the desired lift coefficient.

3.2. <u>Payload: Twelve Softballs</u>: The task of determining the parameters necessary for cruise condition in this case, is somewhat different from the task above. In this case the parameters have been already been specified. The payload is twelve balls; this leads to a thirty-pound airplane. The lift coefficient at cruise is estimates to be about 0.7, and the wing area has to be eight feet. The speed determined above for the payload amount of twenty-four softballs, is an ideal speed for this configuration as well. The task here is to justify the assumptions made earlier. Therefore,

Lift coefficient assumed, CL = 0.7

Cruise speed, V = 69 ft/s (21 m/s)

Wing area, S = 8 ft^2 (0.74 m^2)

Density at 1000 ft, ρ = 0.002308 slug/ft^3

Weight, W = 0.5 * CL * ρ * V^2 * S = 30.2 pounds

The result attained justifies the assumptions that were made earlier. This calculation and analysis proves that the best maximum payload the airplane should carry is twelve softballs. To confirm this, the next step is to screen the parameters of both maximum payload configurations using figures of merit.

3.3 Figures of Merit: wingspan, simplicity

3.3.1. Mission Feature:

i. Wingspan: A long wingspan requires additional internal structural weight. Adding to the weight of the aircraft and also making the wing more difficult to construct. A long wingspan creates higher bending loads on the root of the wing. Due to the manner with which the wing will be joined to the fuselage, consideration of bending loads is important.

ii. Simplicity: Although the values of the speed and CL are the same for both payloads, the higher capacity airplane will be harder to design than the airplane of less capacity. The airplane carrying twenty-four softballs does require a longer wing. Based on the success of the confirmation of the previous assumption, it is very likely that this payload will require two motors. It is also very likely that an aircraft that uses two engines would require flaps, especially during landing, to be efficient. Otherwise, with one engine would require flaps during take-off. Flaps where ruled out in the conceptual design stage. Simplicity is certainly not a characteristic of an airplane designed to carry twenty-four softballs.

Payload	Wingspan	Simplicity	Total
Twenty-four Softballs	0	0	0
Twelve Softballs	1	1	2

3.3.2. Conclusion: The choice becomes clear now that more design parameters have been considered. The best airplane to design, build and fly has a maximum payload of twelve softballs.

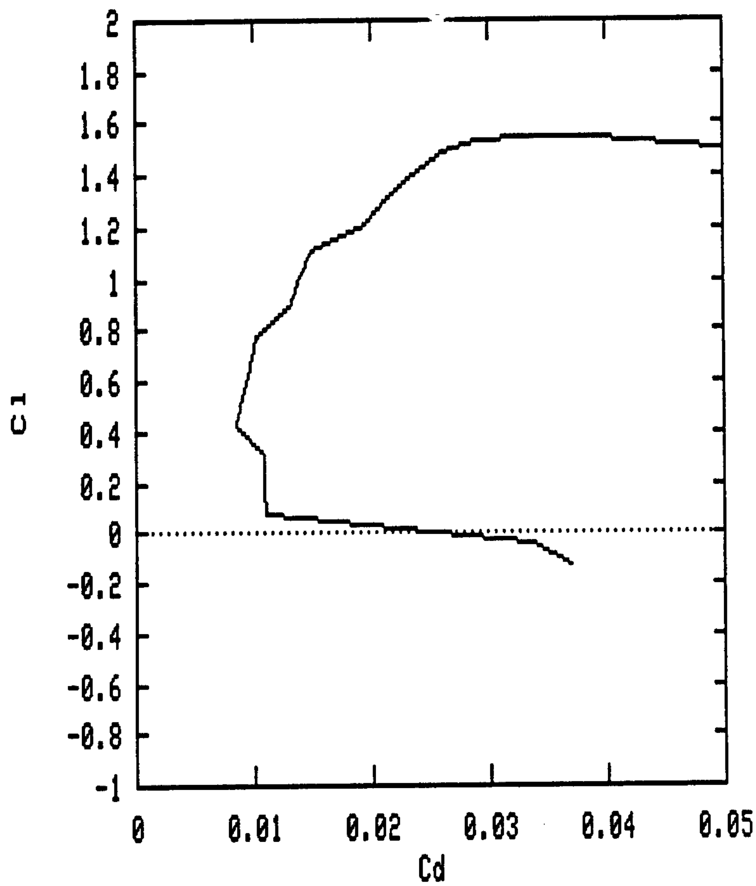
4. WING DESIGN

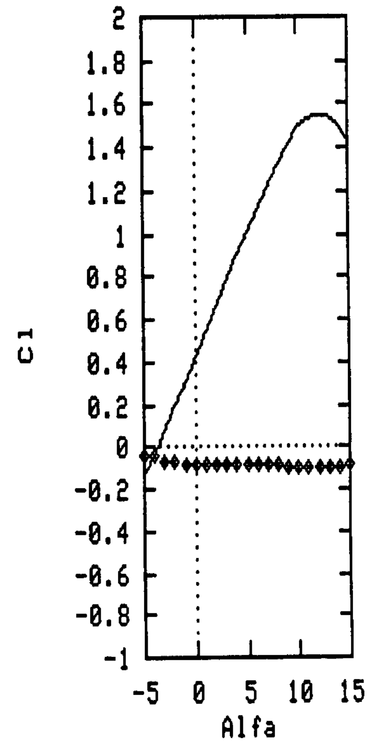
The design of the wing comprises selection of airfoil, correcting the lift curve slope of the airfoil to an aspect ratio of eight, as well as the determination of the wing incident angle.

4.1 Airfoil Selection: From the conceptual design phase, two airfoils where chosen for further analysis. These two airfoils where chosen, because of their high-lift and stall characteristics. The analysis of these airfoils was performed using a program called Calcfoil [1]. This is a program available over the Internet. The coordinates of the airfoils are input into Calcfoil. The coordinates where obtained from a NACA airfoil generator [2] also accessible through the Internet. The other

major input is the Reynolds number. The Reynolds number of the airflow over the wing is 400,000 but at the time this was not known. The Reynolds number used in this airfoil analysis is an average value of 300,000. This average value is the same encountered by model airplanes of comparable size to our airplane. The Reynolds number is an important factor for airfoil analysis. Although, airfoil characteristics do not vary much for Reynolds numbers within a certain range of each other. As a result, we were not concerned when we later determined the Reynolds number over the wing to be 400,000. The two airfoils chosen in the conceptual design phase are NACA 5412 and NACA 6412.

4.1.1. <u>Results of NACA 5411 Airfoil Analysis</u>:





<u>Polars in Table Format</u>

Airfoil Analysis by http://beadec1.ea.bs.dlr.de/Airfoils/calcfoil.htm

Data Table

Airfoil: NACA 5411

Alfa ... angle of attack relative to the x-axis

cl ... lift coefficient

cd ... drag coefficient

cm ... moment coefficient

T.U. ... x/c location of transition on upper surface

T.L. ... x/c location of transition on lower surface

S.U. ... x/c location of separation on upper surface

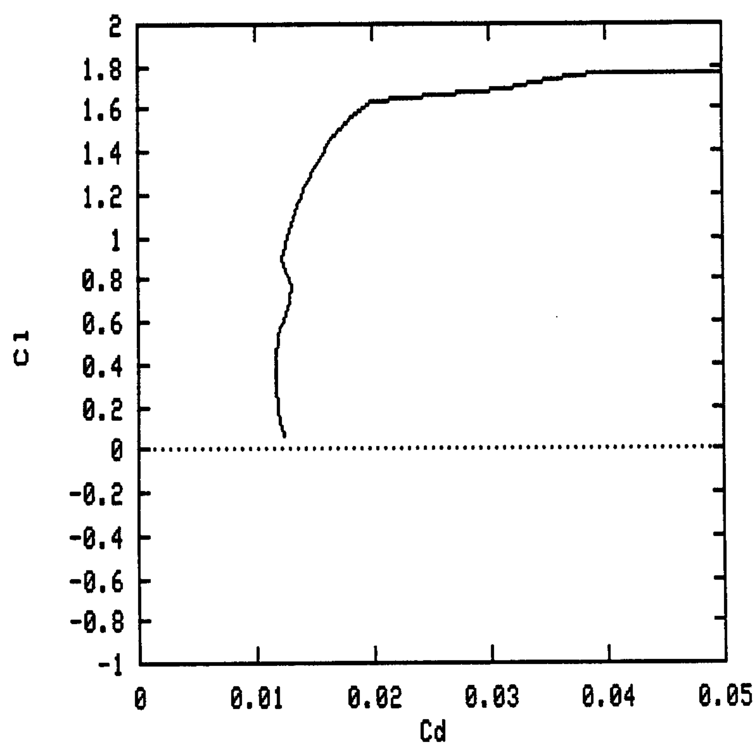
S.L. ... x/c location of separation on lower surface

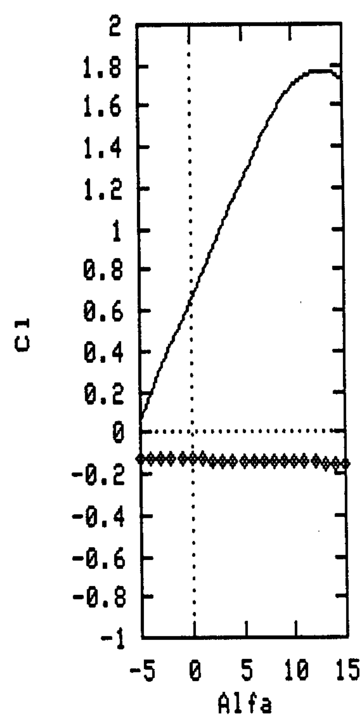
L/D. ... lift over drag ratio (glide ratio)

Alfa	Cl	Cd	Cm	T.U.	T.L.	S.U.	S.L.	L/D	L^(3/2)/D
-5.00	-0.13	0.0370	-0.0393	0.6977	0.0413	0.9642	0.0526	-3.5463	0.0000
-4.00	-0.04	0.0337	-0.0406	0.6702	0.0422	0.9640	0.0561	-1.0993	0.0000
-3.00	0.07	0.0110	-0.0732	0.6462	0.0436	0.9633	0.9772	6.5158	1.7470
-2.00	0.19	0.0108	-0.0746	0.6217	0.0460	0.9624	0.9800	17.5873	7.6785
-1.00	0.31	0.0108	-0.0761	0.5965	0.0383	0.9620	0.9822	28.7792	16.0282
0.00	0.43	0.0085	-0.0771	0.5688	0.8615	0.9606	0.9574	50.7006	33.2765
1.00	0.55	0.0090	-0.0785	0.5263	0.8989	0.9592	0.9490	60.5445	44.7019
2.00	0.66	0.0096	-0.0799	0.4882	0.9090	0.9573	0.9498	69.0986	56.2377
3.00	0.78	0.0102	-0.0814	0.4579	0.9191	0.9553	0.9491	76.0293	67.1094
4.00	0.89	0.0130	-0.0829	0.2648	0.9314	0.9503	0.9472	68.7780	65.0261

Table cut for brevity

4.1.2. **Results of NACA 6412 Airfoil Analysis**:





Polars in Table Format.

Airfoil Analysis by http://beadec1.ea.bs.dlr.de/Airfoils/calcfoil.htm

Data Table

Airfoil: NACA 6412

Alfa ... angle of attack relative to the x-axis

cl ... lift coefficient

cd ... drag coefficient

cm ... moment coefficient

T.U. ... x/c location of transition on upper surface

T.L. ... x/c location of transition on lower surface

S.U. ... x/c location of separation on upper surface

S.L. ... x/c location of separation on lower surface

L/D. ... lift over drag ratio (glide ratio)

Alfa	**Cl**	**Cd**	**Cm**	**T.U.**	**T.L.**	**S.U.**	**S.L.**	**L/D**	**L^(3/2)/D**
-5.00	0.07	0.0124	-0.1255	0.6941	0.0442	0.9348	0.9380	5.2953	1.3582
-4.00	0.18	0.0119	-0.1268	0.6651	0.0451	0.9352	0.9437	15.5106	6.6534
-3.00	0.30	0.0117	-0.1280	0.6320	0.0496	0.9355	0.9448	26.0521	14.3605
-2.00	0.42	0.0117	-0.1293	0.5925	0.0606	0.9355	0.9453	36.3671	23.6910
-1.00	0.54	0.0119	-0.1306	0.5413	0.0852	0.9355	0.9456	45.6594	33.6870
0.00	0.66	0.0126	-0.1320	0.4649	0.1306	0.9350	0.9457	52.2871	42.5187
1.00	0.78	0.0131	-0.1334	0.4327	0.2012	0.9345	0.9455	59.6658	52.6554
2.00	0.90	0.0122	-0.1344	0.4209	0.9267	0.9336	0.9307	73.4284	69.4956
3.00	1.01	0.0128	-0.1360	0.4169	0.9270	0.9331	0.9306	79.2439	79.7209
4.00	1.13	0.0134	-0.1375	0.4100	0.9276	0.9320	0.9308	84.0469	89.2442

4.1.3. Discussion: This analysis probably contains many inherent errors, but it serves as a relatively good approximation. Also, because both airfoils are analyzed with the same methods their characteristics can be compared with accuracy. The first and only criteria that has been considered so far is the coefficient of lift at cruise condition, which was determined to be 0.7. The NACA 5411 can achieve this lift coefficient at angles of attack above 3 degrees. The NACA 6412 can achieve this lift coefficient at angles of attack above 1 degree. If an approximation of the amount of error inherent within the analysis, and the lift curve slope correction for a wing of aspect ratio 8, we can intuitively assume that NACA 5411 can achieve a 0.7 lift coefficient at 4

degrees angle of attack. The NACA 6412 can be assumed to achieve this lift coefficient at 2 degrees angle of attack.

The NACA 6412 shows superiority in achieving the determined cruise lift coefficient of 0.7. Further proof of this conclusion is that at 2 degrees NACA 6412 has a coefficient of drag of 0.0122. This is less than the coefficient of drag of the NACA 5411 at 4 degrees angle of attack. Another distinguishing characteristic of the NACA 6412 over NACA 5411 is the value of clmax achieved. The clmax of NACA 6412 is 1.77 at 13 degrees angle of attack, while that of NACA 5411 is 1.55 at 12 degrees angle of attack. Therefore the slope of the lift curve for NACA 6412 is greater than that of NACA 5411. This means that realistically we can expect to gain more lift per angle of attack increased for NACA 6412 than NACA 5411. This is another why NACA 6412 is more efficient than NACA 5411 for our airplane.

4.1.5. Figures of Merit:

i. *0.7 Cruise Lift Coefficient*: the mission feature of this figure of merit is to select an airfoil that would give us a lift coefficient of 0.7.

ii. *Angle of Attack at Cruise*: The angle of attack that the airfoil to be selected meets the cruise lift coefficient criteria must be low within reason. If this angle of attack is too high, it could reflect in the drag generated and also effectively decreases the angle of attack at which the airplane can achieve clmax. This would be disadvantageous during landing and when the airplane is hit by a gust. The higher this angle of attack is, the more chances the airplane stall.

iii. *clmax*: an airfoil with a high clmax value is more desirable, because a high clmax value will be an asset for a wing without flaps during landing and taking-off. A high clmax value, achieved at a high angle of attack, also adds to the airfoil's superiority.

iv. *Drag at 0.7 lift coefficient*: An airfoil that can achieve the desired lift coefficient of 0.7 and generates less drag at this lift coefficient is more desirable for our purposes.

Figures of Merit Table

Airfoil	Cl = 0.7	Alpha at 0.7	Clmax	Drag at 0.7	Total
NACA 5411	0	0	0	0	0
NCA 6412	1	1	1	1	4

4.1.4. Conclusion: NACA 6412 is conclusively the more superior airfoil for our purposes. At this stage of the design, NACA 6412 has met all the criteria that have been specified for the wing section of the airplane. The only possible setback to the NACA 6412, are its moment coefficients. The disadvantage may arise in the size of the horizontal stabilizer. At this stage the moment coefficient will not be considered a design parameter, because of simplicity.

4.2. Lift Curve Slope Correction: After the selection of the airfoil, it is necessary to fit the lift curve of the airfoil to a wing. An airfoil is the two-dimensional equivalent of a wing, which is three-dimensional. This correction is necessary, because the lower the aspect ratio of a wing,

the higher the downwash at a given CL, and the higher the angle of attack (geometric) required to achieve that CL. Therefore, $CL = ao * \alpha$

Where, ao is the slope of the airfoil lift curve and α is the angle of attack (geometric angle of attack). The geometric angle of attack decreases with downwash linearly.

The slope of the wing lift curve is as follows:

$$dCL / d\alpha = a = ao / (1 + (57.3 * ao / (pi * AR)))$$

If AR, aspect ratio, is infinite, then a is equal to ao. Evidently, an airfoil has an infinite aspect ratio, while the wing of our airplane has an aspect ratio of eight.

4.2.1. <u>Slope of Airfoil Lift Curve, ao</u>: To determine the slope of the airfoil lift curve, the lift coefficient was plotted against the angle of attack using Matlab. The lift curve is linear over a certain range of cl values, before the stall. The range over which the lift curve is exactly linear is for cl values corresponding to the alpha values ranging from –5 to 6 degrees. For cl values corresponding to alpha values between –5 and 8 degrees the lift curve is approximately linear. The range that would give an exact linear curve was selected.

Plotting cl versus alpha that range from –5 to 6 degrees, and then using the Matlab feature that does a basic linear curve fit, resulted in the linear equation:

$$cl = 0.12 (\alpha) + 0.66$$

This equation gives the slope of the airfoil lift curve slope to be 0.12.

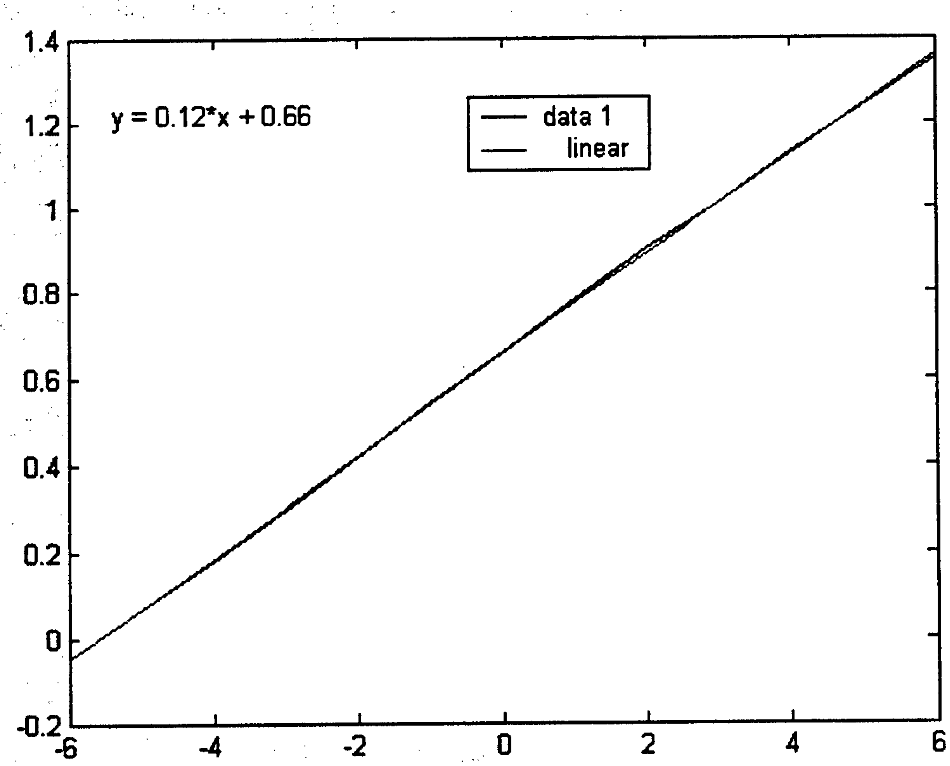


Figure: Airfoil Lift Curve Slope: This figure can also be used to approximate the zero-lift angle of

attack, which is about 5.5 degrees.

4.2.2. Slope of Wing Lift Curve a: The slope of the wing is determines from the value of the slope of the lift curve using the following equation:

$$dCL / d\alpha = a = ao / (1 + (57.3 * ao / (pi * AR)))$$

where, AR = 8

ao = 0.12

Therefore,

a = 0.0942

Using this new slope, we can find the equation of the linear part of the wing lift curve

4.2.3. Lift Curve of the Wing: Using the value of a, the equation of the linear part of the wing lift curve is determined using ordinary differential equations.

$a = dCL / d\alpha = 0.0942$

the initial condition is: $CL(\alpha o) = 0$

$$\alpha o = -0.66 / 0.12 = -5.5$$

where, αo is the angle of zero lift.

Solving for CL in terms of alpha gives,

$CL(\alpha) = 0.0942(\alpha) + 0.52.$

This equation gives the linear part of the lift curve of the wing.

To determine the entire lift curve, that is the linear portion as well as the stall area, the cruise lift coefficient is used. The cruise lift coefficient was determined to be 0.7, and the angle attack at which the wing would achieve this cruise lift coefficient has been assumed to be 2 degrees. To confirm these assumptions and approximations, the CL at 2 degrees is determined using the equation for the linear part of the wing lift curve. Plugging in α being equal to 2, the CL value obtained is 0.7084, or approximately 0.71.

Our approximations were very accurate and have been justified beyond reasonable doubt. Since the approximated lift coefficient at 2 degrees has been confirmed, a ratio of 0.7/0.9 is multiplied by the values of the airfoil lift coefficients to determine the wing lift coefficients. This ratio is used, because the lift coefficient at 2 degrees is 0.7 and 0.9 for the wing and airfoil, respectively. 0.7 is used instead of 0.71 to accommodate any error that might occur both in design and in building of the airplane. The resulting wing lift curve, the linear wing lift curve and the original airfoil lift curve, are shown below:

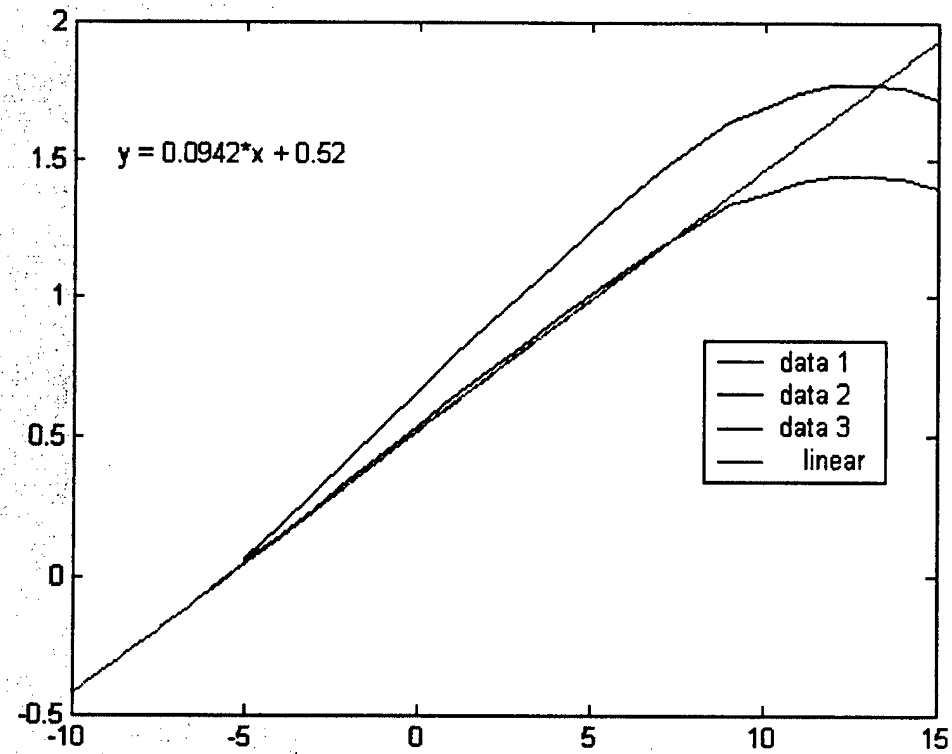


figure : lift curves of wing, and airfoil. Depicts the accuracy of the calculations, and justifies all prior assumptions.

4.3. <u>Wing Incident Angle</u>: The selection of the wing incident angle is an important aspect in wing-fuselage integration. The wing incident angle is the angle between the wing root chord and the fuselage reference line. The fuselage reference line lies in the plane of symmetry, and is parallel to what may be called the cabin floor. Since the airfoil has been chosen, the main criteria here is that the wing incident angle to be the angle at which the wing can achieve the 0.7 cruise lift coefficient. That angle has been determined to be two degrees angle of attack. An added benefit of choosing two degrees as the wing incident angle is the relatively low drag of the airfoil at this angle than at the immediate angles below and above it. The wing incident angle shall be two degrees.

5. <u>DIMENSIONS OF THE AIRPLANE</u>:

The dimensions of the airplane were obtained from reference 3 of the preliminary report. The dimensions were written in ratios of the chord length of the wing. The dimensions are those of a light aircraft.

5.1. <u>Wings</u>: The wing has been defined to have a wingspan of eight feet and a chord length of one foot.

i. Wingspan, b = 8 ft

ii. Chord length, cw = 1 ft

5.2. <u>Fuselage Length</u>:

i. Distance from approximated center of gravity to trailing edge of fin = 4.75.0 * cw = 4.75 ft

ii. Distance from the fuselage nose to approximated center of gravity = 2.25 * cw = 2.25 ft

iii. Distance of horizontal stabilizer apex aft of wing apex = 4.6 * cw = 4.6 ft

iv. Distance of wing apex aft of fuselage nose = 2.0 * cw = 2.0 ft

v. Distance of wing apex forward of fin apex = 4.2 * cw = 4.4 ft

Therefore, the total length of the fuselage = (i) +(ii) = 2.25 ft + 4.75 ft = 7 ft

5.3. <u>Fuselage Width and Softball Configuration</u>: The width of the fuselage was determined by the configuration of the softballs. The circumference of a softball is one foot, therefore, 1 ft = $2*\pi*r$, and the diameter, d = $2*r = 1/\pi = 0.32$.

Twelve softballs is the maximum payload of the aircraft. The possible configurations in which the balls can be arranged are specified length by width as follows, three by four, four by three, and six by two.

5.4. <u>Horizontal Stabilizer and Elevator</u>:

i. ct = 0.6 * cw = 0.6 ft

ii. ARt = 4

iii. Therefore, bt = 4 * 0.6 ft = 2.4 ft

iv. Elevator, ce / ct = 0.33; therefore, ce = 0.6 * 0.33 = 0.2 ft

v. Taper ratio, $\lambda t = 0.5$

5.4. <u>Vertical Stabilizer and Rudder</u>:

i. cf = 0.6 * cw = 0.6 ft

ii. ARf = 2

iii. b = 2 * 0.6 ft = 1.2 ft

iv. $\lambda f = 0.75$

v. Rudder, cr / cf = 0.42; therefore, cr = .42 * 0.6 ft= 0.25 ft

6. <u>DRAG PREDICTION</u>:

Drag forces are usually measured in the wind tunnel. Before any wind tunnel testes, the drag force can be predicted analytically. Predicting the drag force is important, because it determines the thrust required for flight. The drag force, which the airplane shall experience, comprise of parasite drag, induced drag and interference drag. Analytically the induced drag and the parasite drag are determined, but the interference drag is ignored. The drag generated by the undercarriage, that is the landing gears is also taken into account.

The coefficient of drag is,

$$CD = CDp + k * (CL^2) + (CL^2) / (\pi * AR * u * s) + \Delta CD$$

The first term is the parasite drag contribution due to the boundary layer, the next term is the parasite drag component that varies with the lift coefficient, and the last term is the induced drag contribution. The lift dependent part of drag coefficient can be combined to give,

$$Cdi = (CL^2) * (k + 1 / (\pi * AR * u * s))$$

This equation can be further simplified by using the Oswald efficiency factor e,

$$Cdi = (CL^2) / (\pi * AR * e)$$

where, $e = 1 /((\pi * AR * k) + (1 / (u * s)))$

The Oswald efficiency factor is a measure of the efficiency of the wing planform.

The last term in the equation for the total drag coefficient is the contribution of the undercarriage. With these equations the drag coefficient components are determined.

6.1. Parasite Drag: Parasite drag is determined analytically by this equation,

$$Dp = CDp * q * Sref$$

Where, $$CDp = \Sigma K * Cf * Swet / Sref$$

Total CDp is the sum each airplane component CDp. Cf is the skin friction coefficient, for which a turbulent boundary layer is assumed. The Schlichting empirical formula is used to determine the skin friction coefficient for turbulent boundary layers,

$$Cf = 0.455/(\log RN)$$

where, $$RN = (\rho * V * L) / \mu = (V * L) / \nu$$

Swet is the actual area exposed to the air. Swet for surfaces such as airfoils is usually twice Sref, which is the planform area, multiplied by a correction factor of 2%, due to curvature. K is the correction factor for pressure drag and increased local velocities. It is a function of airfoil or body thickness expressed as a ratio of a maximum thickness, t/c for airfoils, and length / thickness for fuselages. K is an empirical and was obtained from the equation,

$$K = (1 + z *(t/c) + 100 * (t/4)^4)$$

where, $$z = (2 - M^2) / (\sqrt{(1- M^2 * (\cos \Lambda)^2)}) \qquad \partial$$

To determine the Mach number, M, 300 m/s is used for the speed of sound instead of 340 m/s, because Wichita, Kansas might be still be cold in April of 2002.

With these equations the coefficient of parasite drag for all the separate components of the airplane are determined analytically.

6.1.1. Wings: The wing has a rectangular planform area, a combined winspan of eight feet and a chord length of one foot.

K is determined as follows,

$$z = (2 - M^2) / (\sqrt{(1- M^2 * (\cos \Lambda)^2)})$$

with, M = (21 m/s) / (300 m/s) = 0.07

Sweepback angle, $\Lambda = 0$

NACA 6412 thickness ratio, t/c = 0.12

$z = 1.9951 / 0.9977 \cong 2$

therefore, K = 1.26

Cf is determined as follows:

$$Cf = 0.455 / ((\log RN)^{2.58})$$

Reynolds number, RN = (V * L) / ν = ((72.2) * 1) / (0.00016105)) ≅ 430,000

The Reynolds number over the wing is 4.3 * 10^5

Therefore, Cf = 0.0053

Sref = 8 ft * 1 ft = 8 ft^2

Swet = 2 * 8 ft^2 * 1.02 = 16.32 ft^2

Therefore, the parasite drag coefficient contribution of the wing is determined as follows:

CDp = K * Cf * Swet / Sref = 0.014

6.1.2. __Fuselage__: The length of the fuselage is seven feet, its longitudinal width is one foot and its lateral width is nine inches. To determine the parasite drag contribution of the fuselage it is necessary to determine the wet area and reference areas. To determine the wet area, Swet, the fuselage was approximated to be a cylinder of length seven feet and diameter one foot. The reference area is half the value of the wet area. Therefore,

K is determined as follows,

z = (2 – M^2) / (√(1- M^2 * (cos Λ)^2))

with, M = (21 m/s) / (300 m/s) = 0.07

sweepback angle, Λ = 0

length / width = 0.14

z = 1.9951 / 0.9977 ≅ 2

therefore, K = 1.32

Cf is determined as follows:

Cf = 0.455 / ((log RN) ^ 2.58)

Reynolds number, RN = (V * L) / ν = ((72.2) * 7) / (0.00016105)) ≅ 300,000

The Reynolds number over the fuselage is 3.0 * 10^5

Therefore, Cf = 0.0037 ≅ 0.004

Swet = (2 * π * r * L) + (π * r^2) = 47.14 ft^2

Sref = Swet / 2 = 23.57 ft^2

Therefore, the parasite drag coefficient contribution of the fuselage is determined as follows:

CDp = K * Cf * Swet / Sref ≅ 0.01056 = 0.011

6.1.3. __Empennage: tail and fin__: The empennage is the tailplane and the fin, also known as the horizontal and vertical stabilizers respectively. The reference area of the vertical stabilizer is half that of the horizontal stabilizer, if we ignore taper. Furthermore, since the chord lengths of both surfaces are the same, the coefficients determined will be the same for both. This includes the parasite drag coefficient, since the wet area divided by the reference area is the same for both.

K is determined as follows,

z = (2 – M^2) / (√(1- M^2 * (cos Λ)^2))

with, M = (21 m/s) / (300 m/s) = 0.07

Sweepback angle, $\Lambda = 0$

$t/c = 0.09$

the thickness ratio is assumed to be 0.09, because NACA 0009 may be used

$z = 1.9951 / 0.9977 \cong 2$

therefore, $K = 1.20$

Cf is determined as follows:

$Cf = 0.455 / ((\log RN)^{2.58})$

Reynolds number, $RN = (V * L) / \nu = ((72.2) * 0.6) / (0.00016105)) \cong 260{,}000$

The Reynolds number over the wing is $2.6 * 10^5$

Therefore, $Cf = 0.006$

$Swet / Sref = 2.04$

Therefore, the parasite drag coefficient contribution of the wing is determined as follows:

$CDp = K * Cf * Swet / Sref = 0.010$

6.2. <u>Total Parasite Drag Coefficient</u>: The sum of the coefficients determined above:

$\Sigma\, CDp = 0.014 + 0.011 + 0.01 + 0.01 = 0.045$

Therefore, the drag coefficient independent of lift is 0.045.

6.3. <u>Lift Dependent Drag Coefficient</u>: The lift dependent drag coefficient is made up of lift dependent parasite drag due to viscous effects, and the induced drag. These two components combine to give the equation for the coefficient of lift dependent drag,

$$Cdi = (CL^2) * (k + 1 / (\pi * AR * u * s))$$

Where,

CL, is the value at cruise condition, which is 0.7.

k, is a fraction of CDp. For unswept wings, k is determined by the equation [4],

$$k = 0.38 * CDp = 0.38 * 0.045 = 0.017$$

s, is the correction factor for fuselage interference. It is deduced from figure 11.7 of reference 4 by using the ratio of the fuselage diameter to the wingspan. s is deduced to have the value 0.96.

u, is the correction factor applied non-elliptic wing planforms and distribution. For unswept wings, u is about 0.98 to 1.00 [4]. Since the planform of the wing is rectangular, and in general rectangular planforms are not as efficient as tapered planforms and elliptical planforms, 0.98 was assumed to be the value of u.

When the terms in the equation for the lift dependent drag coefficient are combined, the airplane efficiency factor also known as the Oswald efficiency factor e, is obtained.

$$Cdi = (CL^2) / (\pi * AR * e)$$

Where, $e = 1 /((\pi * AR * k) + (1 / (u * s))) \cong 0.67$

Therefore, $Cdi = 0.0291 \cong 0.03$

6.4. Drag of the Landing Gears: An empirical equation for the drag of the landing gears was obtained from reference 5. This equation is used when the dimensions of the landing gears are not yet known. It is an equation that has been derived from a number of civil transport aircraft. Although the airplane is not a civil transport aircraft, taking this equation into consideration serves as an approximation that could be useful in determining the amount of thrust that would be required. The increase of drag coefficient due to the landing gears is approximately,

$$\Delta CD = w * Kuc * m^{(0.215)}$$

Where,

w is the wing loading in N/m^2 = (13.56 kg * 9.8 m/s^2) / (0.743 m^2) = 179.3 N/m^2

Kuc is 5.81 * 10^(-5)

m is the maximum mass of the airplane = 13.56 kg

Therefore,

$$\Delta CD \cong 0.006$$

6.5. Total Drag Coefficient: The total drag coefficient in this analysis comprises the lift independent drag coefficient, the lift dependent drag coefficient and the increase in the drag coefficient due to the landing gears. The estimated total drag coefficient of the airplane is,

$$CD = 0.045 + 0.030 + 0.006 = 0.0756 \cong 0.076$$

6.6. Drag Force at Cruise Condition: The drag force on the airplane at cruise condition is,

$$D = CD * (1/2) * q * V^2$$

$$= 0.0760 * 0.5 * 0.0023081 * (72.2)^2 * 8 = 3.658 \text{ slugs ft s}^{-2} \cong 4 \text{ slugs ft s}^{-2}$$

This is relevant because this number is also the approximated thrust required. A motor, that can give this amount of thrust at cruise condition, will have to be selected. The motor must at least be able to generate twice this thrust the thrust available should be at least eight pounds.

6.7. Lift to Drag Ratio at Cruise: The lift force at cruise should be equal to the airplane weight, but it can be determined using the lift equation,

$$D = CL * (1/2) * q * V^2$$

$$= 0.70 * 0.5 * 0.0023081 * (72.2)^2 * 8 = 3.658 \text{ slugs ft s}^{-2}$$

$$\cong 33.7 \text{ slugs ft s}^{-2}$$

The estimated lift to drag ratio at cruise condition is,

$$L/D_{cruise} = 33.7 / 4 = 8.4$$

7. AERODYNAMIC DESIGN:

The aerodynamic design of the airplane has comprised of obtaining the correct lift coefficients, and predicting the total drag coefficient. The lift coefficients of the wing was derived from the lift coefficient of the airfoil. The induced drag coefficient was determined from the lift coefficient of the wing. The profile drag coefficient obtained from the airfoil analysis is used as the drag coefficient of the wing. By adding the induced drag coefficients to the airfoil profile drag coefficients, the total

drag coefficients of the wing is obtained. These coefficients are used to determine the lift to drag ratios. The aerodynamic design could be summarized as follows:

α	cl	CL	CDi	CDp	Airplane CD	L/D

The values are excluded for brevity.

8. <u>Power Loading Requirements</u>: Power loading is defined as the power divided by the weight of the aircraft. Power can be categorized into power required and power available. The power loading of the loaded aircraft is considered only. Essentially power is the product of thrust and velocity. Although, propeller driven aircraft operate on power rather than thrust, and the custom is to study their performance in terms of power; for power available, the maximum thrust of the motor shall be taken into account, while for power required only drag shall be taken into account.

8.1. <u>Power Required</u>: Power required is the power needed to overcome drag. In order to determine the power required, it is necessary to define the performance in straight and level flight. In straight and level flight the thrust equals the drag force and the lift force equals the weight of the airplane. Therefore the drag encountered in straight and level flight can be referred to as the thrust required. Multiplying drag by velocity gives,

$$Preq = D * V$$

The drag is defined by the following equation,

$D = CD * (0.5 * \rho * V^2 * S) = CDo * (0.5 * \rho * V^2 * S) + (2 * K * W^2) / (\rho * V^2 * S)$

Where, $K = 1/(pi * AR * e) = 0.060$

Therefore power is defined as follows,

$P = CD * (0.5 * \rho * V^3 * S) = CDo * (0.5 * \rho * V^3 * S) + (2 * 0.060 * W^2) / (\rho * V * S)$

The drag polar can be constructed by plotting the value of drag, D versus Speed, V.

8.2. <u>Power Available</u>: The thrust produced by the motor is the thrust available. The power available is the product of this thrust and velocity,

$$Pav = T * V$$

At this stage, the takeoff net thrust can be estimated is to be 20 percent of the takeoff gross weight of the airplane [6]. Drag of the airplane at takeoff is determined by first determining the speed of the airplane at takeoff. The takeoff is speed is 58.8 ft/s. The analysis of this numbers is discussed in greater detail in the detail design report. Also, if it is assumed that the induced drag is approximately zero at zero altitude, then CD is 0.046. Using these values, the drag at takeoff is estimated to be 1.5 pounds. Therefore, the thrust available can be estimated to be:

$$T - D = 0.2 * W$$

Therefore, $T = (0.2 * W) + D$

$= (0.2 * 30) + 1.5$

$= 6 + 1.5 = 7.5$ lb $= 120$ ounces

The power available is the product of 7.5 pounds and the speed, V. The thrust available shall be assumed to be constant with speed.

8.3. Power Loading Curves: The power loading curves for both the power available and the power required are plotted against velocity. The data used to plot the curves were obtained from the aerodynamic design and the program used is a modified MATLAB code obtained from reference eight. The curve is drawn below,

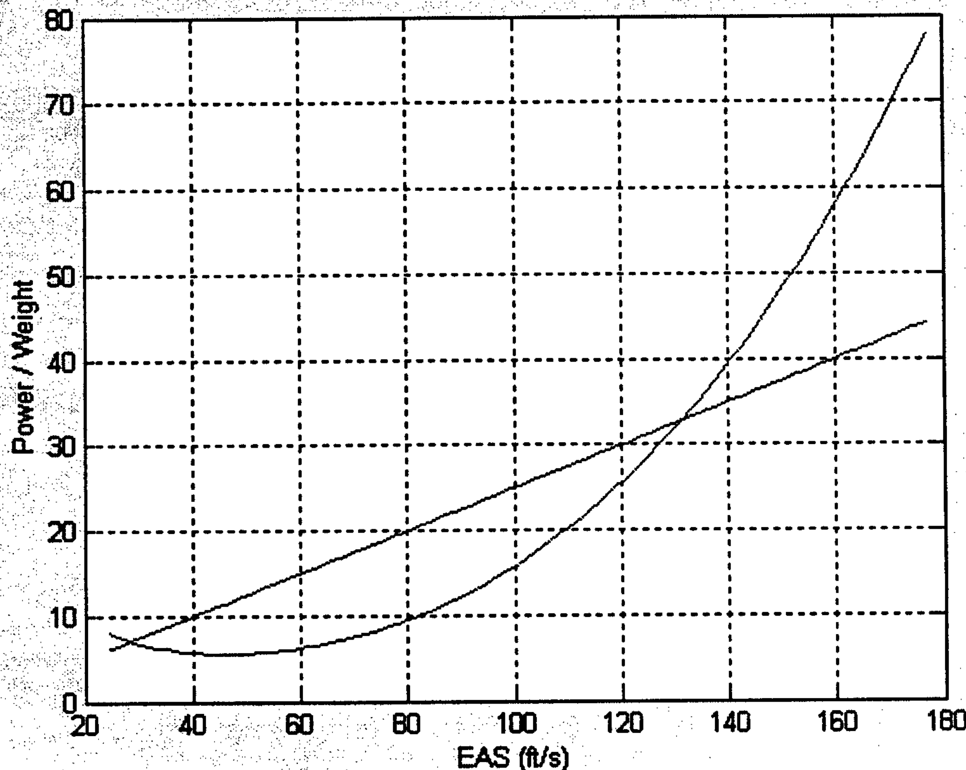


Figure: This shows the power loading curves. P is in units of lb ft/s, and W is in units of lb.

9. Wing loading:

The wing loading of the aircraft for the loaded sorties, is 30 lb divided by 8 square feet, which is 3.75 lb / ft^2. The wing loading for the unloaded sorties is 25 pounds divided by 8 square feet, which is 3.125 lb / ft^2. These are decent wing loading values. The Wright Flyer had a wing loading of 7.3 lb / ft^2, and also a wing loading less than 10 lb / ft^2 for a propeller driven airplane is a good value. An equation for wing loading was obtained from reference 6. A Matlab code was produced a wing loading versus speed curve, using this equation. The equation is contained within the Matlab code below,

MATLAB Function:

```
function Wl=Wl(v)      % wing loading
% density,d = 1.25 kg/m^3, 0.3*d= 0.38
v=10:1:30          % m/s
```

```
for i=1:21
   Wl(i)= 0.38*v(i)^2;        % kg
   Wl(i)= Wl(i)*(0.225/(3.281^2));
   v(i)= v(i)*3.281;
end
plot(v,Wl,'.');
```

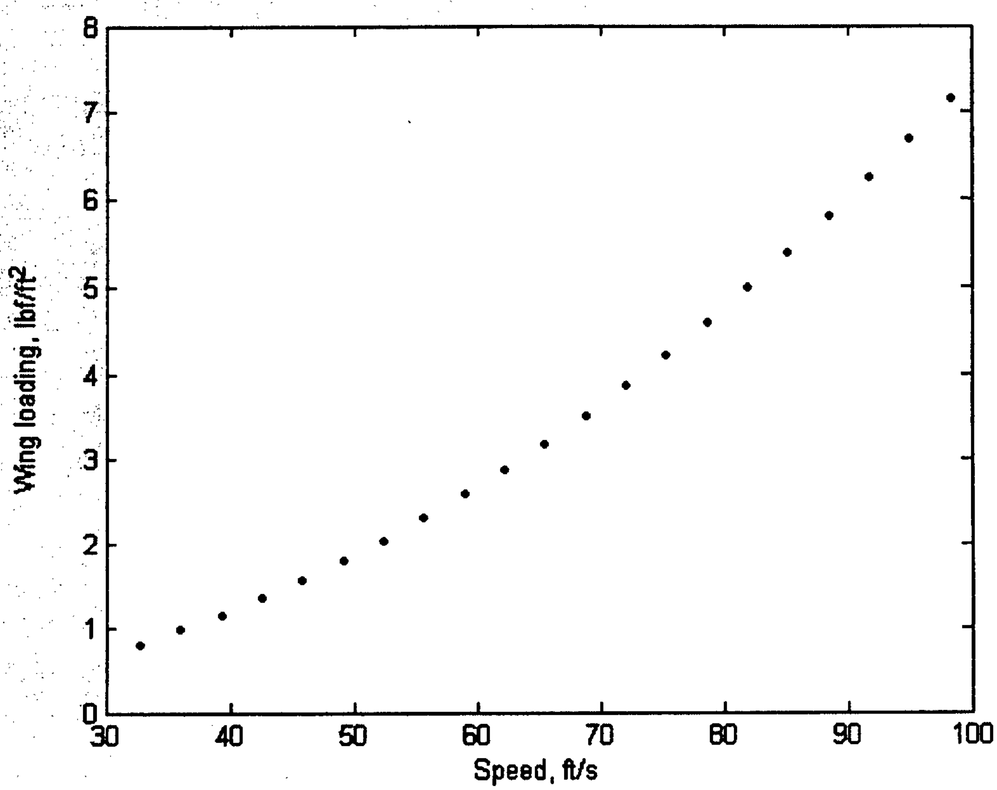


Figure: Wing Loading Versus Velocity Curve.

This curve is an attestation of the effectiveness method used in determining the wing loading of the airplane, which is 3.75 pounds per square feet at 72.2 ft per second, earlier in this design phase.

10. SUMMARY OF KEY FEATURES:

The key features that are produced the final configuration are:

10.1. Airfoil: NACA 6412 was chosen because of its high lift characteristics. The wing lift coefficient at 2 degrees angle of attack met the requirement for the lift coefficient at level flight.

10.2. Wing Area and Wing Loading: The wing area was determined at the beginning of the design to be eight square feet, for manufacturing purposes. This requirement became a limiting factor. Wing loading and the type of airfoil determined were all affected. The wing loading requirement only enables the airplane to carry at most twelve softballs. The wing loading of the loaded airplane is 3.75 pounds per square feet.

DETAIL DESIGN REPORT

1. PERFORMANCE DATA:

1.1. Level Flight Performance: At level flight, the lift force is equal to the weight of the airplane, 30 pounds. From the equation for lift coefficient, the level flight speed can be determined. The lift coefficient at level flight has been determined to be 0.7, therefore the level flight speed at this lift coefficient is 72.2 ft/s. The thrust needed to maintain this speed in level flight is a function of the aircraft weight,

$$T = (CD / CL) * W$$

The CD at level flight has been estimated to be 0.076, therefore the thrust required for level flight at 72.2 ft/s is,

$$T = (0.076 / 0.7) * 30 = 3.26 \approx 3.3 \text{ pounds}$$

1.1.1. Stall Speed: The maximum lift coefficient of the wing is used in determining the stall speed of the airplane. From the aerodynamic design table, Clmax has the value of approximately 1.37.

$$Vs = \sqrt{((2* W) /(\rho * S * CLmax))}$$
$$= \sqrt{((2* 30) /(\rho * S * CLmax))}$$
$$= 48.7 \text{ ft/s}$$

1.1.2. Drag and Thrust Required: For straight and level flight thrust must equal the drag force. Drag is a function of the base drag and the induced drag. The drag of the airplane is found from the drag coefficient, the dynamic pressure and the wing area,

$$D = (CDo + (K * CL\text{^}2)) * (q * S)$$

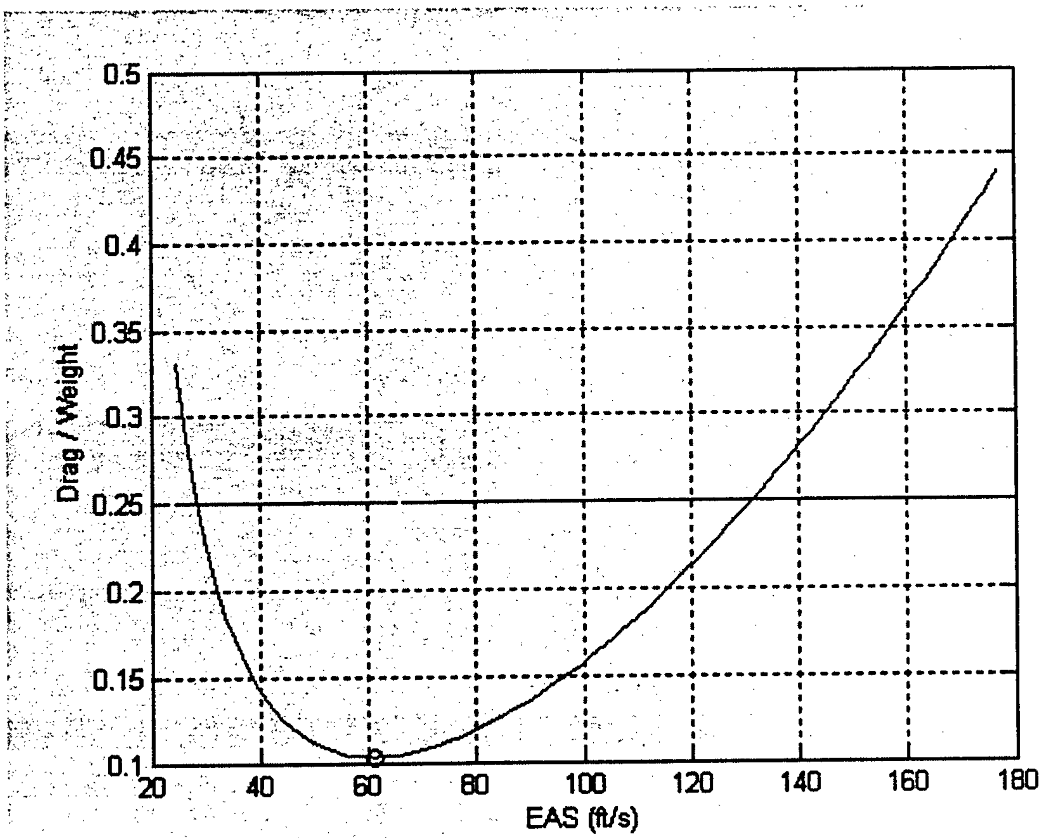
K, is determined from the aerodynamic design, and it is a function of the aspect ratio and the planform shape of the wing,

$$K = 1 / (\pi * AR * e)$$
$$= .060$$

The drag equation as a function of velocity becomes,

$$D = CDo * (0.5 * 0.002381 * V\text{^}2 * 8) + (2 * 0.060 * 30\text{^}2) / (0.0023081 * V\text{^}2 * 8)$$

This equation is the parabolic drag polar which gives the drag variation with velocity of the aircraft in straight and level flight. A Matlab code was used to obtain the parabolic drag polar of the airplane in straight and level flight, which is shown below.



Drag Polar of the airplane at straight and level flight. This figure shows the speed for minimum drag and the constant thrust available curve.

1.2. Climbing Performance

For steady-state constant-speed climb, the rate of climb RC is the vertical component of velocity and can be calculated by

$$RC = V \sin\theta = \frac{V\,(T-D)}{W}$$

θ = angle of climb,

V = constant airspeed,

T = available thrust,

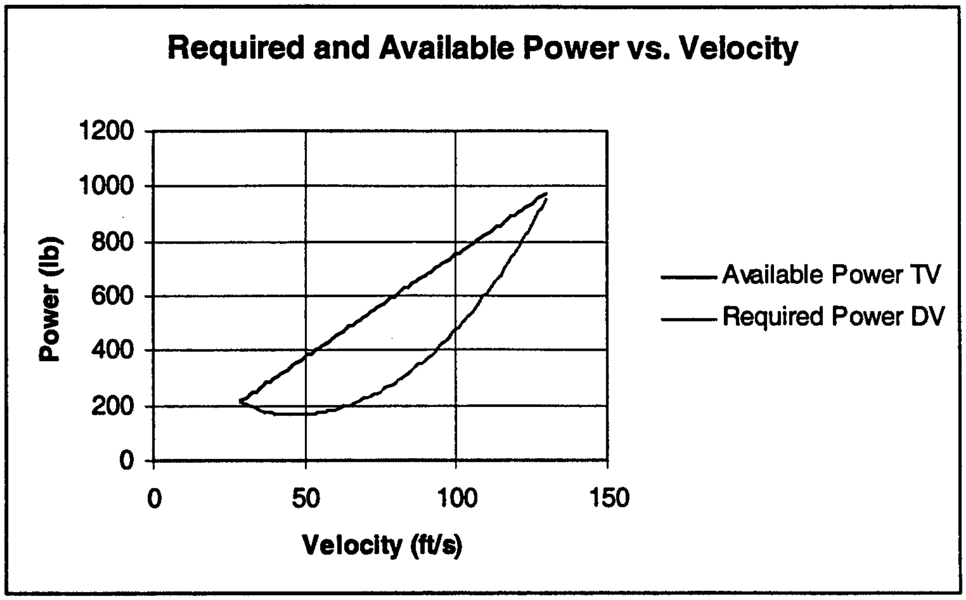
D = drag, or thrust requirement, and

W = total weight of the airplane.

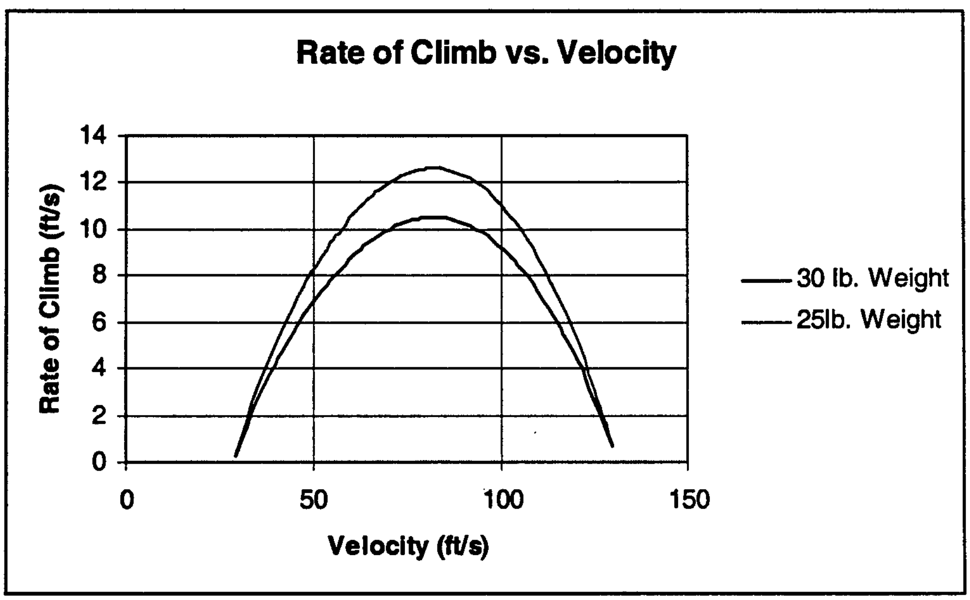
Expressing this relationship in terms of another quantity, the rate of climb is proportional to *excess power.* Excess power, p_{excess}, is defined as the difference between the available power VT and power required for level flight, DV:

$$p_{excess} = VT - DV$$

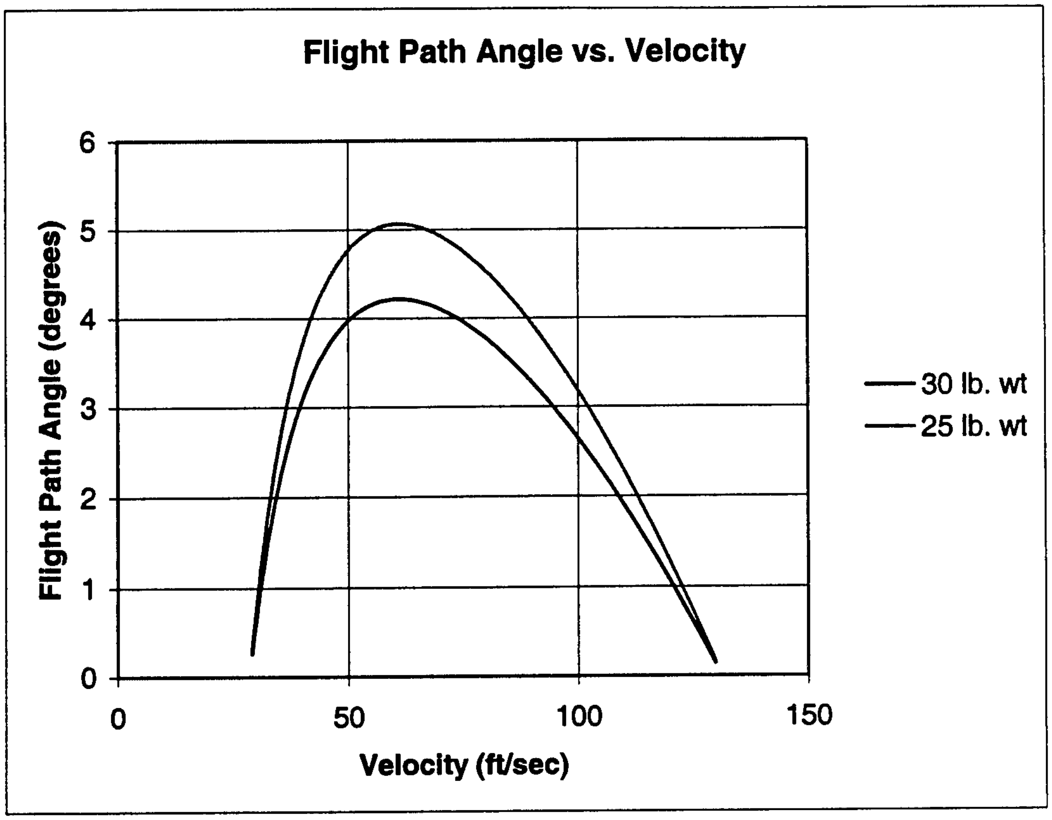
For our airplane, at level flight thrust T = 7.5lb



The distance between the blue plot (available power) and the purple plot (required power) is the excess power; the maximum excess power in our case occurs at around 82.48 ft/s. This is where the maximum rate of climb occurs; this can be seen in the graph of Rate of Climb vs. Velocity.



Also, the maximum flight path angle can be found using the following plot:



Maximum flight path angle occurs at V = 61.09ft/s

2. <u>MISSION PERFORMANCE</u>:

Theoretical Background:

<u>A) Takeoff</u>

During takeoff, the aircraft will experience an additional force, the ground resistance R. R is due to the rolling friction between the airplane's tires and the ground, and is proportional to the normal force:

$$R = \mu (W - L)$$

Our airplane will take off from a smooth, hard runway, so that

$$\mu \approx 0.02$$

Then the sum of forces in the flight path direction becomes

$$F = T - D - R = T - [D + \mu (W - L)]$$

In addition, the induced drag force is reduced during takeoff and landing due to the interaction of the ground (which weakens the wing tip vortices); this is called the *ground effect.*

To account for this phenomenon, we use the following simplification:

a. Calculate D and R at 0.7 V_{LO}, and
b. Assume $V_{LO} = 1.2\ V_{stall}$, where

V_{LO} is the velocity at takeoff.

Next, to obtain the ground roll distance, use the following relations obtained from Newton's second law:

$$S = 0.5\,(a\,t^2)$$

$$S = 0.5\,(F/m)\,(Vm/F)^2 = V^2 m/2F$$

$$S_{LO} = \frac{V_{LO}^2\,(W/g)}{2\{T - [D + \mu\,(W - L)]_{0.7V \text{ at lift off}}\}}$$

B) Landing

In landing, T becomes zero, since the plane will land with the engine idling. Also, D and R are desired to have large values to decelerate the airplane.

Then the sum of forces in the horizontal direction becomes

$$F = -D - R = -[D + \mu\,(W - L)] = m\,(dV/dt)$$

Similarly to takeoff, the ground effect is taken into account by the following assumptions:

a. Calculate D and R at 0.7 V_T, and

b. Assume $V_T = 1.3\,V_{stall}$, where

V_T is the velocity at touchdown.

The distance is

$$S_T = -V_T^2 m/2F$$

$$S_T = \frac{V_T^2\,(W/g)}{2\{T - [D + \mu\,(W - L)]_{0.7V \text{ at lift off}}\}}$$

2.1. <u>Unloaded Airplane Performance</u>:

Takeoff

$$V_{LO} = 1.2\,(48.7\text{ ft/s}) = 58.44\text{ ft/s}$$

$$D\,(0.7\,V_{LO}) = 0.5 * C_D * \rho * (0.7\,V_{LO})^2 * S = 1.174\text{ lb}$$

$$R\,(0.7\,V_{LO}) = \mu\,\{W - [0.5 * C_L * \rho * (0.7\,V_{LO})^2 * S\,]\}$$

$$= 0.02\,\{\,25\text{lb} - [0.5 * 0.9632 * 0.0023081\text{slug/ft}^3 * (0.7*58.44\text{ft/s})^2 * 8\text{ft}^2\,]\}$$

$$=0.2024\text{ lb}$$

$$S_{LO} = \frac{(58.44\text{ft/s})^2\,(25\text{lb}/32.2\text{ft/s}^2)}{2[7.5\text{lb} - (1.174\text{lb} + 0.2024\text{lb})]}$$

$$= 216.5\text{ ft}$$

This is slightly over the rules, but given that some assumptions were made, we think it is acceptable for now. Increasing the thrust or reducing ground resistance will decrease the ground roll.

Landing

$$V_T = 1.3\,(48.7\text{ ft/s}) = 63.31\text{ ft/s}$$

$$D\,(0.7\,V_{LO}) = 0.5 * C_D * \rho * (0.7\,V_{LO})^2 * S = 1.174\text{ lb}$$

$$R\,(0.7\,V_{LO}) = \mu\,\{W - [0.5 * C_L * \rho * (0.7\,V_{LO})^2 * S\,]\}$$

$= 0.02 \{ 25lb - [0.5 * 0.9632 * 0.0023081 slug/ft^3 * (0.7*58.44ft/s)^2 * 8ft^2]\}$

$=0.2024$ lb

$$S_{LO} = \frac{(63.31ft/s)^2 (25lb / 32.2ft/s^2)}{2[7.5lb - (1.174lb + 0.2024lb)]}$$

$= 276.7$ ft

2.2. Loaded Airplane Performance

A) Takeoff

$V_{LO} = 1.2 (48.7 ft/s) = 58.44 ft/s$

$D (0.7 V_{LO}) = 0.5 * C_D * \rho * (0.7 V_{LO})^2 * S = 1.174$ lb

$R (0.7 V_{LO}) = \mu \{W - [0.5 * C_L * \rho * (0.7 V_{LO})^2 * S]\}$

$= 0.02 \{ 30lb - [0.5 * 0.9632 * 0.0023081 slug/ft^3 * (0.7*58.44ft/s)^2 * 8ft^2]\}$

$=0.3024$ lb

$$S_{LO} = \frac{(58.44ft/s)^2 (30lb / 32.2ft/s^2)}{2[7.5lb - (1.174lb + 0.3024lb)]}$$

$= 264.1$ft

Same observations as in Unloaded Plane Performance apply here.

B) Landing

$V_T = 1.3 (48.7 ft/s) = 63.31 ft/s$

$D (0.7 V_{LO}) = 0.5 * C_D * \rho * (0.7 V_{LO})^2 * S = 1.174$ lb

$R (0.7 V_{LO}) = \mu \{W - [0.5 * C_L * \rho * (0.7 V_{LO})^2 * S]\}$

$= 0.02 \{ 30lb - [0.5 * 0.9632 * 0.0023081 slug/ft^3 * (0.7*58.44ft/s)^2 * 8ft^2]\}$

$=0.3024$ lb

$$S_{LO} = \frac{(63.31ft/s)^2 (30lb / 32.2ft/s^2)}{2[7.5lb - (1.174lb + 0.3024lb)]}$$

$= 310$ ft

3. AIRCRAFT WEIGHT AND BALANCE WORKSHEET:

3.1. Aircraft Weight Sheet:

Term / Item	*Pounds*
Empty Weight	25
Payload	5
Basic operating Weight	25
Maximum Operating Weight	30
Maximum Allowable Takeoff Weight	33
Useful Load	8
Landing Gear, nose	0.7
Landing Gear, main	1.3
Motor	1

Propeller	0.3
Battery Pack	5
Speed Controller	0.2
Aileron Servo	0.1
Elevator Servo	0.1
Rudder Servo	0.1
Wings (2)	4
Fuselage	9
Horizontal Stabilizer	1
Vertical Stabilizer	0.5
Miscellaneous, wiring, etc	0.7

3.2. Balance Worksheet:

Balance refers to the location of the center of gravity (c.g.) of an aircraft. It is of primary importance to stability and crash prevention (safety) in flight. The c.g. is the mass center of the airplane and has to be within specific limits for safety purposes. The primary concern of balancing the airplane is longitudinal balance. This is the location of the c.g. along the longitudinal axis. Location of the c.g. on the lateral axis, lateral balance, is also important. Lateral balance of the airplane is easier to achieve than longitudinal balance, because the airplane is designed to be laterally symmetric. Lateral symmetry is whereby for each item of weight that exists on the left of the fuselage centerline there is a corresponding item of weight that exists at the same location to the right of the fuselage centerline. Reference to c.g. will mean the location of the mass balance on the longitudinal axis.

To improve the efficiency of the each flight, the c.g. location of the unloaded airplane is made to coincide with the c.g. of the airplane when it is loaded with softballs. Without this precaution, the airplane could be tail-heavy or nose-heavy. If either of these occur, the pilot will have to make adjustments by control pressure that could reduce the control surface travel, and also reduce the efficiency of each flight.

There are fore and aft limits beyond which the c.g. should not be located for flight. The forward c.g. limit is established at a location determined by the landing characteristics of the airplane. The forward c.g. limit is also restricted to assure that sufficient elevator deflection is available at minimum flight speed of 48 ft/s. The aft c.g. limit is the most rearward position at which the c.g. can be located for the most critical maneuver or operation. The 360 degrees turn is probably the most critical maneuver and operation of the aircraft. If the c.g. moves aft of the aft c.g. limit a less stable condition occurs, which will decrease the ability of the airplane to right itself after this maneuver or after disturbances by gusts. The softballs are adequately restrained, so as not to cause a hazardous condition of the c.g. changing in flight.

3.2.1. <u>The Reference Datum</u>: The datum is an imaginary plane or line from which all measurements of arm are taken. All moment arms and the location of permissible c.g. are taken with reference to the front of the fuselage. The front of the fuselage, is the location of the firewall on which the motor is mounted. The motor is exposed to the atmosphere for cooling purposes and is not housed in the fuselage. In other words, the firewall is in the plane of the datum. The datum is identified as station zero. The each station is measured inches from the datum.

3.2.2. <u>Finding the Center of Gravity</u>: The center of gravity can be found by using an equation obtained from reference 3. To use this equation the airplane shall have to be weighed with three scales. One scale shall be placed underneath the nose wheel, another placed underneath the left main wheel, and the last placed underneath the right main wheel. The landing gear wheels are used as reaction points. The equation used for locating the c.g. for a nose-wheel type airplane with the datum forward of the main wheels is,

$$C.G. = D - ((F * L) / W)$$

Where, C.G., is the distance from the datum to the center of gravity of the airplane

W, is the weight of the aircraft at the time of weighing

D, is the horizontal distance measured from the datum to the main wheel weighing point

L, is the horizontal distance measured from the main wheel weighing point to the nose Weighing point

F, is the weight at the nose weighing point

The airplane must be level when taking readings from the scales. The weight of the airplane, W shall be determined by adding the three different scale readings.

At this time it is not possible to use this equation, but it can still be of use in determining the amount of weight that shall be read at the nose weighing point F. This is because the desired location of the center of gravity is at station 24. This station arises, because it coincides with the quarter-chord point on the mean aerodynamic chord. The mean aerodynamic chord if one foot, and the leading edge of the wing is at station 21, that is two feet from the front of the fuselage. The weight of the aircraft is estimated to be 30 pounds loaded, and 25 pounds unloaded. The main landing gear is located at station 33. The nose landing gear is estimated to located at station 5. Therefore,

C.G. = 24 inches

W = 30 pounds, and 28 pounds

D = 33 inches

L = 33 – 5 = 28 inches

Therefore, F = (D - C.G.) * (W / L)

= (33 - 24) * (30 / 28) = (33-24) * (25 / 28)

= 9.6 pounds, loaded = 8.0 pounds, unloaded

To achieve these target readings from the scale underneath the front wheel is maybe necessary to move systems and items within the fuselage accordingly. Also, 9.6 pounds and 8.0 pounds must also be attained with the same system and component configuration / architecture within the fuselage for the loaded and unloaded airplane, respectively. If these values of F are attained, the center of gravity in percent of mean aerodynamic chord is 25%. If it becomes necessary, the airplane is designed to carry three extra pounds, which could be used for ballast. F may also be used to define forward and rear c.g. limits.

There is another method of determining the c.g. location in aircraft. In this method, the c.g. is determined by the total moments by the total weight. The total moments are obtained by multiplying the weight of each item of the airplane by its corresponding moment arm and then summing them up. An item located ahead of the datum has a negative arm and vice-versa. This method is more complicated than the previous method, because of the multiplication steps. Also, it is not convenient at this time, because the airplane is under construction. Using our current estimates the c.g. can be partially determined using this method,

Item	***Weight (lb)***	***Arm***	***Moment***
Nose landing gear	0.7	5	3.5
Main landing gear	1.3	34.5	44.85
Motor + Propeller	1.3	-3	-3.9
Payload	5	24	120
Battery Pack	5	24	120
Wings	4	4.5	18
Empennage	1.5	80.5	120.75
Total	18.8	Not necessary	423.5

Therefore, c.g = 423.5 / 18.8 = 22.5 inches, behind the datum.

The target location of the c.g. is 27 inches behind the datum. This method shows that our assumptions are on the right track, because not all of the aircraft was considered in the table above. This result of this method will aid in the determination of the location of the items that where not considered above, including the mass concentration of the fuselage shell. For instance, assuming that the rest of the weight unconsidered above is the fuselage, the center of mass of the fuselage shell can be determined as follows,

c.g. of fuselage shell = (423.5 + (xf * 9)) / (18.8 + 9) = 27

therefore, xf = ((27 * 27.8) – 423.5) / 9 = 36.34 inches aft of the datum

3 ft behind the datum, is a decent approximation for the center of mass of the fuselage shell. It may make sense to actually make this the center of mass of the fuselage shell, because the weight considered here is 28.8 pounds, the loaded airplane weight is 30 pounds and the weight of the unloaded airplane is 25 pounds. The average weight of the loaded and unloaded airplane is 27.5 pounds, which is close to 28.8. The unconsidered weights would just be placed accordingly.

This is just an example of how the second method can be applied to meet the target c.g. location of the entire loaded and unloaded airplane.

4. Component Selection:

4.1. Landing Gear Design:

Aim: To design a landing gear which will give the airplane a leverage of 20 inches above the ground

Diameter of propeller: 24 inches

Radius of propeller: 12 inches

Allowable ground clearance: 6 inches (between the tip of the propeller and the ground)

Diameter of fuselage: 0.75 feet

Description of landing gear: The landing gear is shaped like a trapezoid, and the angle made between the perpendicular line drawn at the intersection between the sides and the smaller base is 22.5deg. The angle between the edge of the fuselage and the aluminum section connected to the wheel is 22.5deg.

Calculation of dimensions:

We need a leverage of 20 inches above the ground, therefore the height of the trapezoid will be 20 inches. Knowing the angle made between the perpendicular line drawn between sides and the smaller base we can then calculate the length of the sides. We therefore have:

$$\cos(22.5) = 20/d1$$

Where, d1 = length of side

Therefore $d1 = 20/\cos(22.5) = 21.65$ in

Converting now the diameter of the fuselage from feet to inches we have:

$$0.75ft = 0.75ft \times 12in/1ft = 9in$$

The total length of the aluminum bar is therefore:

$$(21.65)*2 + 9 = 52.3in$$

To calculate the length of the larger base of the trapezoid (the distance between the tires), we first need to find the distance between the tire and where the perpendicular line drawn to calculate the length of the side meets the larger base. We therefore have:

$$\text{Tan}(22.5) = d2/20$$

Therefore $d2 = 20\text{Tan}(22.5) = 8.28in$

The total length between the two tires is therefore: $(8.28)*2 + 9 = 25.56in$

4.2. Power Package:

In the analysis of the aircraft, we have managed to learn how to do the calculations to pick the right engine, propeller and batteries. By picking the right engine, we have enough power to rotate the propeller, thus generate the necessary advance velocity. There are several approaches, which can be used to make the calculations so that we pick the right engine.

Calculating the Current and Voltage

One of the approaches that we used is called the Keith Shaw's rule of thumb. This rule is used to make the calculation of the current and the voltage for the engine.

Here we propose to find the current I, the voltage V, the number of cells necessary to power the airplane. We assume that the Iron lost is about 3 Amp, the Copper resistance is 0.069 Ohms, and the engine constant will be about 1488rpm/volts.

1) We know Plcu = I^2 *0.069 and Plfe = V*i1 i1 = 3Amp

2) Plcu = Plfe, so

I^2 *0.069 = 3*V (equation 1)

3) But we need 40 watts/lb, and we know that the plane will be no more than 30 pounds. Therefore,

40*30 = V*I or V = 1200/I (equation 2)

4) Put (eq 2) into (eq 1), we have

0.069*I^2 = 3*(1200/I)

I^3 = 3600/0.069

I = 37.37 Amp and V = 32.11 volts

The values above will be the characteristics of the motor we have to pick. The motor selected should be able to operate between 19 and 37 Amps.

The voltage calculated is the necessary voltage needed to power the engine fully. If we put less of this voltage, then we will get less angular rotation output. Since the voltage has to be provided by batteries and since batteries deliver a certain amount of voltage per cell basis, we can divide this voltage by the amount of voltage delivered per cell to find the number of battery cells needed. Once we have the numbers of cells needed, we need then to calculate the total weight of the batteries. If we know the mass of one cell, the total weight of the batteries will be:

Weight = # of cells * mass of one cell.

If the total weight is no more than five pounds, then we can move on to the next calculations. If not, we have to select a lower voltage and a more efficient engine. And the engine should be able to operate within the current limit stated above.

Selecting the Batteries

To select the batteries, we needed to calculate their milliamp hour characteristic, which is the rate at which the cells run. Since we needed only half throttle for level flight, dividing the power by two was done. Now the current calculated above for the engine is twice the current needed from the batteries. So we divide it by two. Then we proceed to calculate the typical mah of each cell.

5) Having I = 37.37 Amp, we divide by 2 and obtain

I1 = 18.69 Amp

Targeting a 10 minutes flight will give us

I2 = 18.69*10 = 186.9 Amp minutes

If we write this per hour we have

$$I3 = 186.9*1hr/60minutes \quad = \; 3.115 \text{ Amp hour}$$

If we want to express the value in milliamp hour we write

$$I4 = 3.115*1000 \; = \mathbf{3115\ mah} \text{ (milliampere hour)}$$

Having found 3115 mah for the cells, we conclude that a typical 3200 mah cell is sufficient. In a catalog that we looked at, we see that the cell delivers 1.2 constant volts, which is good because we expected to get cell design that delivers only 1 volt per cell. We also found that the weight of each cell is about 0.176 pound. Then we calculate the number of cells needed now:

cells = 32 volts/1.2 volt = 27 cells

So we calculate the total weight of the numbers of cell used to power the engine:

Total weight = 27*0.176 = **4.75 lbs,**

Which is less than 5 pounds.

Selecting the Propeller

Choosing the propeller is independent of the engine, although they will be related either directly or by a set of gears.

Since it would take too much time to design a propeller, we open a catalog and select a propeller that will give at least 120 ounces of thrust to deal with the drag, and absorb about 900 watts, which is (0.75*1200 watts) available from the engine. The propeller is designed to run at a certain rpm and its rpm value may not be equal to that of the motor. For example, we need to use a set of gears to step the speed up if the rpm is of the engine is less and a set of gears to step it down if the speed of the engine is more.

Conclusion

With these specifications the Astroflight Cobalt 60 motor with Model 714 Super Box is selected. The propeller shall be 24 inches in diameter, and the pitch shall have a value of 10.

4.3. Systems Architecture: The landing gear goes right behind the wing underneath the fuselage. The motor is mounted on its mount, which is connected to the nose of the fuselge. Three servos are employed for the control of the aircraft. The rudder servo is connected to the front landing gear to give directional control on the ground. The batteries are placed in between the wings, underneath the payload mount. A battery door will enable accessibility to the batteries before each flight period. Some of the other components include electronic speed controller and fuses.

5. DISCUSSION ON SOME ASSUMPTIONS:

5.1. The c.g location was approximated by two separate methods. The first method sort to define a requirement that would ensure the assumed location of the c.g. The second method used some of the estimated parameters in to first method to showed that the estimated c.g. location is plausible.

5.2. Location of nose landing gear, 3.22: for a nose wheel type aircraft drawn to scale in reference 3, the nose landing gear is located at 0.25 inches from the front of the fuselage. The fuselage is 3.25 feet long. The ratio of the 0.25 and 3.25, gives approximately 0.077. If this number is multiplied by length of our airplane, which is 7 ft, the result is 0.54 feet. This means that the location of the nose wheel on our aircraft is about the same as the location of the nose wheel on any nose wheel type aircraft. To further justify this length, the distance from the main wheel to the tip of the tail on the drawn aircraft is 2.5 inches. The same distance on our aircraft is 4 feet. If 2.5 divided by 3.25, 4 divided by this result, the result is now is about 0.742. 0.742 multiplied by 0.54 gives 0.4 ft or 4.8 inches, which is approximately 5 inches. 5 inches is the distance from the nose of our aircraft to the nose landing gear. In essence, the drawing of a nose wheel aircraft [3] is a scaled up down version of our airplane in terms of landing gear distances. This method of scaling validates the assumptions made for the landing gear distances.

6. <u>Handling Qualities, Stability and Control</u>:

In designing our airplane we had to make sure that the center of gravity of the aircraft is in the right place along the wing. Once this is done then we worry about longitudinal stability, Lateral stability and Directional stability for better maneuverability.

6.1. <u>Longitudinal Stability</u>: Longitudinal stability has to do with how stable the airplane is in pitching. If the mean aerodynamic chord has an increasing nose down pitching moment with increasing angle of attack then it is said to be longitudinally stable. The chord length of our wing is 1 foot and in setting the dimensions for the fuselage we took in to consideration the length forward and aft of the wing. One of the problems we had last year was that the length of the fuselage, forward of the wing, was too short (about 6 or 7 inches) while the length aft the wing was about 5 feet. Our center of gravity turned out to be too far back on the cord length of the wing when the plane was built (with the batteries and motor installed). We were forced to add a weight in the fuselage at the front to bring forward the center of gravity (cg) to the correct place but that didn't work and the plane turned out to be too heavy. For this year's plane, the fuselage has 1.75 feet forward of the wing, which that would give us more moment arm about the cg at the front to play with, and 4.2 feet aft of the wing. For the aft dimension, we took into consideration the tail moment arm so that if we were to decrease it we would have to increase the size of the tail i.e. *this was done on the Airbus A330-200 when the fuselage of the A330-300 was shortened both forward and aft of the wing and it was given a larger tail to compensate for the tail moment arm.*

Other considerations were: the engine thrust pitching moment, the tail and elevator drag moments and the cg of the fuselage with respect to the cg of the wing. The thrust moment arm is not that far from the axis of the wing chord. Since the airplane is a low wing configuration, the thrust moment arm will produce a nose down moment, which is stabilizing. The elevators moment arm is a little bit higher than that of the engine thrusts and it produces a nose up moment but it is

not too great to make the aircraft unstable. The elevators are controlled by one servo. The cg of the fuselage is not too far above from the cg of the wing.

6.2. Lateral Stability: Lateral stability has to do with how stable an aircraft is about its Roll Axis, which runs from the nose of the plane through to the back of the plane. Even though the airplane is a low wing aircraft, the wings do not have dihedral. We took into account that with a wingspan of 8 feet, the wings will have enough deflection upwards to give us a nice dihedral angle when the aircraft is in flight and this would help keep the aircraft stable laterally. The airplane only has ailerons on the main wings and they have a sufficient size to give the aircraft a good rolling moment for good maneuverability considering the windy conditions the plane will be flown in. The ailerons also have a pretty good deflection angle and they are connected to one servo. They are located at near the tips of the wings to maximize the moment arm. There are no flaps or spoilers on the airplane.

All the payload and batteries and servos etc. are distributed evenly about the roll axis and placed near the belly of the plane, below the center of gravity of the fuselage. This would help to make the plane as stable as possible along with the dihedral angle of the wing.

6.3. Directional Stability: Directional Stability concerns movements about the yaw or vertical axis. The tail has a rudder of sufficient size to give good moment about the vertical axis. Since the aircraft has to be flown in windy conditions and since it has to make a sharp 360-degree turn, a good yaw moment is necessary to give good maneuverability and to help maintain a straight glide path and sideslip angles in flight or more importantly in landing especially in crosswinds. The rudder and nose gear are connected to the same servos on our airplane.

7. MORE PERFORMANCE PREDICTIONS:

With the data from the aerodynamic design it is possible to predict certain performance criteria that pertain to maximum lift to drag ratio, and minimum power. Endurance is the focus of minimum power related values, and range is the focus of maximum lift to drag ratio related values.

7.1. CL for (L/D)max: A major design objective in airplane designs is to minimize drag. For minimum drag, the lift coefficient is the coefficient for which the drag due to lift is equal to the parasite drag. Therefore,

CL for (L/D)max = (CDp * pi * AR * e) = (0.045 * pi * 8 * 0.67) ≅ 0.87

7.2. Maximum Lift to Drag Ratio:

(L/D)max = (√ (pi) / 2) * b * √(e) / √(CDp * S) = (√ (pi) / 2) * 8 * √(0.67) / √(0.045 * 8) = 9.7

7.3. Speed Corresponding to CL for (L/D) max: To obtain minimum drag in flight the airplane must fly at the speed corresponding to the CL for (L/D)max. Therefore,

V for (L/D)max = √((2 * W) / √ (CDp * pi * AR * e)) = √((2 * 30) / √ (0.45 * pi * 8 * 0.6)) = 68.4 ft/s.

This speed is more important for jet-powered airplanes than propeller airplanes. It is almost always important to design and operate airplanes somewhat off the optimum, because very little

is lost and some other characteristics are gained or improved. In process of design of this airplane, the speed at which the airplane shall operate is off the optimum speed above. Flying at 72.2 ft/s second rather than 68.4 ft/s increases the range of the aircraft and will reduce its flying time per lap.

7.4. Thrust Horsepower Required for level Flight: The thrust horsepower required for level flight is determined for the estimated cruise speed of 72.2 ft/s and the cruise speed for the maximum lift to drag ratio. The value of drag used in both calculations is the value obtained in the aerodynamic design of the airplane for level flight.

7.4.1. For Estimated Cruise Speed: Practically, in order to meet the requirements specified in the aerodynamic design for level flight, the airplane will fly at the estimated cruise speed of,

V = 72.2 ft/s

Therefore, thp = (T * D) / 550 = (D * V) / 550 = (4 * 72.2) / 550 = 0.53 hp

7.4.2. For Speed for (L/D)max, 64.8 ft/s: Ideally, propeller powered aircraft obtain best specific range at the speed corresponding to the maximum lift to drag ratio,

V = 68.4 ft/s

Therefore, thp = (T * D) / 550 = (D * V) / 550 = (4 * 68.4) / 550 = 0.53 hp

7.5. Minimum Power Estimates: Performance data for minimum power is important, because ideal maximum endurance will occur at values pertaining to minimum power.

7.5.1. Lift Coefficient for Minimum Power: This is the lift coefficient at which the airplane would fly at steady level flight, so as to maximize endurance.

CLmp = $\sqrt{3}$ * (CL for (L/D)max) = $\sqrt{3}$ * 0.87 = 1.5

7.5.2. Speed for Minimum Power: Maximum endurance may occur at this speed. The speed for minimum power is less than the speed for (L/D) max.

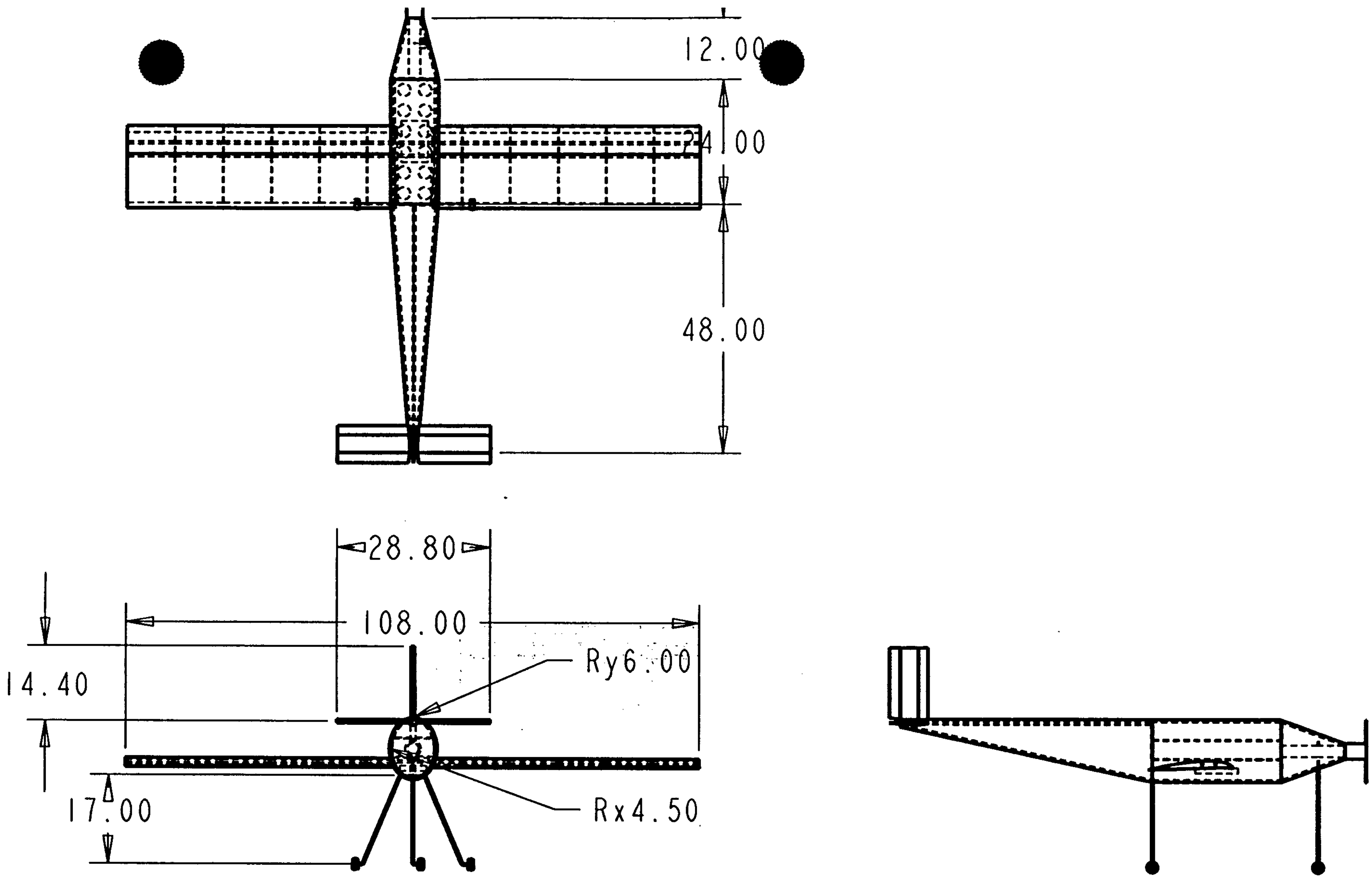
Vmp = 0.76 * V for (L/D)max = 0.76 * 64.8 = 49.3 ft/s

7.5.2. Speed for Minimum Power: Long-range endurance airplanes are designed for minimum power. In this competition aircraft are designed for ten minutes endurance, which mainly depends on the propeller, motor and cell selections.

thpmin = (1.052/550) * √((W^3) / (ρ * S)) * (CDp / (AR * e)^3)^(1/4) = 0.36 hp

7.6 Maneuvering Performance: the load factor, n = (L/D) = sec ø. The stall speed at bank angle ø, Vsø = Vs√(n). Thus stall speed increases in a turn.

Bank Angle, ø (degs)	n	% incr. in Ind. Drag., Levl Flight	Vsø
0	1.0	0	49
15	1.036	7.2	49.8742
30	1.154	33.3	52.63795
45	1.414	100.0	58.26675
60	2.000	300.0	69.29646


12.00
24.00
48.00
28.80
108.00
14.40
Ry6.00
17.00
Rx4.50

<u>Rated Aircraft Cost</u>, $ (Thousands) = (A*MEW + B*REP + C*MFHR)/1000

Coef.	Description	Value
A	Manufacturers Empty Weight Multiplier	$100 ~~* Material Multiplier~~
B	Rated Engine Power Multiplier	$1500
C	Manufacturing Cost Multiplier	$20 / hour
MEW	Manufacturers Empty Weight 25	Actual airframe weight, lb., with all flight and propulsion batteries but without any payload **Material Multiplier:*DELETED***
REP	Rated Engine Power (1+(1-1))*5 = 5	(1+.25*(# engines-1)) * ***Total Battery Weight "Total Battery Weight"*** will be the weight of the propulsion battery pack(s) as determined by the judge's scale during technical inspection. Total propulsion battery pack weight may not exceed 5 lbs., but may be lighter.
MFHR	Manufacturing Man Hours 8*8=64 3*0 = 0 10*7 = 70 5*0 = 0 10*1 = 10	Prescribed assembly hours by WBS (Work Breakdown Structure). MFHR = □ WBS hours **WBS 1.0 Wing(s):** 8 hr/ft. Wing Span 8 hr/ft Max exposed wing chord (measured at the point on the wing(s) where the chord is greatest) 3 hr/control surface Sum values for multiple wings **WBS 2.0 Fuselage** 10 hr/ft body maximum length Note: Maximum length of the body is defined to be the longest longitudinal length possible

	10*2 = 20 0 5*3 = 15 5*1 = 5 5*1 = 5	to measure on the aircraft, no mater what physical elements it is composed of **WBS 3.0 Empennage** 5 hr./Vertical Surface (Any vertical surface, including winglets, struts, end plates, ventrals etc) with no active control 10 hr/Vertical Surface (Any vertical surface) with an active control 10 hr./Horizontal Surface. A horizontal surface is a "wing" if it is more than 25% of the span of the greatest span horizontal surface.) A "V" tail is considered to be a Vertical surface without control (5 hr) plus a horizontal surface with controls (10 hr), for a total of 15 hrs. **WBS 4.0 Flight Systems** 5 hr./servo or motor controller **WBS 5.0 Propulsion Systems** 5 hr./engine 5 hr./propeller or fan

MANUFACTURING PLAN

Various materials were considered for the construction of our aircraft. The weight, price and availability of different materials were considered before the decision was made.

The team decided to construct the aircraft out of wood since we had previous experience with this material. Wood is also within our budget limitations and has a considerably low weight compared to other materials. An option we had was to construct the airplane out of composite materials, but since we had no previous familiarity with its handling and the its price is very high we decided to decline the idea.

We used diverse types of the wood for the body of the aircraft, balsa and spruce primarily. We used pine spruces for the construction of the wing frame. The airfoils of the wing were engendered out of balsa wood, so were the ribs used to create the body of the fuselage. The outside sheeting of both the wing and the fuselage were also made out of balsa wood.

Aluminum alloy rods were used to mate the wing with the fuselage, and also to connect the stabilizer. The aluminum offers strength with out compromising the weight and maneuverability considerations.

Figures of Merit

Cost, Ease of construction, Durability, Past experience, Availability of materials, and Weight

The fund raising for this project began last year. Students from last year's team began to seek for corporate donations. Other fund raising efforts came from the executive board of the AIAA student chapter at the college. We acquired necessary funds to build a modest airplane. Different stores and hobby shops, and also different kind of wood were researched in order to obtain the best price possible. The final price of the total amount of wood used was $300. The tools used to form the wood into the required shapes were owned by team members. We also made use of the mechanical engineering machine shop.

Wood was our choice due to price and ease of handling. Construction of an aircraft made out of wood is relatively simple, cutting and pasting. No advanced skill is necessary to work with wood. Any imperfections or inaccuracies in the in the cutting or molding of wood can be easily corrected. This flexibility makes it easier for last minute changes on the structure to take place. Other materials such as carbon fiber or fiberglass can be very difficult to mold or shape.

Wood's ease of handling also accounts for a great disadvantage. It can easily rupture in the event of a crash. The entire parts made out of wood could be completely destroyed. Entire sections of the plane may need to be replaced depending on the kind and place of impact. The impact loads of the crash would be distributed throughout the total body of the aircraft. Anticipating any unfortunates we made an additional wing and outside fuselage structures. We can easily attach any new structures to the aircraft without severe time loss.

All the team members have previously worked with wood. It was very easy for the whole team to picture the frame design since the design process began. All the team members easily visualized the final frame design since knowledge of how the loads are transferred from one section to another had been previously acquired. Three of the team members were on last year's DBF competition and had a vast experience on wooden frames since last year's aircraft was also constructed from wood.

Different wooden modeling parts and tools for the construction are easily found in any hobby shop and we can also make use of the mechanical engineering wood shop. The wood is also available from different companies on the internet. The types of wood chosen for the construction control the weight of the airframe. The adhesives used to bind the wood together also account for the weight of the airframe.

All the specifications about wood established above make it a perfect choice for the construction of the aircraft. It is the most suitable material to use with all of our limitations and constraints.

Manufacturing Process:

All the components of the aircraft were manufactured simultaneously in order to save time. The components also need to be manufactured simultaneously so that they can be put together. As each component was manufactured it was tested for strength (as needed) and compatibility between each other.

Wing Spar Construction

The use of four struts or spars of the following dimensions $\frac{1}{2} in X \frac{1}{4} in \; X\, 48$in. was the main idea behind the design of the wing structure. Two of the struts were placed on top of the airfoil while the remaining two struts were placed on the bottom of the airfoil. Making this kind of construction gave us the option of using the space left between the top and the bottom struts to form an I-beam for a better structural support. The struts were also covered with carbon fiber tape at the $\frac{1}{4} in$ width of the struts to provide higher strength. For the construction of the I-beam we used struts of the same width and thickness, but with a length of 4in, which is equal to the space between each airfoil. Epoxy resins were used to bind the elements of the wing structure.

Wing Construction

The airfoil was chosen by the design team to be of the NACA four digit series (naca 6412). The wing span was decided to be 9ft. divided into three parts, an 8ft. wing divided equally in two parts, each side of the wing, right and left, and 1ft. for the width of the fuselage. A jig was built to ease the manufacture of the wing. The purpose of the jig was to keep each airfoil straight while the manufacturing of the wing was performed. The ribs were cut using a special mold of the naca 6412 made out of plywood. The mold was made in order to craft all the ribs on the wing without any variations.

The airfoils were cut nearly perpendicular to the cord in order to accommodate the needed structural continuity of the wing spar, so that when they were glued to the spar, the desired airfoil shape was still obtained. The four spars used are made out of pine wood spruce measuring $\frac{1"}{2} X \frac{1"}{4} \; X\, 48$ in. A leading edge shaped piece of balsa wood was obtained for front edge of each wing. The maximum diameter of the leading edge is 1in. and the length is 36in.

The area from the leading edge to the quarter chord was then sheeted with 0.0625in thick balsa wood. The flaperons made up 20% of the length of the chord. They were built at the same time as the wings and were attached using cyanoacrylate hinges. Stabilizer Construction

The horizontal and vertical stabilizers were built at the same time to ensure precise cohesion between the two parts. Both stabilizers were built from $\frac{3}{16}in.$ thick balsa wood. The specifications are as follows:

Horizontal Stabilizer and Elevator:

i. chord length = 0.6 ft

ii. span = 2.4 ft

iv. elevator chord = 0.2 ft

v. taper ratio, lt = 0.5

Vertical Stabilizer and Rudder:

i. chord length = 0.6 ft

ii. span = 1.2 ft

iv. taper ratio, lf = 0.75

v. Rudder chord = 0.25 ft

Each piece was covered with heat shrink-fabric applied to the outside, the elevator and the rudders were also attached with cyanoacrylate hinges, the same way as the flaperons.

Fuselage Construction

Pro-Engineer 2000i[2] was extensively used to visualize the construction of the fuselage in three dimensions. The use of this software served as a helpful tool in the manufacturing of the fuselage. With 2-D printouts of each of the three sections of the fuselage we were able to cut out the mold of the fuselage frame, including the ribs of the frame.

In order to easily transport the airplane, the fuselage and the wing were broken up into different sections.

The fuselage is in divided into three sections, the nose cone, the center section, and the tail section.

The specifications for each part are as follows:

i. the length of the nose section is 1 ft

ii. the center section is 2 ft

iii. the tail section is 4 ft

iv. the front of the nose section is 3 inches in diameter

v. the leading edge of the wing should be attached to the fuselage at 1.75 ft from the nose of the airplane. (that is it should be attached 0.75 ft from the front of the center section)

vi. The lateral cross section is 1 ft from top to bottom, and 0.75 ft (9 inches) from side to side.

vii. The total length of fuselage is 7 ft.

For the center section we used aluminum alloy square bars as the main support for the total gross weight of the aircraft. The center section of these bars hold the landing gear, the servos, and the wing. The center frame has circular ribs which were cut and glued to each side

forming the circular shape of the fuselage with a low wing. After the center frame was built, the two remaining sections were attached and glued.

The circular cross-section was covered with thin balsa and then covered with heat-shrink fabric, leaving an opening on the top for the softballs to be introduced. The idea behind the circular ribs was to obtain a maximum cargo space. Two thin sheets of balsa were attached to the inside of the circular section. These sheets have cut out circular pieces in which the softballs tightly fit.

In the front section of the fuselage a motor mount was placed as well as a support structure for the nose gear. The nose section was constructed using balsa ribs in a decreasing manner in order to obtain an aerodynamic cone like shape. This section was strongly reinforced with epoxy resins to ensure the stability of the motor.

Joiner Construction

Due to the measurements of the whole aircraft the there is a constraint to transport it to the flight location. For this reason we needed to divided the aircraft into various sections. The wing must be attached to each side of the bottom of the fuselage. In order for this to be accomplished, the use of a joiner structure was required.

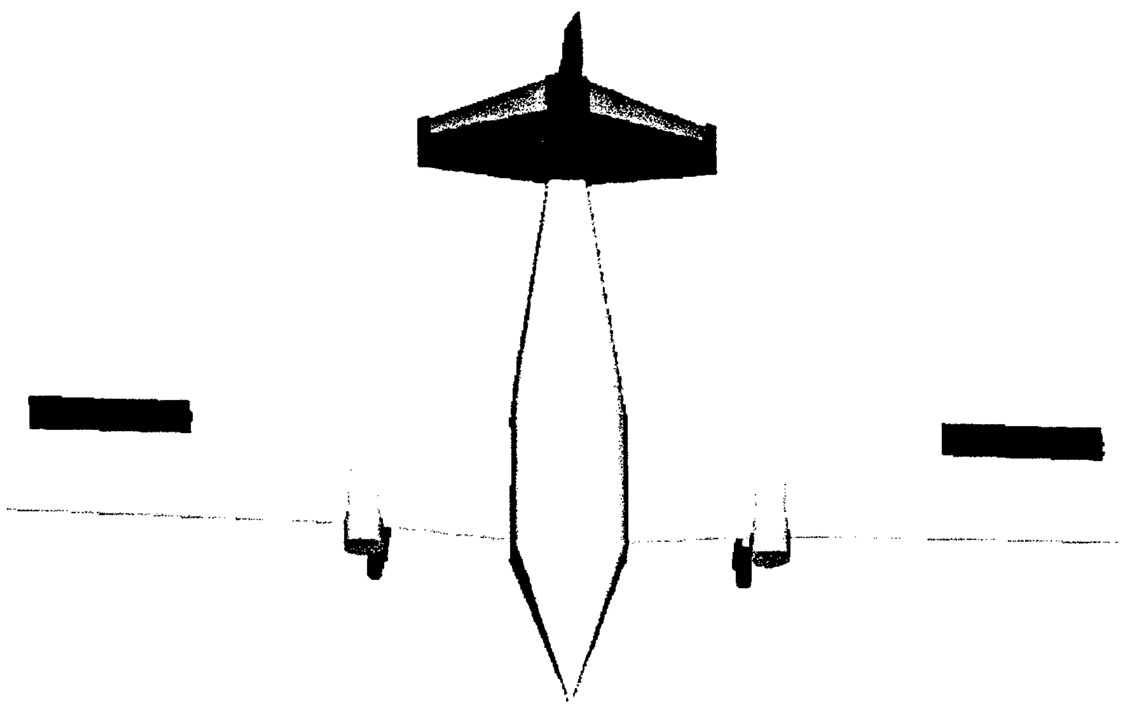
A hole of ¾ in. diameter was drilled towards the front part of each of the last three ribs on each wing. The purpose of this hole is to pass a hollow aluminum alloy rod that is connected to the fuselage in order to mate both structures. The same procedure is done on the back end of the airfoils, but with a smaller diameter rod, this will stabilize the wing.

Landing Gears and Motor Mount

Alloy	Tensile Strength	Yield Strength	Percent Elongation	HB	Notes
2014	27,000 psi	14,000 psi	18%	45	2XXX series aluminum is primarily alloyed with copper.
6061	18,000 psi	8,000 psi	25%	30	6XXX series aluminum is primarily alloyed with magnesium and silicon.

Two alloys of Aluminum were considered for the fabrication of the landing gear and motor mount. Aluminum was selected as the base material, because of its high strength to weight ratio. The tensile strength of the material is the amount of stress that it can take before complete failure, and the yield strength is the amount of stress before the material is permanently deformed. Percent elongation is the amount of deformation that the material undergoes before it breaks and the HB is a measure of the hardness. From the chart it can be seen that 2014 is stronger than 6061. As a result the 2014 aluminum alloy was selected for the fabrication of the the motor mount and the landing gears.

Syracuse University
Design Build Fly Competition
2001/2002



ACKNOLEDGEMENTS

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Dr. Higuchi
Dr. Murthy
Dr. LaGraff
Dr. Davidson

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ABSTRACT

This report is a summary Syracuse University's team in the Cessna/ONR Student Design/Build/Fly Competition. This is the 6th year of competition for this team, and past results have included a 3rd place finish in 1998. Contained in this document are is a description of the design steps, the people on the team, the design itself, and a manufacturing plan for constructing this aircraft.

The aircraft design for this year's competition is a conventional design, featuring two wing mounted engines, a low wing placement, a double tapered planform, a conventional tail configuration, and a tail dragger configuration. This is the result of many months of effort, and the culmination of many varied analysis methods.

At the date of this report, the design is completed and analyzed, several test models have been constructed, a half-scale model has been constructed, and the competition entry is approximately 75% complete. It is anticipated that the aircraft will be completed, and test flown on or before April 1. This will allow approximately two weeks of flight testing, before the aircraft is packed and shipped to the contest site in Wichita, Kansas.

1.0 EXECUTIVE SUMMARY

1.1 Report Overview

This report presents the details of Syracuse University's entry into the 2001/2002 AIAA Cessna/ONR Student Design/Build/Fly competition. Included are details on the team management structure, the evolution of the design, and a comprehensive plan for manufacturing the competition aircraft. Also included is a detailed drawing package, as well as the aircraft's documented rated aircraft cost.

1.2 Design Development

1.2.1 Design Alternatives Investigated

The team seriously investigated three concepts for the competition aircraft: a conventional aircraft, a twin boom aircraft, and a biplane. The conventional aircraft was eventually chosen as this year's competition aircraft concept. This concept was chosen because it ranked highest in the team's Figures of Merit. There is also a large database of information available on this type of model aircraft proving the viability of the design.

1.2.2 Conceptual Design

During the conceptual design phase, the team made decisions regarding the wing configuration, fuselage configuration, landing gear, tail configuration and number of motors. For the wing, a span limit of 9.5 feet was set, and a high aspect ratio wing was considered preferable. A fineness ratio (ratio of length to width) of approximately five was decided upon for the fuselage to minimize drag, and a conventional tail was also decided upon. Since weight is a primary figure of merit in model aircraft, fixed landing gear were chosen over retractable. The use of two motors was tentatively chosen based upon the limitations on battery pack weight and type.

1.2.3 Preliminary Design

During the preliminary design phase, more details of the aircraft are investigated and decided upon. Decisions were made based upon five FOM: Rated Aircraft Cost, ease of manufacture, strength to weight ratio, efficiency, and cost. The team made three decisions regarding the wing and empennage. The team decided on a double taper wing, balsa for a building material for both the wing and the empennage, and hardwood dowels and nylon bolts for attaching the wing to the fuselage. For the fuselage, the team decided on a payload configuration of either 2-3-3-2 or 3-3-3-3 tennis balls, and a

balsa/hardwood fuselage. Additionally, fixed commercial landing, two engines, and the use of hidden servos was decided upon during this phase.

1.2.4 Detailed Design

For the detailed design phase, the team focused on finalizing design parameters, and analyzing the performance of the aircraft. By the end of this phase all sizes and components were finalized, and a manufacturing plan developed for manufacturing the competition aircraft.

1.3 Design Tools

Several different design tools were used in the course of arriving at a final design. The team employed Syracuse University's library to do research into different configurations. Once a configuration was selected, the team used several commercially available computer software packages to calculate or to confirm calculated values. The team employed Calcfoil to compare different airfoils in terms of their lift, drag, and moment coefficients. Calcfoil was helpful in selecting an airfoil with the desired characteristics, and then Profili was used to print out the airfoils so wing and tail surfaces could be built. Other computer programs used for design include Excel, which was used to keep track of large amounts of data, MathCad, which performed all initial calculations for the conceptual and preliminary design phase and was also used for some calculations in the detailed design phase. The team also developed an optimization code in C that was used during the detailed design phase to optimize the aircraft for the given mission requirements. During the preliminary design phase the team modeled the selected airfoil in Gambit, a grid generator produced by Fluent, Inc., and did a CFD analysis of the model using Fluent in order to help validate the Calcfoil data.

Models were also used extensively to test manufacturing possibilities, as well as aircraft design possibilities. Models testing different designs include the full-scale engine mount and a half-scale model. The engine mount showed that it was possible to mount an engine on the wing, and the half-scale model has tested many design features including the double tapered wing configuration. Models testing different manufacturing methods include the "lost foam" tail, the half wing, and the half-scale model. The "lost foam" tail showed the team that it is possible to manufacture using the lost foam technique, but that this technique has drawbacks and limitations. Both the half wing and half scale model employed hidden servos, and the half scale model also had a carbon fiber tube for the main wing spar.

2.0 MANAGEMENT SUMMARY

The Design Build Fly Team from Syracuse University is made up of students of varying experience levels and backgrounds. Leading the group was senior aerospace engineer Reid Thomas. Senior members Victoria Garnier and Doug Hemphill rounded out the upperclassmen. Also on the team were junior members were Chris Corbin, Greg Conley, and Eliza Honey. Sophomore members included

Mike Czabaj, Mike Nagy, Ben Nesmith, and Brian Foo. All team members are aerospace engineering students.

The team met once a week starting in September to discuss their plans and progress. The preliminary tasks involved utilizing figures of merit to define the aircraft configuration and initial sizing. After the team decided upon a design, members began to work on a half scale model and test sections of design possibilities. Individuals were assigned different tasks and updated the team on their progress and ideas at the weekly meetings. Once building began, all team members worked on construction of the aircraft and its parts. The writing of the report was also a team effort.

Reid Thomas is a three-year veteran to Design, Build, and Fly. Using his experiences from past years and personal model aircraft he has built, he was able to coordinate and direct the design and construction of the aircraft. Thomas was widely involved in all design stages, which included performing calculations, making initial drawings of the aircraft, and construction of the aircraft. Thomas, with the help of other members, purchased many of the items necessary. For manufacturing, Thomas was in charge of the fuselage section.

Senior Tori Garnier and sophomore Mike Czabaj also held leadership positions. Garnier, while helping with the initial calculations, was in charge of the empennage. At meetings she kept minutes and took note of all sketches and ideas. She also helped plan the transportation and accommodations for the trip to the competition in Kansas. Czabaj worked closely with the manufacturing of the aircraft using his knowledge acquired from years of building personal model aircraft. Czabaj, with the help of Thomas, built most of the half scale model as well as test sections of the wing and the aileron. When it came time to build the actual model, he was in charge of making the wing. These team members put a great deal of effort into writing the final report.

The rest of the group consisted of Doug Hemphill, Eliza Honey, Chris Corbin, Greg Conley, Ben Nesmith, and Brian Foo. Doug Hemphill helped with the initial design phase. He came up with sketches and design ideas for the aircraft and helped the team narrow down our choices. Nagy used his modeling experience to assist Czabaj in the manufacturing process and assisted him with his section of the report. He also helped build the test sections, half scale model, and final aircraft. Honey, Nesmith, and Foo helped with the initial design stages, worked countless hours on the final aircraft assembly, and helped write the final report. Corbin and Conley helped with the manufacturing of the aircraft.

2.1 Design Deadlines

At the beginning of each semester, a list of deadlines for the project was handed out. The two lists with the projected and actual completion dates are attached in Table 2.1.

A new approach this year was to build a half scale model before the fall semester was over. Tests will be run on it and modifications to the final model will be made accordingly. Our goal is to have the final plane built and tested a month before the competition.

Thomas and other team members met with faculty members throughout the process to get advice as well as to update them on the process. Toward the end of the design process the team prepared and gave a presentation to faculty members on status of the project and the steps taken to design and construct the aircraft.

Table 2.1 Timeline

FALL SEMESTER		
Task	**Projected Completion Date**	**Actual Completion Date**
Empirical Database	9/01/01	9/01/01
Configuration Selection	9/10/01	9/10/01
Weight Estimation	10/1/01	10/5/01
Wing Sizing	10/10/01	10/12/01
Initial Design	10/10/01	10/12/01
Component Placing	10/14/01	10/16/01
Preliminary Sizing	10/25/01	11/4/01
Landing Gear Selection	10/25/01	10/25/01
Entry Form Due	10/30/01	10/25/01
Airfoil Selection	11/1/01	10/22/01
Power Configuration	11/7/01	12/1/01
CG Location	11/10/01	12/5/01
Control Surface Sizing	11/10/01	12/1/01
Optimization	12/10/01	12/20/01
Half Scale Model Completed	12/21/01	2/1/02
SPRING SEMESTER		
Begin manufacturing	1/15/02	1/20/02
Begin Drafting Report	1/27/02	1/27/02
Half Scale Model Tested	2/10/02	TBD
Rough Draft of Report Complete	2/20/02	3/5/02
Fuselage complete	3/24/02	TBD
Empennage Complete	3/20/02	TBD
Wing Complete	3/27/02	TBD
Aircraft Assembled	3/27/02	TBD
Presentation to Faculty	3/6/02	3/6/02
Components placed	3/27/02	TBD
Aircraft Complete	3/27/02	TBD
Hand report to faculty	3/3/02	TBD
Report Due	3/12/02	3/12/02
Flight testing	4/1/02	TBD
Final Modifications	4/10/02	TBD
Competition	4/26/02	4/26/02

3.0 CONCEPTUAL DESIGN

The goal of conceptual design is to choose a basic design that will meet the mission requirements and, above all, be practical both for design and for manufacturing purposes. The team investigated several different configuration concepts, and ranked them quantitatively by Figures of Merit (FOM). The team then considered other aspects of the airplane, including the wing, motors, landing gear, tail

configuration, airfoil and fuselage. All decisions were made based on previous team experience, research on airplane design, and test models constructed during the decision process.

3.1 Alternate Configurations

The team identified seven possible configurations: standard, biplane, flying wing, blended wing fuselage, twin boom, joined wing, and canard. Based on team experience, blended wing fuselage and canard were immediately eliminated: in the past, these configurations have proven too difficult to design and manufacture. The team then eliminated a flying wing configuration because the design is unstable and difficult to fly without an onboard computer system. The last configuration eliminated was a joined wing, due to the difficulty of such a design and the lack of information about this type of design.
The three remaining configurations were ranked by Figures of Merit (FOM) to choose the best design, as shown in Table 3.1.

Table 3.1 Configuration Rankings by FOM

Figure of Merit	Weighting Factor	Standard Configuration	Biplane	Twin Boom
Payload capacity	1	0	0	2
Manufacturing complexity	2	0	0	0
Manufacturing cost: manpower	2	0	-1	-1
Manufacturing cost: materials	1	0	-1	-1
Rated Aircraft Cost	3	0	-3	-1
Empty weight	1	0	-2	-3
Stability	1	0	0	0
Design difficulty	4	0	-1	0
Total Weighted Score		**0**	**-18**	**-11**

Each configuration was ranked for each FOM on a scale from –3 to 3, with a standard configuration being zero. Each FOM was given a weighting factor, with 1 being the least important and 4 being the most. From these rankings, it can be seen that the standard configuration would be best for this application.

3.2 Wing

There were several design aspects considered relative to the wing: wingspan, aspect ratio, and issues related to low Reynolds number flight. Though contest rules no longer specify a wingspan limit, the team estimated that the total wingspan should be kept to less than ten feet. This keeps the required structural strength to support the wing within acceptable limits; it also helps keep Rated Aircraft Cost lower. A shorter wingspan is also needed to ensure that wing loading is high enough for wind conditions at the competition site. The wing's general shape was also determined by the necessity of keeping

aspect ratio high to increase wing efficiency. Wingtip stall can be a significant problem in low Reynolds number flight. There are several ways to correct for this: increasing the thickness-to-chord ratio; increasing the chord length; and washout, giving the wing incidence at the root and not at the tip. The team constructed a wing test model and found that washout is very difficult to manufacture accurately. This model and a second wing model also tested a double taper shape, changing the taper along the wing to increase the chord length at the tip; this worked well and was chosen as part of the team's design. The team also decided to use a thicker airfoil.

3.3 Fuselage

The preliminary size and shape of the fuselage was estimated based on the payload requirements. For a fineness (length to width) ratio of approximately 5, to reduce drag, the payload configuration would have to be approximately 3-4 balls wide.

3.4 Motors

The team considered using one, two, or three motors and ranked their attributes using Figures of Merit. A scale of 0 to 3 was used, with 3 being the best; for weighting factors, the same scale was used as above.

Table 3.2 Motor Ranking By FOM

Figure of Merit	Weighting Factor	1 motor	2 motors	3 motors
Takeoff under 200 ft. (thrust provided)	4	1	2	3
Under 40 A	4	0	2	2
RAC	3	3	2	1
Battery time	3	3	2	1
Weight	3	3	2	1
Total Weighted Score		**31**	**34**	**29**

By this ranking, it can be seen that the best configuration would use two motors.

3.5 Landing Gear

There were two types of landing gear considered: fixed and retractable. These were ranked using the following Figures of Merit. The same scale was used as for the motors.

Table 3.3 Landing Gear Ranking by FOM

Figure of Merit	Weighting Factor	Fixed Gear	Retractable Gear
Drag	2	0	2
Weight	4	2	0
RAC	3	3	1
Manufacturing complexity	3	3	0
Total Weighted Score		26	7

By this ranking, it can be seen that fixed landing gear is much better than retractable gear for this application.

3.6 Tail

Two types of horizontal tail were considered: a T-tail and a conventional tail. These were ranked using Figures of Merit and the same scale as above.

Table 3.4 Horizontal Tail FOM

Figure of Merit	Weighting Factor	T-tail	Conventional Tail
Drag	2	3	1
Weight	3	1	2
RAC	3	3	3
Manufacturing complexity	4	1	2
Total Weighted Score		22	25

By this ranking, it can be seen that a conventional tail is best for this application.

3.7 Airfoil Selection

Airfoil selection is a key development during the conceptual design phase. It is important to find an airfoil with characteristics that best match the flight regime. For this competition, that regime is low speed flight. For the previous year's competition, the team looked extensively into choosing an airfoil to best fit the flight parameters. Based upon this study, NASA's low speed airfoils, LS-413mod, and LS-417mod were selected as the best candidates. At this stage of the design the LS-413mod was selected for further analysis in the preliminary design phase.

4.0 PRELIMINARY DESIGN

During the conceptual design phase, an aircraft configuration downselect was done to find an aircraft that best satisfied the mission requirements. Mission requirements include flying five sorties, one

with payload, four without, in a ten-minute time period. The aircraft selected during the conceptual design phase had one wing and one fuselage.

The purpose of the preliminary design phase is to determine, in more detail than the conceptual design phase, an aircraft that meets the given mission requirements and goals. This was accomplished using both analytical methods and prototype testing.

4.1 Figures of Merit

Figures of Merit (FOM) provide a means for comparison of design parameters. They allow for a quantitative comparison between different design parameters. Figures of merit used during the preliminary design phase are shown at the end of the preliminary design portion, in Section 4.3, in Table 4.1.

4.1.1 Rated Aircraft Cost

Since Rated Aircraft Cost is a divisor in the team's final score, the lowest possible Rated Aircraft Cost is most beneficial. Rated Aircraft Cost includes the size of the fuselage, the wingspan, the number of motors, and the size of the empennage. At this stage more focus was given to defining the design, based upon sizing and design constraints. Rated Aircraft Cost was analyzed in detail during the detailed design phase, where an optimization code was used to minimize Rated Aircraft Cost and maximize the aircraft performance.

4.1.2 Ease of Manufacture

The team's ability to construct the airplane is an important consideration. While some members of the team have experience constructing model aircraft, more have little or no experience in that area. A simple aircraft design, made of easy to work with materials, would decrease the manufacturing time and allow more team members to participate in the process. The team also had access to only a limited number of facilities available for use, so any manufacturing techniques that needed special facilities not available at Syracuse University would be very difficult to complete.

4.1.3 Strength to Weight Ratio

The aircraft should be as light as possible, but still able to withstand the required flight and landing loads. In order to simulate flight loads, the aircraft must be able to be picked up by its wingtips;

this simulates an approximately 2.5g wing loading. The aircraft structure must also be able to withstand landing.

4.1.4 Efficiency

The aircraft mission requirements stipulate that the aircraft take off in 200 ft and fly five sorties in ten minutes or less; therefore, the batteries need to last through five take-off landing cycles and need ten minutes or five sorties of runtime. An aircraft configuration that is more aerodynamic will be more efficient and require less battery power. Also, a lighter aircraft will require less take-off power, so low-weight, aerodynamically efficient aircraft is best.

4.1.5 Cost

The team, like a business, has been allotted a budget for the year and is expected to stay within that budget. Material costs are a large percentage of the total budget, so lower-cost materials are better than higher-cost ones.

4.2 Design Parameters Investigated

During the conceptual design phase, an aircraft configuration was selected. For the preliminary design phase, design parameters were refined and the aircraft was broken down into parts. The design parameters investigated include wing configuration, wing and empennage structure and materials, wing/fuselage attachment scheme, payload configuration, fuselage materials, landing gear materials, engine configuration, and standard vs. hidden servos.

4.2.1 Wing Configuration

Three wing configurations were investigated: a straight wing, a tapered wing, a double-tapered wing. A straight wing and a tapered wing are easy to design and manufacture, and the team has experience with both configurations. Advantages to a double-tapered wing include better aerodynamic efficiency than either a straight or tapered wing and the possibility of putting incidence in either section of the wing. However, this type of planform is more difficult to manufacture. Since the team had little experience with a double tapered wing, a test section was manufactured in order to assess the manufacturing difficulty. It was found that the double-tapered wing was not much more difficult and required little extra time and skill to manufacture properly, but that adding incidence to the inner section of

the wing increased manufacturing complexity greatly. Figure 4.1 below shows the double-tapered wing model without incidence.

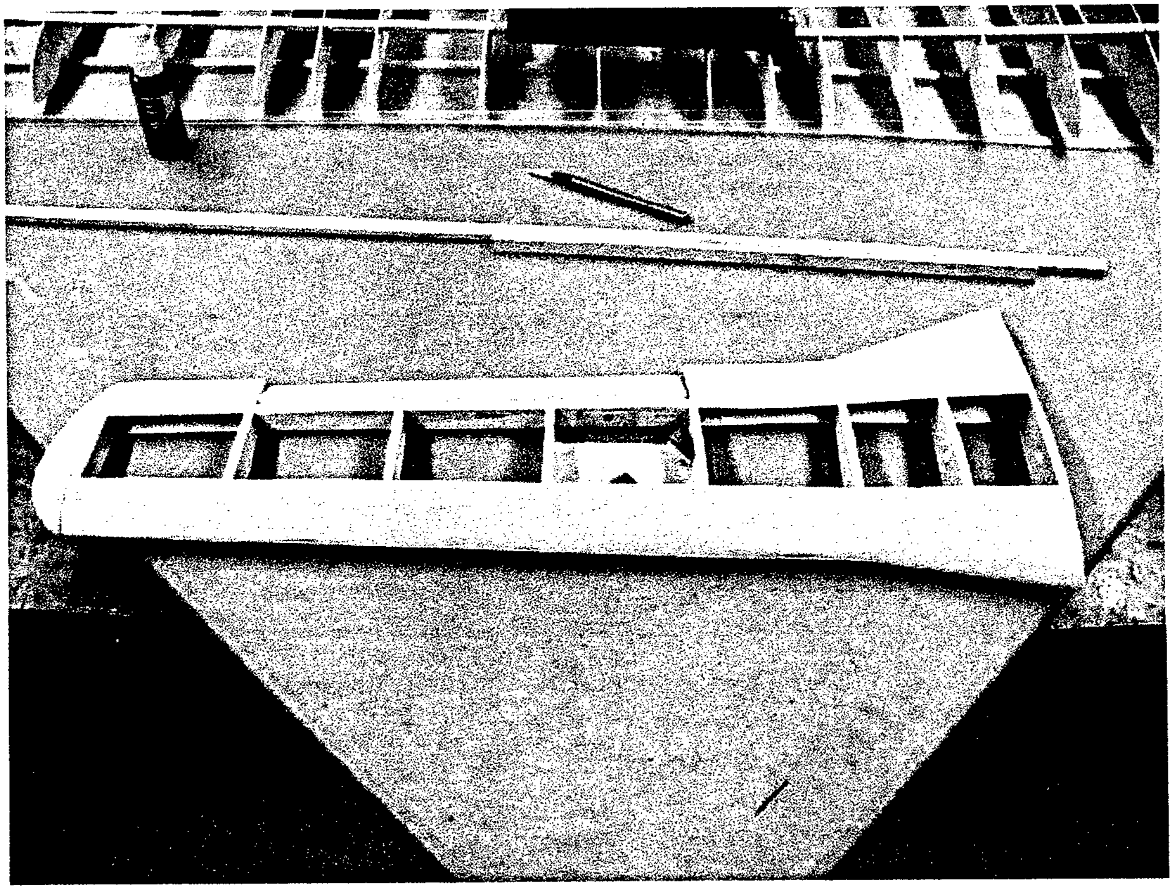


Figure 1 Double Taper Wing Model

4.2.2 Wing and Empennage Structure and Materials

Materials used to construct the wing and Empennage must be strong enough to withstand the aerodynamic and g-loadings, but should be as light as possible. Manufacturing complexity was also heavily considered in choosing a wing/empennage material.

Several different materials were investigated for the wing and empennage: foam-core wing/empennage overlaid with fiberglass, a balsa/hardwood wing/empennage and a "lost foam" fiberglass wing/empennage. A balsa/hardwood wing and empennage would be lighter than a foam-core overlaid with fiberglass, but the foam core with fiberglass wing would be stronger. The team also has experience with both balsa/hardwood wings and foam core/fiberglass wings, and therefore knew from experience that the foam core wing is more difficult to manufacture correctly and, while somewhat stronger, is much

heavier. There was some question as to whether the "lost foam" technique would work, so a model of the horizontal tail was manufactured to test this technique. Figure 4.2 shows the manufactured tail section.

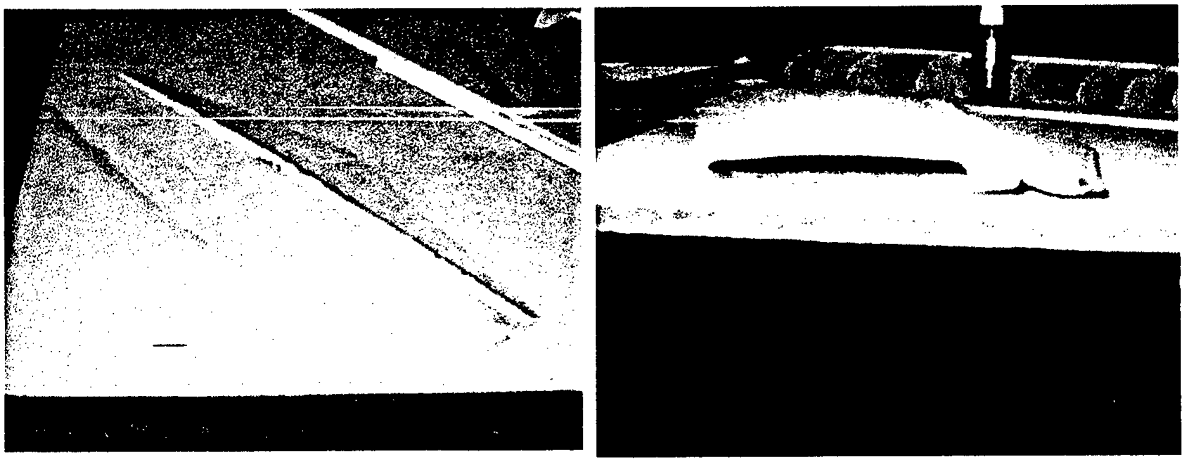


Figure 2 Lost Foam Tail Section

The tail was manufactured with a five-ply fiberglass layup, and was vacuum-bagged and allowed to cure overnight; the foam was melted out with acetone. The result was a tail that was heavier than a balsa/hardwood tail and not as strong. Because of the result produced, the lost foam technique was not utilized for any structural parts.

4.2.3 Wing-Fuselage Attachment Scheme

Since a low wing is used, the wing will be attached using dowels and bolts. Materials investigated for attachment include hardwood and aluminum dowels and nylon and steel bolts. Nylon bolts were determined to be strong enough and lighter than aluminum, and hardwood dowels were also determined to be strong enough and are lighter than aluminum.

4.2.4 Payload Configuration

The payload is in part determined by the contest requirements: softballs must be carried at least two abreast and no more than one deep. Also, because the plane must be balanced both with and without payload, the payload must be centered on the aircraft C_g. In terms of aircraft weight, the team determined that having a payload of 10-12 softballs would be ideal. Using a "fineness" (fuselage length to width) ratio near 5, but keeping in mind RAC and that a shorter fuselage leads to a larger tail, the team investigated the following payload configurations: for 10 balls, 2-2-2-2-2 and 2-3-3-2, for 12 balls, 2-2-2-2-

2-2 and 3-3-3-3. A very long or very boxy fuselage adds drag and so would be less aerodynamically efficient.

4.2.5 Fuselage Materials

The fuselage is the heaviest component of the aircraft. Any materials that could be used to decrease the weight without sacrificing structural integrity should be investigated. Materials investigated include a balsa/hardwood combination, carbon-fiber, and a balsa shell overlaid with fiberglass.

A balsa/hardwood fuselage is easiest to construct, but is heavier than carbon-fiber. However, carbon-fiber is more expensive and more difficult to manufacture, particularly in the complex shapes needed. While team members have experience with carbon-fiber layups, it is felt that this experience is not sufficient to guarantee a quality manufacturing job. A balsa shell overlaid with fiberglass was investigated because the team has experience with this technique: the result is a very heavy, albeit extremely strong, aircraft. A balsa/fiberglass fuselage is no more difficult to construct than a balsa/hardwood fuselage.

4.2.6 Landing Gear Materials

In past years, landing gear have been a problem for Syracuse University teams. For the past two years, the team has used high-grade commercial landing gear. Because of the success of these commercial available landing gear, it was decided that they would again be used.

4.2.7 Engine Configuration

The number of engines plays a large role in rated aircraft cost. The team looked at having one, two, or three engines. It was discovered that three engines decreased the runtime too much and the aircraft wouldn't be able to finish its sorties, so that configuration was abandoned. The configurations left were two engines mounted on the wing or one engine mounted on the nose. After analytical calculations, it was decided that one engine could, in theory, provide the necessary thrust; however, after testing the engines, the team was not confident that, in reality, one engine could provide the necessary thrust to take off several times. Having two engines would increase the thrust, but would also increase the weight of the overall aircraft and would require that the wing be built to withstand necessary loads. The team is using the same engines that were purchased the previous year, so purchasing cost is non-existent.

4.2.8 Servo Selection

Two kinds of servos were investigated for this year's aircraft. Standard servos are used in most model aircraft in this class, and involve the servo being placed in the aircraft body with wire running to the control surfaces. Running wire increases the likelihood that the servos could malfunction, and also creates drag on the aircraft. Hidden servos, which would be placed in the wing or tail, eliminate these problems. They have no exposed parts, so they have no adverse effect on drag, and they don't require wire to be run from the servo to the control surface, thus eliminating the risk that wires might break or get caught on something.

4.3 Figure of Merit Table

The figure of merit results are displayed in Table 4.1. Each design parameter was ranked from –3 to 3 with a simple, balsa aircraft being the baseline. The highest ranking parameters were kept for refinement in the detailed design phase.

Table 4.1 Investigated Design Parameters

Design Parameters		Rated Aircraft Cost	Ease of Manufacture	Strength to Weight Rati	Efficiency	Cost	Total
4.2.1 Wing Configuration	Straight	0	0	0	0	0	0
	Taper	0	0	0	1	0	1
	Double Taper	0	-1	1	2	0	2
4.2.2 Wing and Empennage Structure and Materials	Foam Core with Fiberglass	0	-2	2	-1	-1	-2
	Balsa/Hardwood	0	0	0	0	0	0
	"Lost Foam" Fiberglass	0	-2	-2	-1	-1	-6
4.2.3 Wing-Fuselage Attachment Scheme	Hardwood Dowels	0	0	0	0	0	0
	Aluminum Dowels	0	0	1	-1	-2	-2
	Metal Bolts	0	0	1	-1	-1	-1
	Nylon Bolts	0	0	0	0	0	0
4.2.4 Payload Configuration	2-2-2-2-2	0	0	N/A	0	N/A	0
	2-3-3-2	1	0	N/A	1	N/A	2
	2-2-2-2-2-2	-1	0	N/A	-1	N/A	-2
	3-3-3-3	1	0	N/A	1	N/A	2
4.2.5 Fuselage Materials	Balsa/Hardwood	0	0	0	0	0	0
	Carbon-Fiber	0	-2	2	-1	-2	-3
	Balsa Overlaid with Fiberglass	0	-1	1	-1	-1	-2
4.2.6 Landing Gear Materials							0
							0
4.2.7 Engine Configuration	One Engine	0	0	N/A	0	0	0
	Two Engines	-1	0	N/A	1	0	0
	Three Engines	-2	-1	N/A	2	-1	-2
4.2.8 Servo Selection	Standard Servos	0	0	N/A	0	0	0
	Hidden Servos	0	1	N/A	1	0	2

4.4 Models

Several models were built to test some untried manufacturing techniques and to determine actual manufacturing complexity. Concerns arose during the design process that necessitated the building of models. Models built include a full size engine mount and a half-scale model.

4.4.1 Engine Mount

When selecting material for the wing, one main concern was strength. Since the engines could conceivably be mounted on the wing, the wing needed to be strong enough to take those additional loads. Since balsa was the material of choice for the wing, a balsa test section of an engine mount was built to ascertain that a balsa wing would be strong enough to hold the engine. The balsa test section proved strong enough to hold the engine and so was selected as the wing material.

4.4.2 Half-Scale Model

The team built a halfscale model of the competition aircraft. The purpose of this model was to test some new materials and techniques including a carbon-fiber main wing spar and hidden servos. The wing was manufactured in two parts and then joined in the center. The model is shown in Figure 3.

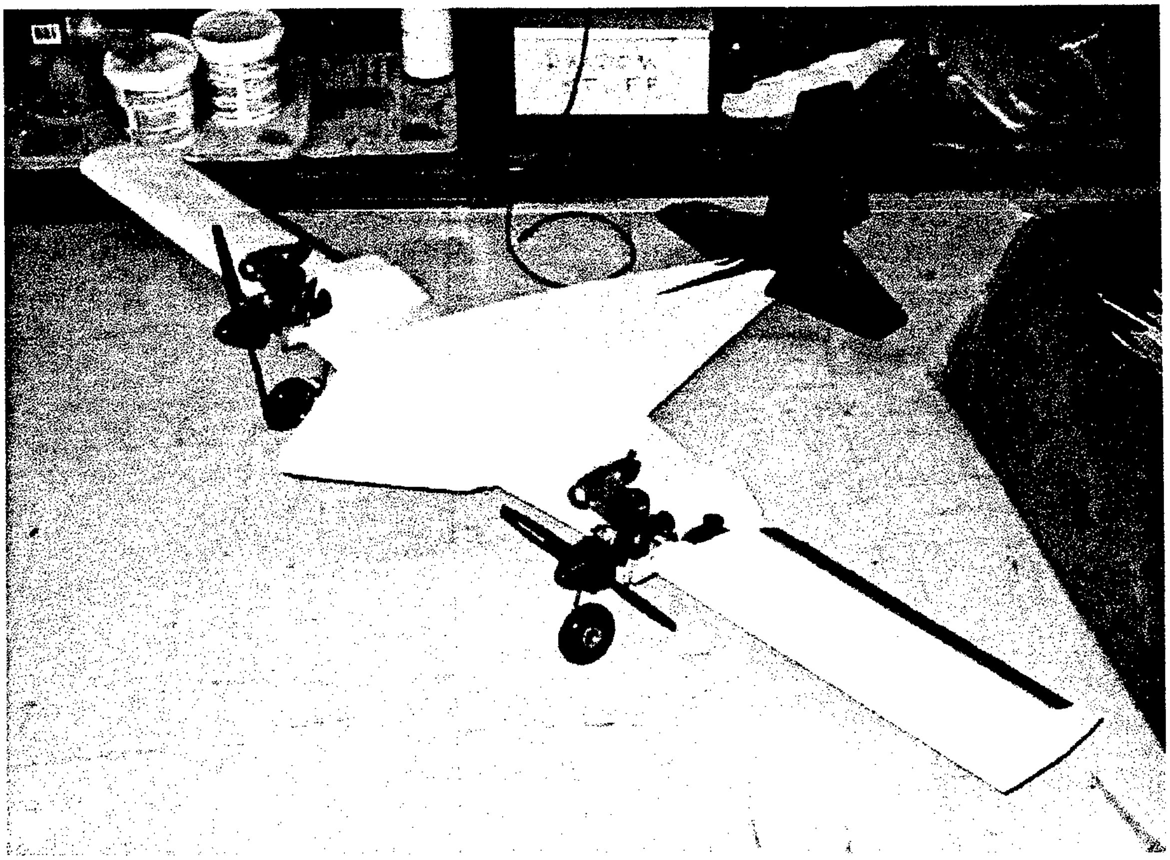


Figure 3 Half Scale Model

The carbon fiber spar worked well on each half, but the team was unable to join the halves well. In order to use the carbon-fiber spar for the wing on the competition aircraft a better method of joining the wing halves would have needed to be developed. The team decided not to use the carbon-fiber wing spar for the competition aircraft due the concerns about the strength of the cent er joint.

Unlike the carbon-fiber wing spar, the hidden servos worked better than planned. The hidden servos are easier to install than conventional servos and they move control surfaces just as easily. Since the servos are inside the wing, there are no outside servo horns to add drag to the aircraft surface. The team decided to use hidden servos for all control surfaces in the competition aircraft.

5.0 DETAIL DESIGN

During the Preliminary Design phase, discussed in Section 4.0 an extensive MathCad file was created of aircraft sizing and performance. Using that as a base, the team developed an optimization code in C. The purpose of this code was to refine the sizing parameters from the preliminary design phase, and optimize them to achieve the highest possible flight score. The aircraft performance ranking criteria was a combination of Rated Aircraft Cost and mission performance (i.e. the time to complete the

entire mission). Using this performance index, the optimal sizing parameters were taken from the code for further analysis.

5.1 Performance Data

The code used for sizing optimization also included a performance subfunction. This subfunction was used in calculating take off distance, mission performance, and aircraft weights for balancing the aircraft. With this information, stability derivatives were calculated to verify the aircraft stability. Along with the stability, the control and handling of the aircraft was also addressed with the final sizing of the control surfaces: rudder, elevator and ailerons.

5.1.1 Take Off

Take off distance is an important characteristic of every aircraft, and with the competition limit of 200 feet, it is of particular interest to this model. The takeoff distance was found as part of the performance code. The equation for take off distance was taken from Ref 1. and is shown in equation 1.

$$S_G := \int_0^{V_{10}} \frac{V}{g \cdot \left[\left(HP - \mu_g \right) - \frac{\left(C_{Dg} - \mu_g \cdot C_{Lg} \right) \cdot q}{WL_{min}} - \phi \right]} \, dV \qquad (1)$$

Using this equation, the take off distance for both the loaded, and unloaded conditions were found. The take off distance fully loaded was found to be 141 ft. The take off distance without payload was found to be 97 ft.

5.1.2 Handling Characteristics

With the sizing output from the code, stability derivatives were found. The required derivates were found from Ref 2. to be:

- $C_{l\alpha} > 0$ – aircraft lift curve slope
- $C_{m\alpha} < 0$ – pitching moment curve slope
- $C_{mq} < 0$ – variation of pitching moment coefficient with pitch rate
- $C_{Du} > 0$ – variation of drag coefficient with speed
- $C_{mu} > 0$ – variation of moment coefficient with speed
- $C_{y\beta} < 0$ – variation of moment coefficient with sideslip angle

The derivatives were found per the methods in Ref 2. Two assumptions were made for the calculation of these derivatives; first, that compressibility effects could be ignored due to the low flight speeds, and second, that the aircraft was flying at small angles of attack in normal flight. This design met all of the requirements laid out for stability. The results of this analysis are displayed in table 5.1.

Table 5.1 Stability derivatives

$C_{l\alpha}$	1.113 rad^{-1}
$C_{m\alpha}$	-0.1116 rad^{-1}
C_{mq}	-1.027 rad^{-1}
C_{Du}	0.00
C_{mu}	0.00
$C_{y\beta}$	-0.2543 rad^{-1}

5.1.3 Load Capability

The load capacity of the aircraft is another important aspect. This was initially found during preliminary design, and then further refined in this portion of the design phase. The code result was a load capacity of 3.123 during sustained turn. A comparison was made between this and a structural analysis of the wing. The wing was modeled as a cantilever box beam, with an applied point load at the tip. Given the box beam sizes and materials, a load factor of 5.78 was found for the wing. For safety purposes the code output of 3.123, the minimum of the two, was taken to be the final load capacity.

This load meets the competition minimum of 2.5, and implies that a smaller and lighter structure could be attained. For several reasons, the sizing was not changed. The first reason is due to the sizes available for the given material. Optimizing to reduce the load limit to the 2.5 specified would result in non-standard sizing of the main structural members. Altering the main spars to reach this state was not favored as it would increase time and complexity, as well as adding another chance for error. The second reason for maintaining the high load factor was the unpredictability of flight conditions. Given the competition location, Wichita Kansas, and the history of high winds during competitions, additional strength was favored as an added factor of safety. Also, throughout the life cycle of the aircraft, numerous conditions, poor handling, hard landings, transport, exist where added strength is a benefit.

5.1.4 Drag Polar

In order to determine the range and endurance of the aircraft, an accurate measure of the drag is needed. With these range and endurance calculations the mission performance can be found. A level one estimation was used for drag calculations; this was expected to provide an accurate drag prediction, with a slight tendency for over-prediction. Drag is comprised of three main components: induced, compressibility, and skin friction drag. Because of the low Mach numbers of the flight, drag do to compressibility affects was neglected. At this stage the drag polar could be evaluated by equation 2.

$$C_D = C_{Do} + KC_L^2 \qquad (2)$$

Here C_D is the total drag, C_{Do} is the zero-lift drag, K is the induced drag constant, and C_L is the aircraft lift coefficient. Part of C_{Do} is the skin friction drag. This can be found after computing the skin friction coefficient for each body. Assuming a turbulent boundary layer, the skin friction coefficient is found with equation 3.

$$C_f := \frac{0.455}{(\log(Re))^{2.58}} \qquad (3)$$

C_f is the skin friction coefficient, and Re is the Reynold's number, based upon body length. Reynold's numbers were found at the stall speed of the aircraft. C_f values were found for the fuselage, the wing, and both the horizontal and vertical tail. This was found with equation 4.

$$C_{Do} := \frac{1.2 \cdot \left(\sum_n C_{fbody} \cdot S_{wetbody} \right)}{S} \qquad (4)$$

In equation 4, the C_{fbody} is the skin friction coefficient of the specific body, and $S_{wetbody}$ is the wetted area of that body. Here S is the reference wing area.

This left the calculation of induced drag to finalize the aircraft drag polar. The induced drag coefficient was found using equation 5

$$K = 1/\pi Ae \qquad (5)$$

A is the aspect ratio of the wing and e is Oswald's span efficiency factor. With unswept wings this value is estimated using equation 6.

$$e = 1.78(1 - 0.045A^{0.68}) - 0.64 \qquad (6)$$

The resulting drag polar is shown in Figure 4.

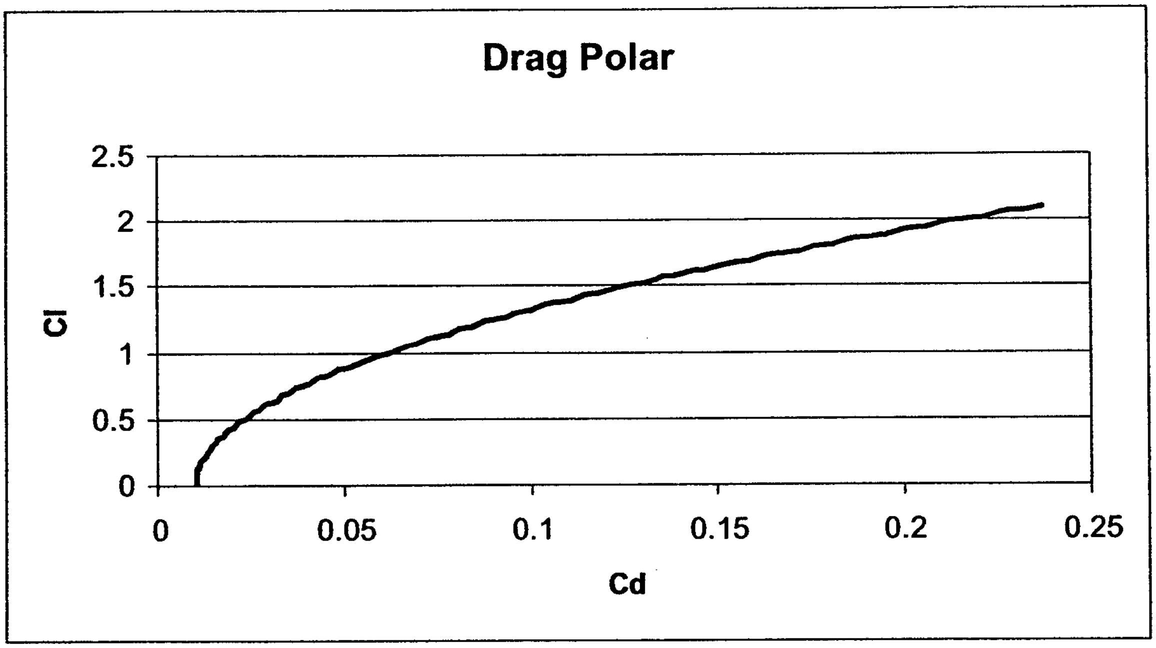


Figure 4 Drag Polar for Aircraft

5.1.5 Mission Performance

Overall mission performance was also found with our code. Rated Aircraft Cost was found as part of the overall performance, but was removed from the calculation of mission performance and is detailed in Section 5.3. Mission performance was found by discretizing the mission into the separate portions, and then summing the time for all of those sections. The mission was broken into small sections, representing the main portions of the total mission. The cruise speed was found as part of our battery and motor configuration, to be 56 mph. With this known, as well as the course geometry, take off performance, and turn performance, the total mission time was found to be 9.705 minutes.

5.1.6 Aircraft Weight and Balance

The structural weight of the aircraft was found from the code outputs, and was added to the individual component weights already known. Details about the selected components are given in Section 5.2. The weight models used were taken from Ref 4, and the weight data is shown in Table 5.2.

Table 5.2 Weight breakdown

Component	Weight
Wing	3.998
Empennage	3.226
Fuselage	10.474
Components	4.375
Payload	5.000
Total	***27.073***

With the component weights all known (including all structural weights) the aircraft could then be balanced. More details on component placement can be seen in Section 6.0, the aircraft drawing package. The aircraft balance was achieved at 23.42 inches from the nose of the aircraft.

5.2 Component Selection

There are several important components that required attention on an aircraft wide, systems approach. The first is the radio control equipment. The team chose to continue using their Futaba 6XA transmitter system. This is a six channel computer radio equipped with dual rates, programmable channels, and the required safety features. Due to the large scale of these aircraft, high torque servos are required. Based on testing, hidden servos were decided upon for all control surfaces, meaning that high performance servos would be required to meet the torque and space requirements. Tower Hobbies TS-70MG S2K servos were selected for all control surfaces.

Landing gear, another key feature were selected based upon previous experience. Robart Robostruts were again selected for their high performance, and quality. These were used in conjunction with Sullivan Skylite four inch tires for the main gear. A Dubro large scale tail wheel and tail wheel assembly were also selected.

The batteries were selected to be Sanyo 1800SCR, and were purchased from Advanced Battery Systems Inc. These batteries were paired with Astro Flight 640G engines. These engines were purchased for last year's competition, and were also suitable to the competition this year.

5.3 Rated Aircraft Cost

Rated Aircraft Cost is one of the most important parameters of the competition as it is a direct score multiplier. The performance parameter of the code was based upon mission performance, as well as Rated Aircraft Cost. The equation for Rated Aircraft Cost is shown in equation 7.

$$RAC := \frac{(A \cdot MEW + B \cdot REP + C \cdot MFHR)}{1000} \quad (7)$$

A description of these variables, as well at the actual Rated Aircraft Cost is shown in Table 5.3

Table 5.3 Rated Aircraft Cost

Coefficient	Description	Value
A	Manufacturers Empty Weight Multiplier	100
B	Rated Engine Power Multiplier	1500
C	Manufacturers Cost Multiplier	20
MEW	Manufacturers Empty Weight	22.073
REP	Rated Engine Power	6.25
MFHR	Manufacturing Man Hours	220.958
	Total	***16.00***
	REP:	6.25
	Number of Engines	2
	Total Battery Weight	5
	MFHR:	220.96
	Wing Span	9.39
	Max Exposed Chord	19.757
	Control Surfaces	2
	Fuselage Length	5.67
	Vertical Surface, active control	1
	Horizontal Surface	1
	Servos and Controllers	6
	Engines	2
	Propellors	2

5.4 Final Aircraft Parameters

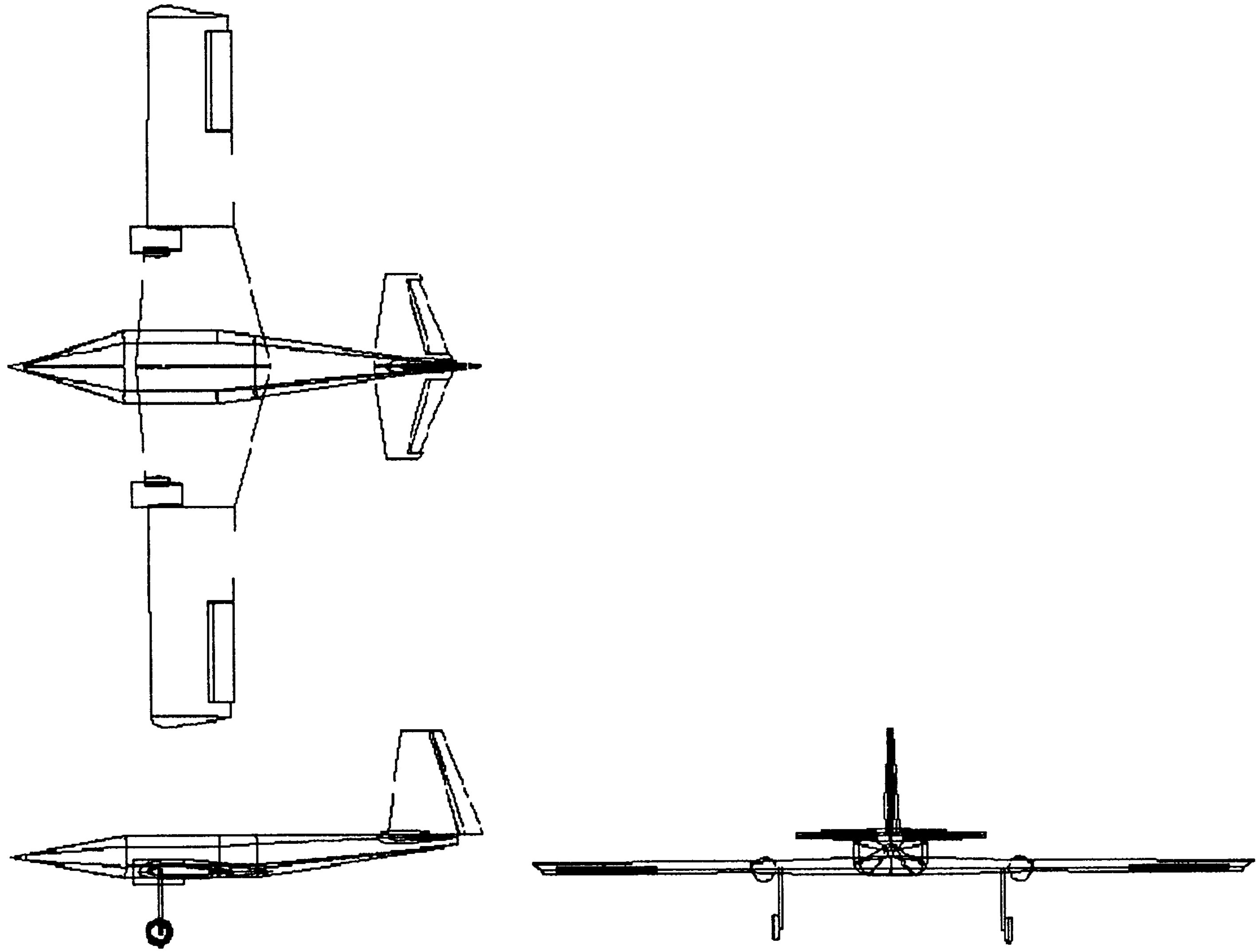
The results of the detailed design phase are shown in Table 5.4. This is the data that has been used to construct the competition aircraft.

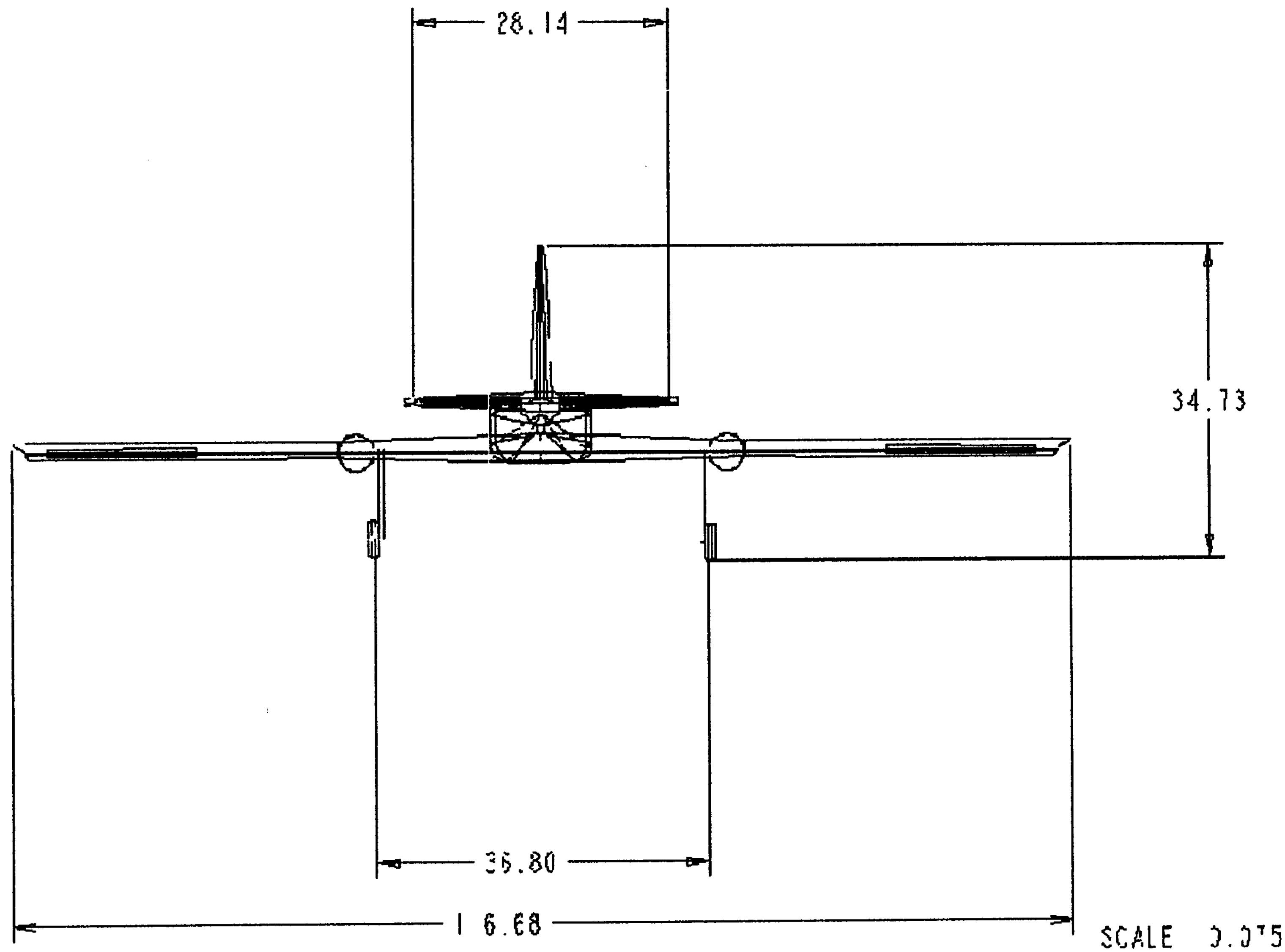
Table 5.4 Final Aircraft Parameters

Configuration	Conventional, aft tail
Wing Location	Low
Engines	2
Motors	Astro Flight 640G
Batteries	Sanyo 1800 SCR
Landing gear	Fixed, tail dragger
Wing Span	9.39 ft
Wing Area	11.7572 ft^2
Fuselage Length	5.67 ft
Horizontal Tail Span	2.345 ft
Horizontal Tail Area	1.834 ft^2
Vertical Tail Span	1.394 ft
Vertical Tail Area	1.296 ft^2
Softballs	10
Payload Configuration	2-3-3-2
Maximum Take Off Weight	27.073 Lbs
Take Off, Empty	97 ft
Take Off, Full	141 ft
Stall Speed	38.462 mph
Cruise Speed	56 mph

6.0 DRAWING PACKAGE

With all drawings in Section 6, the units are inches.

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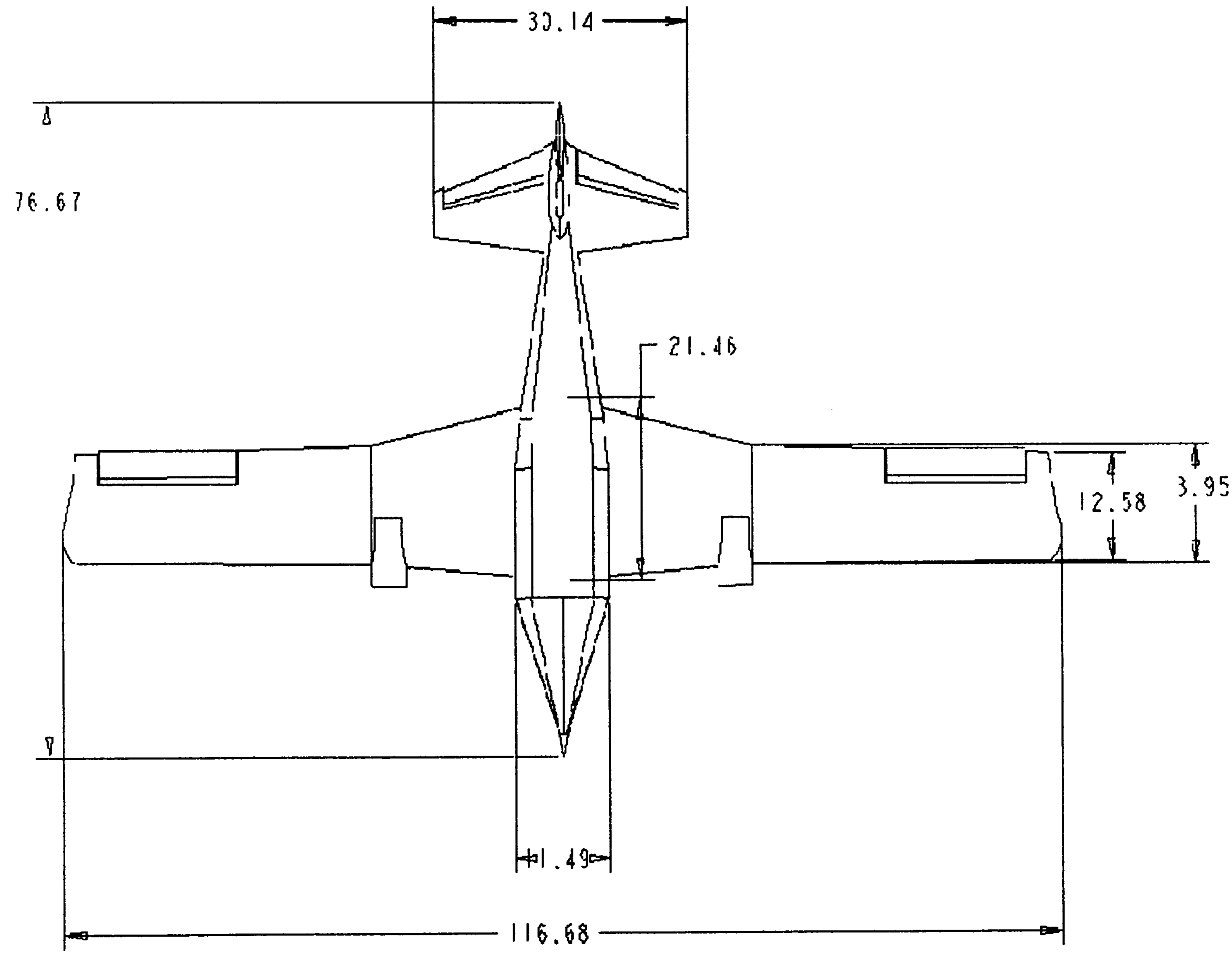
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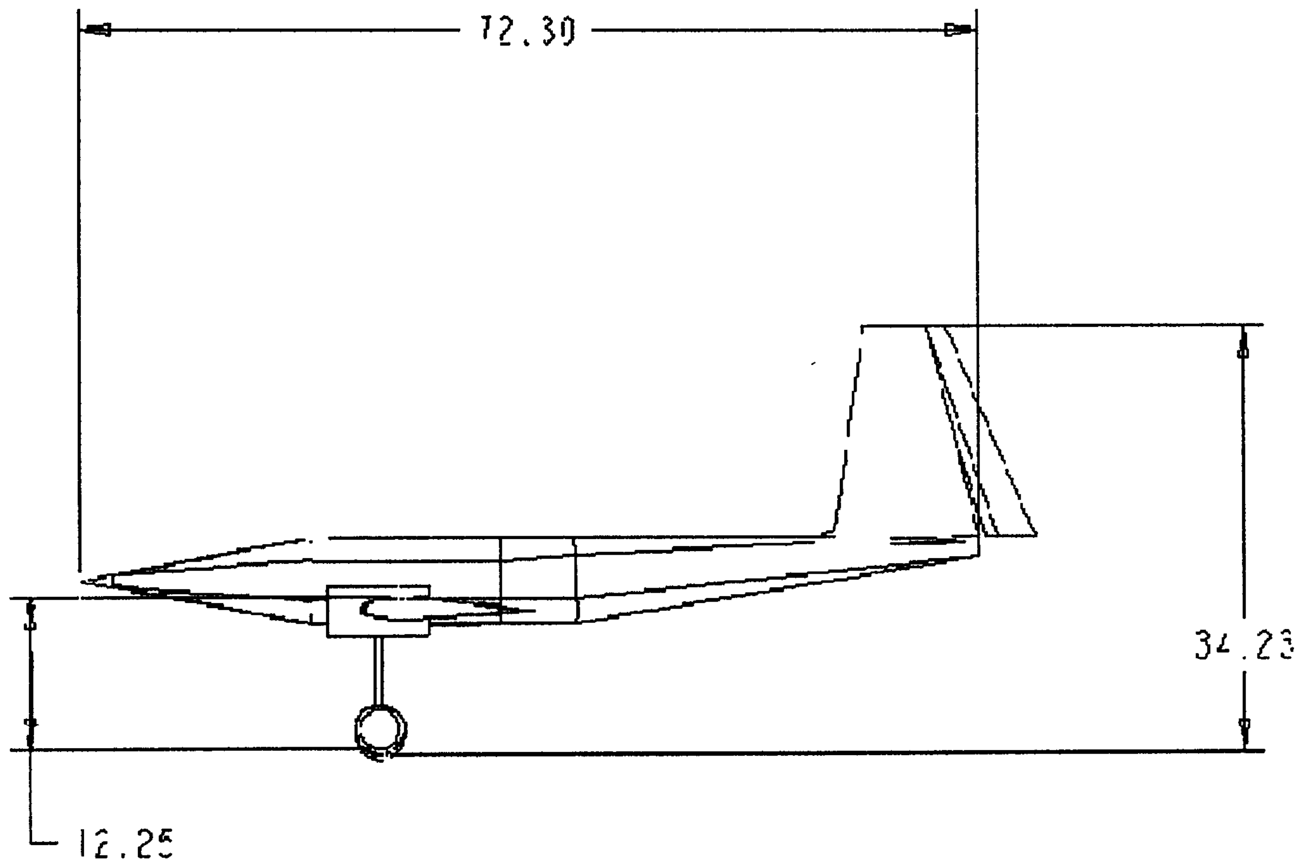
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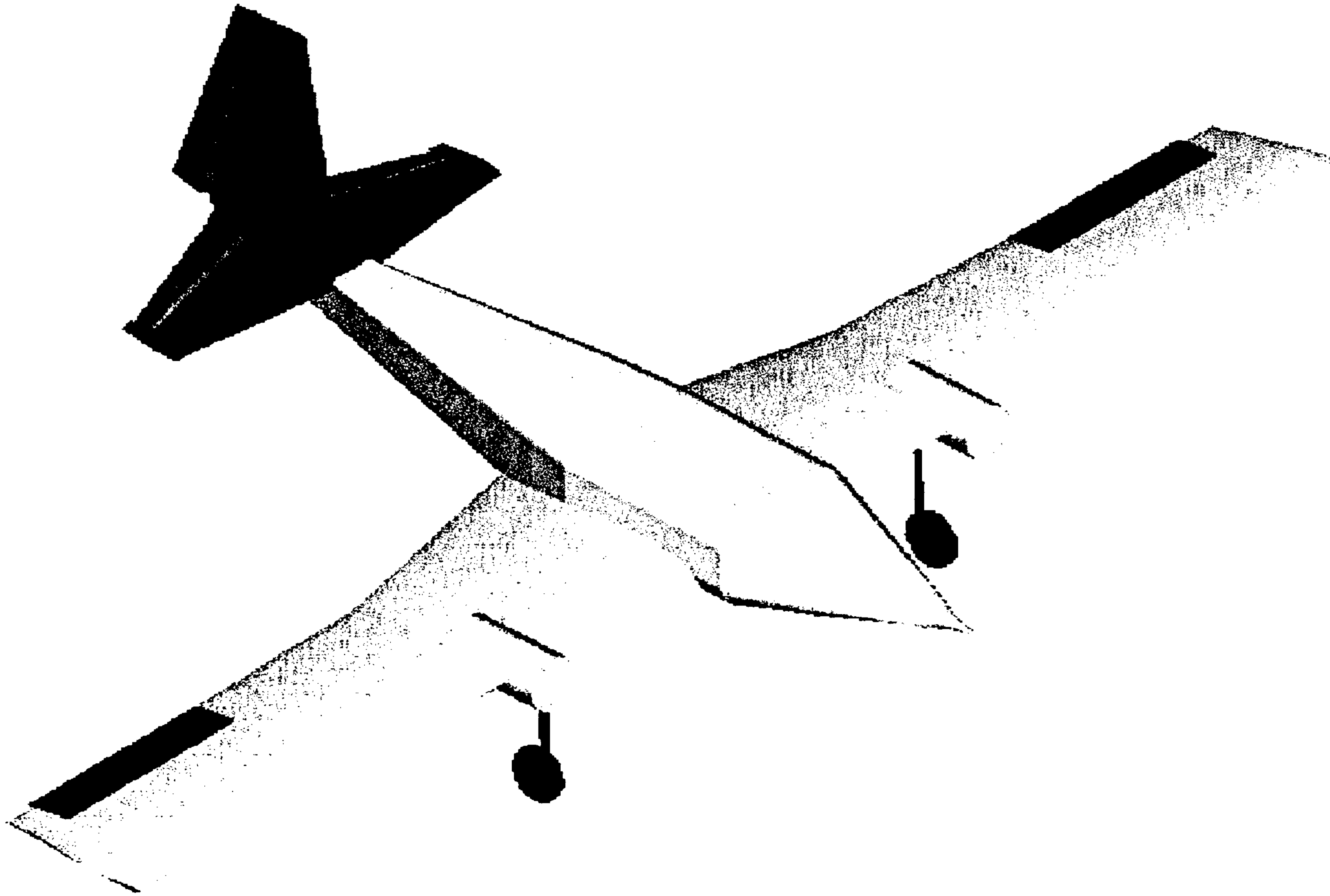

7.0 MANUFACTURING PLAN

The manufacturing plan is designed to minimize cost and man hours while still providing a competitive aircraft. All materials and plan selections are made in accordance with chosen figures of merit while taking into account the team's modeling skills and past experiences. A manufacturing time table is presented at the end of this section, Table 7.2.

7.1 Figures Of Merit

The team used seven figures of merit (FOM) when considering manufacturing materials and plans:

- Availability
- Reparablility
- Strength/Weight Ratio
- Required Skill Matrix
- Past Experience
- Monetary Cost
- Man Hour Cost

The team is placing heaviest consideration on the Strength/Weight Ratio and Past Experience categories. Strength/Weight Ratio is an important consideration due to the large size of this year's plane; the team needs high-strength materials to carry the payload, but light materials to get off the ground. Past experience is also an important consideration because it allows the team to make decisions based on what has and has not worked instead of relying purely on theory.

7.2 Wing Manufacture

From an experience standpoint, wood is the best material for wing construction. It is easy to work with, inexpensive, and common in radio-controlled aircraft. In past years the team experimented with foam core wings covered with a fiberglass/epoxy composite. This configuration has, theoretically, a high strength/weight ratio, but lack of experience has prevented the team from achieving the best structure in past years. The team also experimented with a method called "Lost Foam" technique, which also involves composites. The same exact routine is used as in foam/fiberglass wings but at the final stage the foam is "melted" away with acetone. This method leaves a fiberglass shell, which is lighter than a foam core configuration. However, this method led to a structurally weak specimen and with further

analysis, it was determined that this kind of method would require the use of carbon fiber to ensure that the wing could withstand the design loads. Using carbon fiber would ultimately lead to a very high physical cost.

Finally, it was determined that balsa and spruce wing would be preferable; it has good strength/weight ratio and it is very quick and easy to build. Since wood is the most common material used in model aircraft building, the team has a great amount of experience with it. A summation of the wing material comparison is presented in table 7.1.

Table 7.1 Figures of Merit

Material	*Availability*	*Reparability*	*Strength/ Weight*	*Skills Needed*	*Past Experience*	*Time*	*Cost*
Wood	5	5	2	1	5	2	2
Fiberglass	4	2	4	4	4	4	3
Carbon Fiber	2	2	5	4	2	4	5
Foam Core	2	3	2	5	4	2	3

Because of the large size, the wing will be manufactured in two semi-span panels. The "box" structure of the wing will be made out of 3/32" balsa ribs with ½" by ¼" spruce spar located at quarter cord on top and bottom. The spruce spar will be strengthened by the 1/16" plywood from sides.

The leading and trailing edge will be made out of solid balsa with about a 1/16" of excess on both sides to help the sheeting process. The ailerons will be cut out balsa, and located between ribs 14 and 18. The aileron servos will be placed horizontally between the 14th and 15th ribs.
The engine mounting will made out of 1/8" plywood, placed between seventh and eighth rib, and blended with 1/16" balsa into the wing. Two NACA flush inlets will be incorporated into the mounting to ensure proper cooling of engines and the speed controllers.

The wing panels will be then joined at 5-degree dihedral, and sheeted with 1/16 balsa. The joint will be covered with fiberglass composite to ensue sufficient strength. The wing tips will be shaped out of solid balsa block.

7.3 Fuselage Manufacture

The fuselage was constructed with typical box fuselage techniques. The sides of the fuselage were assembled from 3/32 inch balsa sheeting, with allowances for the wing and horizontal tail. 1/32 inch plywood doublers were added to the sections with the greatest loads, around the wing and tail gear positions. Bulkheads were then cut from varying thicknesses of ply, an epoxied to one fuse side. Both sides were brought together by epoxying the second side to the formers. The front section was added by way of spruce longerons, and was sheeted with 1/8th inch balsa. The top and bottom of the fuse were completely (minus cutouts) sheet with 1/8th inch balsa. Triangular stock was placed at the corners of the

fuse, allowing for significant rounds to be added to the design. A floor for the payload section was added out of 1/16th in ply, and a box was built for the payload. Doing this allowed the payload to be restrained by the sides of the fuselage, as well as formers inside it. The fuselage was covered in monocoat, as was the rest of the aircraft.

7.4 Empennage Manufacture

The empennage consists of one vertical tail with a control surface and one horizontal tail with two joined control surfaces.

7.4.1 Materials Used

Materials investigated for the empennage were balsa, fiberglass-covered balsa, and the "lost foam" fiberglass. From previous experience, the team knew that fiberglass-covered balsa was heavier and more difficult to manufacture, but stronger than just balsa. It was decided that the loads on the empennage were not large enough to warrant this extra weight and complexity, so a fiberglass/balsa tail was discarded.

In order to decide between a balsa tail or a "lost foam" empennage the team tested the "lost foam" technique. The purpose of building the test section was to show whether the "lost foam" technique was a viable building technique and to see how the required skill matrix and manufacturing time compared with an entirely balsa tail. The result was that the "lost foam" fiberglass test section proved that the process is doable with the resources available; however, the resulting specimen was heavier and weaker than a balsa test section of the same shape. The final material selected for the empennage was balsa.

7.4.2 Vertical Tail Manufacture

The vertical tail consists of a monocoated tail and rudder. The tail is unswept around the quarter-chord, and the quarter-chord spar is a balsa box beam. The ribs are 3/32" balsa, and all sheeting is also 3/32" thick. The rudder is attached to the tail with standard CA hinges, and is moved using a hidden servo in the tail. The hidden servo is screwed into a piece of 1/8" plywood sitting between two ribs. The entire tail is covered with monocoat.

7.4.3 Horizontal Tail Manufacture

The horizontal tail consists of a monocoated tail and elevator. The tail was manufactured in two pieces, each of which is unswept around the quarter-chord, and the quarter-chord spar is a balsa box beam. All ribs are 3/32" balsa, and all sheeting is also 3/32" thick. Once the two halves of the tail were sheeted, they were joined with 30 minute epoxy.

Each part of the elevator is attached to the tail with standard CA hinges, and is moved using a hidden servo in the tail. The hidden servo is screwed into a piece of 1/8" plywood sitting between two ribs on one side of the tail. The servo is connected to the elevator on one side of the tail and a piece of copper wire sitting between the two halves of the tail moves the other elevator. The entire tail is covered with monocoat.

7.6 Systems Integration

The control surfaces use an all-internal rotary driver system, which eliminates any external hardware and drag associated with it. It also decreases the slop and improves the overall appearance of the airplane. The system includes a high performance servo combined with an "L" shaped steel drive shaft – placed at a 45 degree angle with the hinge line.

Robart's 670 Robostruts with four inch Sullivan Ultralight wheels will be used as the main landing gear. Struts will be epoxied into a basswood block located on both sides of the wing, ahead of quarter cord, between the seventh and eighth ribs.

The tail wheel will be manufactured by Du-Bro and will include a large scale bracket and a 1.5" tail wheel. It will be mounted on the fuselage, with its steering arm connected to the rudder.

The battery and receiver are wrapped in foam to reduce vibrations and are placed in a forward compartment.

Table 7.2 Manufacturing Time Schedule

Task	Target Date	Completed Date
Manufacturing Start Date	1/15/02	1/20/02
Fuselages Framed Up	1/20/02	2/5/02
Wing Framed Up	1/20/02	2/10/02
Tail Surfaces Framed Up	2/1/02	2/12/02
Control System Assembly	3/1//02	TBD
Tail Assembly	3/20/02	TBD
Fuselage Interior Work	3/24/02	TBD
Fuselage Sheeting	3/24/02	TBD
Wing Sheeting	3/24/02	TBD
Component Placement	3/25/02	TBD
Covering with MonoKote	3/27/02	TBD
Balancing	3/30/02	TBD
First Flight	4/1/02	TBD
Flight Test Modifications		
Competition	4/26/02	4/26/02

2001/2002 Cessna/ONR Design Build Fly Competition
American Institute of Aeronautics and Astronautics

The University of Texas at Arlington

Third Floor Campers

Present.......

"BLACK MAGIC"

Table of Contents

1. Executive Summary

The student chapter of AIAA at The University of Texas at Arlington composed a team of 9 Students who desired to compete in the 2002 AIAA DBF competition. The competition involves designing, building, and flying a remote controlled aircraft with the task of flying 3 sorties per flight period. The sorties demonstrate the aircraft's capability of controlled flight in loaded and unloaded configurations with a cargo of 10 to 24 softballs. Each flight is not to exceed 10 minutes. The team's goals are to design and build an aircraft that can achieve a high score according to the competition guidelines. The score will consist of the product of the written report score times the summation of the best three of five flight scores divided by the aircraft rated cost. Each flight score is based on the amount of circuits completed times the amount of softballs carried during the cargo phase divided by the total flight time. The engineering process included three phases, the conceptual, the preliminary, and the final detailed design, all of which are geared towards optimizing the total score. The aircraft design was consistently under development as phases progressed from one to the next. The project was completed with the manufacturing phase.

1.1 Conceptual Design

In the conceptual design, the group brainstormed several configurations of the aircraft. This addressed cargo location, wing location, tail style, landing gear configuration, propulsive plant placement, and engine direction. 4680 aircraft were systematically sketched and critiqued. Each member of the team was designated to review the sketches from their individual standpoints. The standpoints were in aerodynamics, rated cost, real cost, sex appeal, ease of manufacture, subsystems allotment, mechanical limitations, safety, and ground operation considerations. The selection of only four configurations was allowed for future evaluation. High and low wing tee tail configurations, flying wing, and a high wing canard were the four configurations chosen for further evaluation.

1.1.1 Conceptual Design Alternatives[1]

Of the 4680 different aircraft configurations there resulted a systematic method of combinations of basic parameters. The alternatives used were wing placement consisting of high, low and mid wing. The tails considered were of several configurations including conventional, T-tail, V-tail, H-tail, triple tail, inverted V-tail, twin tail, Y-tail, twin boom, boom inverted V-tail, ring, canard, and tailless. Gear alternatives included tricycle, bicycle, single main, tail dragger, and quad. A bogy landing gear system was not considered for this competition. For weight purposes the cargo positioning was varied from internal of wing, internal of body, and pod mounted. Powerplant options included the amount of engines ranging from 1 to 4 and the option of pusher or tractor means of propelling.

1.1.2 Conceptual Design Tools

The design tools we used at this level were personal knowledge of individual from their educational and personal backgrounds, and brainstorming. Stronger considerations were given to our team discussions and standpoints for argumentation. Reference books were used to study theoretical behavior for certain configurations.

1.1.3 Conceptual Design Results

The individual standpoints of the team members were ranked by importance, as agreed upon by the whole team. By ranking these standpoints, the team was able to narrow down the field of configurations to the selection of just four. The four configurations selected were high wing tee tail, low wing tee tail, canard, and flying wing. Of these four, the canard was the one to make the final cut.

1.2 Preliminary Design[2]

During the preliminary design phase, the canard configuration selected was studied to a greater detail using analytical programs, group reasoning, and discussion. Aerodynamic performance, cost analysis, and manufacturing process were the main focus in optimization of the parameters with respect to performance and scoring ability of the canard configuration.

1.2.1 Preliminary Design Alternatives

The alternatives in this phase pertained to the configuration of the payload which in turn defined the dimensions of the cargo bay, placement of the subsystems, boom lengths, weight, wing area, and stabilizer areas. Also the wing planforms were evaluated, specifically rectangular, elliptical, and tapered. These parameters were found using preliminary calculations for rough drawings to be applied to the KK-AREO[3] code.

1.2.2 Preliminary Design Tools

KK-AREO v. 3.8, a FORTRAN based panel method software package, was used to compare the four different configurations. This code returned aerodynamic data that helped us in the selection of one configuration. A FORTRAN code, AIRFOIL[4], was also used to assist in the determination of the airfoil. The C based "AID2D"[5] was also exercised at this phase.

1.2.3 Preliminary Results

The end of the preliminary design phase came with the determination of dimensions for the wing, tail, boom, canard, and vertical stabilizer based on performance and scoring ability. The parameters were defined on an optimization basis through figures of merit.

1.3 Detailed Design[6]

Once the configuration was roughly tuned, a more detailed evaluation was made on the aircraft. This included final wing parameters, selection of components, and final architecture drawings. Calculations on performances and expectations of performances applied to the competition mission profile were carried out and expressed.

1.3.1 Detailed Design Alternatives

At this point of the design phase, the variables given consideration were, wing parameters such as taper, twist, dihedral, sweep, angle of incidence, and winglets. Spars and rib mounting, spacing, and amount of pieces were studied. The canard's parameters were the setting angle, twist, taper, sweep, dihedral, spars, and rib spacing. The specific placements of parts to include servos, motors, batteries, wires, doors, and hatches. Other areas of concern were with the engine prop sizing and control surface sizing. Expected performance calculations were performed and noted. The materials used for each specific duty, which include structure bonding/attaching and covering. Manufacturing ability, transportation considerations, and cost were also considerations in this phase.

1.3.2 Detail Design Tools

Lifting Line[7], a program that examines the performance of taper, sweep, dihedral, and twist on wings, was used in the detailed design phase to figure out which wing characteristics would maximize the performance of our radio controlled plane. Pro-E, Excel, and AutoCAD were utilized to prepare drawings for manufacturing the aircraft. Also used were several reference books on aircraft design for computations of weight and balance, flight dynamics, g loads, stalling speeds, takeoff and climb performances. Blueprints from previous aircraft were utilized to help us with ours.

1.3.3 Detailed Design Results

The detailed design phase ended with a complete set of construction plans for our aircraft. Details were provided to show location of all sub systems. This also provided a method of manufacturing this aircraft. Included in the detail design are the expected performances of the aircraft in general and specific to the competition mission profile.

1.4 Manufacturing Plan and Process[8]

The manufacturing process consisted of maintaining and fabricating materials for the aircraft. Methods of manufacturing were studied to build a manufacturing plan. The manufacturing plan was then applied to our aircraft. Obtaining a work area, tools, and other supplies prefaced the construction of the most basic pieces. Next the pieces were assembled into major components;

these major component along with prefabricated equipment were then put together to produce the finished aircraft.

1.4.1 Manufacturing Method

The manufacturing process used was that of fabricating pieces to build parts of the aircraft. An order of assembly was observed to ensure that no parts would be left out. Quality control checks, corrections, and recommendations occurred frequently to ensure a quality product and help everyone learn good techniques in the building phase.

1.4.2 Alternative Manufacturing Methods

Options to have parts machined for us or having a more experienced builder come in to show how to perform the assembly were considered. The team had a good grasp on what to do and some trial and error methods were performed on an early prototype. This gave a general understanding on how to cut, glue, hinge, mount accessories, and cover the aircraft. The plans are clear enough to work directly on and no outside help was accepted.

1.4.3 Manufacturing Tools Used

The building process required several tools to include blueprints, razors, sand paper, steel wool, pins, wax paper, rubber bands, rubber gloves, pencils, drill, screwdriver, Allen wrenches, box wrenches, pliers, iron, and heat gun.

1.4.4 Manufacturing Results

The aircraft is expected to be manufactured by following the blueprints created in the detailed design phase. Preparations are being made for the manufacturing lab. The acquisition of materials and parts is in progress. The team members have all been exposed to different manufacturing methods. If the drawings are followed precisely the aircraft should perform well.

2. Management Summary

Nine students from The University of Texas at Arlington set forth in competing in this year's DBF competition. The team "Third floor campers" consists of three upperclassmen and six underclassmen. The management of the design of the aircraft "Black Magic" has been broken down into stages, to change as the development of the aircraft occurred.

2.1 Design Team Architecture

All team members actively participated in each design component of the entire process. At the initial meeting of the team, Kevin Case was selected as team leader. The purpose of the team leader was to maintain order and effectiveness. Along with selecting a team leader at the initial meeting, a schedule was set out and various projects were assigned. Figure 2.1 is a breakdown of each member's assignments. With each phase change new assignments arose and were issued accordingly. The responsibility of each assignment was the teams as a whole. Individuals listed here represent who headed up what areas; the other team members helped where needed.

Name	Phase			
	Conceptual	**Preliminary**	**Detail**	**Manufacture**
Kevin Case	Aerodynamics	Dimensions	Draftsman	Quality cont.
Jason Lee	Rate cost	Press. distr.	Component selc.	Laborer
Stephen Felix	Mech. limit.	Airfoil selection	Subsystems	Quality cont.
Cindy Hollingshead	Real cost	Wing planform	Structural anal.	Laborer
Cecil Bedford	Ground ops	Wing load	Weight & Balance	Laborer
Terry Hankins	Sex appeal	Inactive	Inactive	Inactive
Noah Hicks	Ease of manf.	Flt. dyn.	Handling qualities	Instructor
Hisashi Inoue	Safety	Payload config.	Rate cost	Laborer
Brian Jakubik	Subsystem allot.	Power load	Performance	Laborer

Figure 2.1 Members Leadership Roles

2.2 Structure for Assignments, Schedule, and Configuration Control

A compilation of through records was kept at each meeting in preparation for the final paper. The team leader coordinated semi-weekly meetings held on Tuesdays and Thursdays. At these meetings, the students were able to interact and discuss any issues or successes experienced during their research and/or development. These meetings also provided time to accomplish ant items that must be accomplished as a team. It was the responsibility of the team leader to keep the team on track of the attached scheduled. (See. Fig. 2.2)

2.3 Timeline of Planned and Occurrences Dates

A timeline has been devised to keep team members focused on deadlines and to have an idea of what to expect in the near future. By sticking to the time line the team gets a sense of accomplishment, and starts tasks with ample time for completion prior to the next event. This time line is shown as Figure 2.2.

Task	Start	Finish	Sept.	Oct.	Nov.	Dec.	Jan.	Feb.	Mar.	Apr.
Conceptual Design	09/28/01	12/21/01	***********************							
* Component	09/28/01	10/25/01	^^^^^							
# Component	10/01/01	11/08/01		------------						
* Analytical	10/29/01	12/21/01		^^^^^^^^^^^^^^^^						
# Analytical	11/09/01	01/03/01			----------------					
Preliminary Design	12/23/01	02/05/02				***********				
Construction:										
* Prototype	12/23/01	01/31/02				^^^^^^^^				
# Prototype	01/07/02	02/08/02					--------			
Detail Design	02/07/02	03/07/02						*******		
* Aerodynamics	02/07/02	02/21/02						^^^^		
# Aerodynamics	02/11/02	03/01/02						-------		
* Structures	02/21/02	03/07/02						^^^^^		
# Structures	02/28/02	03/07/02						-----		
Manufacturing Plan	03/04/02	03/31/02							*****	
* Final Construction	03/07/02	03/31/02							^^^^	
Initial Flight Test	04/06/02	04/06/02								*
Evaluation of Flight	04/06/02	04/18/02								***
Documentation	09/28/01	04/20/02	**							
Records	09/28/01	04/20/02	**							
Intent submitted	10/31/01			^						
Paper Preparation	02/07/02	03/10/02						---------		
Paper submitted	03/12/02								^	

* indicates tentative

\# indicates actual

Figure 2.2 Time line schedule.

3. Conceptual Design Phase

In the conceptual design phase, the team members first identified the parameters that were to be used to build the airplane. These parameters were then looked at in depth by a systematic process to include every possible combination. The team scrutinized each configuration with each member paying special attention to his/her respective assigned area. The member's respective areas of interest were as follows: Kevin Case considered aerodynamics; Jason Lee was responsible for rated cost analysis. Cindy Hollingshead gave real cost input, and Terry Hankins accounted for sex appeal. Noah Hicks provided input on the ease of manufacturing the concepts, while Brian Jakubik insured that there was an allotment for all subsystems. Other considerations were mechanical limitations observed by Steve Felix, safety issues by Hisashi Inoue, and ground operations considerations were taken into account by Cecil Bedford. The areas of interest that were given more consideration were the ones that helped to optimize the performance and scoring potential. These areas had a much stronger effect on the decisions than cosmetics or ease of production. Every category was considered and the group democratically voted to keep or discard concepts. After several cycles, the pool of configurations was narrowed down to four. The final configuration was then selected based on a figure of merits.

3.1 Configuration Concepts

To find the optimum performing configuration, several variables were considered upon each other for discussion and selection of four configurations were made based on a pro & con system as the team evaluated each configuration. Each member was given an area of expertise to base his or her case on. The areas of argumentation were aerodynamics, rated cost, real cost, sex appeal, ease of manufacture, subsystems allotment, mechanical limitations, safety, and ground operation considerations. Since there were 6 variables and an average of 8 configurations for each variable we examined 4680 different configurations. By the time the examinations were complete, we had four conceptual designs that we decided were worthy of further examination.

3.2 Conceptual Design Parameters Investigated

Several parameters were chosen to be varied for the selected aircraft. Each parameter combination was selected and critiqued in the respective areas of interest. These variables included wing placement, landing gear style, tail configurations, engine positioning, and cargo bay location. These variables offered a great many configurations to consider. The discarding of configurations came at a strong pace to narrow down the numbers rapidly.

3.2.1 Wing Placement

The decision of wing placement was among three different settings. The three areas of placement included high, middle, and low. Advantages of a high wing are ground clearance, less

fuselage effect on the airflow over the wing. Advantages of a low wing include shorter gear is required and ground operations are assumed to be easier. A disadvantage of a mid wing would be the spar going directly through the body or being two separate pieces.

3.2.2 Landing Gear Styles

Tricycle, bicycle, tail dragger, quad, and single main gear configurations were considered. The tricycle and tail dragger both have three point systems, however, the tricycle has a steering advantage using a controllable nose gear. The tail dragger is subject to momentum during turns that might pose a problem for a ground loop. A bicycle landing gear configuration saves on weight, but leaves instability in the aircraft when not in motion, which can result in wing tip strikes. Quad gear configuration is excellent for heavy loads but is heavy in itself. A single main gear is the lightest solution, however, stability problems arise during ground operations.

3.2.3 Tail Configurations

The tail configuration provided the greatest amount of options. One of the prime concerns with the tail configuration was the rated cost associated with each. A conventional configuration consists of a vertical stabilizer and a low horizontal stabilizer; this is the most common. The Tee-tail may have a lower height due to the horizontal stabilizer acting as an endplate on the vertical stabilizer increasing its efficiency and reducing its length. Its also provides better ground clearance, however, it must be heftier to provide rigidity. A Vee-tail consists of two stabilizers set at angles and use ruddervators. The ruddervators provide a difficulty in control systems and the Vee-tail does not appear to be cost efficient according to the rated cost analysis. An H-tail may be narrow due to endplate effects provided by the vertical stabilizer on the horizontal stabilizer. The tripletail has three vertical stabilizers and has a negative effect on the cost efficiency. The inverted Vee suffers similar problem to the Vee-tail and to worsen this, ground clearance is also lost. From an aerodynamic stand point the inverted Vee is very effective, provided ground operations don't have to be considered. The twin tail has two vertical stabilizers, which will be less in length than a single, but according to cost analysis, it isn't the most efficient. The Y-tail will be a very expensive configuration, according to the rated aircraft cost, and each surface will be full length as there is no endplate effect on any surfaces. Twin boom tail works on the assumption that there is a twin boom, the difficulty in manufacturing a twin boom warranted a single boom. This also implies to the boom-inverted Vee, which would cost more than the twin boom. A ringtail is a very effective tail as each surface enjoys the benefits of endplate effect. The ground clearance is lost and manufacturing difficulty is an issue with the ringtail. Canard configuration will provide a horizontal stabilizer that will provide an up force instead of the down force experienced on conventional tails to counteract the wing pitching moment. Spoiling the airflow in front of the wing is an issue, however, in a stall the canard does not get blanketed, which is a recovering characteristic. Another good characteristic of the canard is the positive

lifting force produced in order to counteract the moment of the wing; whereas with a conventional tail setup a down force is produced. Finally, a tailless configuration is considered which leaves the pitching moments of the wing to be counteracted by control surfaces on the wing and a weight transfer to balance within the wing. The tailless configuration cuts the cost of extra vertical and horizontal surfaces.

3.2.4 Engines

The amount, direction, and placement of engines were considered. There were problems with multiple engines with yaw control, particularly in an engine-out situation, which can be provided for in foresight in the design process by keeping the thrust forces close to the centerline of the aircraft. Multiple engines means increased thrust at the penalty of weight, and cost. Structural considerations would have to be made for each engine mount. The engines work best when the prop gets clean air therefore, makes the pusher type engine less desirable. Mounting the engines on top of the wing help provide ground clearance.

3.2.5 Fuselage

The fuselage primarily had the function of holding the cargo and any subsystems of the aircraft that cannot be mounted elsewhere. The size of the payload dictates the size of the fuselage. Further aerodynamics provides shape of the fuselage. Pods were also considered for the transport of cargo. Pods were deemed undesirable as they offered more drag than a cargo holding fuselage, or internal wing cargo hold. Wing internal payload has been constrained to only the flying wing configuration.

3.3 Conceptual Numerical Figures of Merit

The figures of merit were based upon the individual analysis of each team member. Each team member analysis was a rough estimate and more importantly, the analysis on more important aspects such as aerodynamics was weighted heavier than trivial aspects such as sex appeal. Many planes scored well, however, the four that posed the most interest were subjected to further analysis. Other considerations that were put into this were added interest from others including school faculty. A breakdown of the percentage weight of each aspect and scores for the four highest configurations are displayed in Figure 3.1

Value Allocation	% Weight	High wing T tail	Low wing T tail	Flying Wing	Canard
Real Cost	25	23	25	25	24
Manufacturing	15	14	14	15	13
Aerodynamics	14	12	12	10	14
Mechanical Limitations	12	10	9	9	12
Ground Handling	10	6	10	10	10
Safety	9	9	6	8	8
Rated Cost	6	5	5	6	5
Design Appeal	5	3	2	5	5
Systems Allotment	4	3	2	4	2
Total Score of 100	100	85	85	92	93

Figure 3.1 Figures of Merit

3.4 Conceptual Rated Aircraft Cost

The aircraft rated cost is an estimation of cost based on projected figures. This rated aircraft cost is a divider of the product of the written report score and the flight score to give the total score. It is apparent that the lower the rated cost the less harmful effect on the total score. At this phase of the design, rough estimates were used to make calculations for the rated cost. The formula used to find the rated cost has been adopted from the guidelines of the contest. The following formula was used.

R.A.C. = (A*MEW + B*REP + C*MFHR)/1000

Where: R.A.C. = Rated Aircraft Cost

A = $100

B = $1500

C = $20

MEW = Manufacturers Empty Weight

The manufacturers empty weight is the weight of the aircraft in pounds, not including payload.

REP = Rated Engine Power

The rated engine power is found to be:

REP = (1-0.25*(number of engines – 1))* total battery weight

MFHR = Manufacturing Man Hours

The total battery weight is in pounds and only applies to batteries used for propulsion. This weight is not to exceed 5 pounds. The manufacturing man-hours are a projected amount of man-hours to manufacture the aircraft based on the component's dimensions and amount of complex parts. The man-hour breakdown is as follows. Each foot in the wing span requires 8 hours; the maximum chord length cost 8 hours per foot. Each control surface requires 3 hours; the maximum length of the aircraft goes for 10 hours per foot. 5 hours are used for each vertical surface unless it has a controlling surface on it, in which case, 10 hours are required. Any horizontal surfaces that are not considered wings are 10 hours for the surface. A horizontal surface is concurred a wing if exceeds one forth of the maximum wing span. A Vee-tail comes at the expense of 15 hours. Each servo, motor controller, engine and propeller require 5 hours in labor to fabricate. From these values a rated aircraft cost can be achieved. In this early design phase, many of the values are estimated. The rated cost was part of the decision making process in retaining and eliminating conceptual ideas for the aircraft. A cost analysis is also exercised on the final design of the aircraft. Our rated aircraft cost for the selected aircraft are as follows in Figure 3.2.

	High Wing Tee tail	Low Wing Tee tail	Flying Wing	Canard
RAC (X $1000)	**18.735**	**18.495**	**17.387**	**18.83212**
Mew	43	43	43	43
REP	6.25	6.25	6.25	6.25
MFHR	253	241	185.6	257.856
WBS wing	108	96	85.6	87.856
WBS fus	70	70	40	80
WBS emp	20	20	10	30
WBS fs	35	35	30	40
WBS ps	20	20	20	20
Empty weight (lb.)	43	43	43	43
# Eng	2	2	2	2
Bat weight (lb.)	5	5	5	5
Span (ft)	10	8	5.7	7
Chord (ft)	2	2.5	3.5	2.857
# Wing con suf	4	4	4	3
Max length (ft)	7	7	4	8
Inact vert	0	0	2	0
Act vert	1	1	0	2
Hor surf	1	1	0	1
Vee tail	0	0	0	0
Servo/esc	7	7	6	8
# Prop	2	2	2	2

Figure 3.2 Conceptual rated cost

3.5 Significant Assumptions of Conceptual Phase

In the conceptual design phase, several assumptions were made. The first thing we assumed was that regardless of the configuration selected, a loaded weight of 55 pounds would be observed. This would ensure that our configuration would be evaluated at the maximum allowable weight, as 55 pounds is a limiting factor of the competition and provides a worst case scenario. We wanted to get the most points available due to number of softballs; therefore, we assumed that our configuration would carry the 24-ball maximum. We estimated a softball to weigh half a pound; thus we planned for a load of 12 pounds which in turn meant that the empty weight of the plane weighed 43 pounds. Then we assumed that it would be absolutely necessary for us to use the entire 5-pound maximum permitted for batteries. All of these weight assumptions are limitations that we could not afford to exceed. It was also assumed that we would be able to physically manufacture the aircraft. Since our University has been dormant in the AIAA DBF competitions in the past few years, it would be difficult to obtain sponsors without good detailed plans. With this in mind we assumed that a limited budget would also dictate our spending on materials and testing capability. This also made us rely on the theories heavier as we had in essence one shot to get into the competition. Since several members were more interested in getting the school active in the intercollegiate activities, and the majority of the team members had little to no experience in designing, building, or flying RC aircraft. It was also assumed that extravagant designs were less valuable to the team than more functional ones.

4. Preliminary Design

The main purpose of the preliminary phase of the design process was to determine the general sizing of the wing and canard airfoils based on performance optimization. The first step taken in this phase was the identification of the parameters by the team members. These parameters where then examined in depth for optimization. Each member headed up investigations in the following areas: dimensional variables by Kevin case, pressure distribution by Jason Lee, airfoil selection by Steve Felix, wing planform by Cindy Hollingshead, wing loading by Cecil Bedford, flight dynamics by Noah Hicks, payload configurations by Hisashi, and power load by Brian Jakubik. These parameters were exercised to optimize the performance and scoring potential. These areas were subject to calculations and analysis programs. This resulted in the selection of the dimensions of the canard configuration.

4.1 Preliminary Design Parameters and Sizing Trades

The preliminary design phase included a variance in dimensional and performance parameters. A change in one parameter may improve one aspect yet have adverse effects to other parameters. For instance, the pressure distribution using a selected airfoil, may warrant the wings to be longer. This results in the chord length to decrease to maintain the wing area. However, this might not fit as nice on the payload bay, and require an increase in structural support to maintain the wing load. The dimensional results of the code were also compared to that of similar existing aircraft to decide if they were reasonable. The process of parameter selection went back and forth to find a happy medium for the studies. The team selected the pecking order of importance for each parameter. This ranking system was used to find the best dimensional configuration. Satisfaction for each study area was accepted in proportion to the ranking systems weight of influence.

4.1.1 Use of AIRFOIL Program

The AIRFOIL program is a FORTRAN code developed by one of the team members to perform analysis on four digit N.A.C.A. airfoils. The code evaluates the airfoils at a range of angles of attack, and freestream velocities. This code is based on two-dimensional steady flow. The use of the panel method is incorporated along with the Runge-Kutta condition to produce outputs of coefficients of lift, drag, and moment. This code has been validated for thin airfoil theory.

4.1.2 Use of KK-AERO Program

The FORTRAN based KK-AERO program was provided to the team for analysis. Krzysztof Kubrynski wrote KK-AERO. There is no validation provided, so the output was taken with suspicions, considering the limitations of the code were not apparent. The input to the code is of a three-dimensional drawing that undergoes a panel method to find pressure distributions. An

option to the output is to have the code recommend a body shape to produce a desired pressure distribution. Had a validation been offered or derived, this code would have been relied upon more heavily.

4.1.3 Use of AID2D

AID2D is a C based inverse design program to determine pressure distributions with respect to an input shape. A database of common airfoils is stored in the program to draw them up and represent an ideal pressure distribution. The validation of this program was made readily available and showed that the code required low angle of attacks. This was a constraint to keep in mind with the use of the code.

4.2 Preliminary Figures of Merit

Each member either used calculations or code to produce rough values for numbers of importance in their respective areas of study. Many opted to form Excel spreadsheets for quick value comparison. Each of these members devised their own ranking system for the results of the configurations. The areas of interest were then applied to the individual ranking system. This resulted in an overall figure of merit system for the preliminary design phase. These rankings were obtained by considering each item's effect on flight performance, ease of construction, and design scoring ability. The figure of merit percentage weight is as follows in Figure 4.1.

Attribute	Weight %
Flight Dynamics	20
Airfoil Selection	19
Wing loading	15
Total Pressure dist	14
Wing planform	12
Power Loading	11
Dimensions	7
Payload config	5
Total %	100

Figure 4.1 Figures of Merit-Preliminary

4.2.1. Flight Dynamics

Flight dynamics was chosen as the top priority of design because the aircraft's handling and flight characteristics would be the most critical characteristic of our design. How the airplane flies (pitching moments, axial stability, rates of climb) with the given mission requirements will be the main feature of the design's success.

4.2.2. Airfoil Selection

The airfoil has a wide number of parameters that critically effect flight performance. The airfoil is very close in importance to flight dynamics because the airfoil contributes very much to flight dynamics. We used the program AID2D and the program AIRFOIL as tools to assist us in selecting an airfoil.

4.2.3. Wing Loading

Wing loading is critical to the flight performance. However, the challenge of this design, to make an aircraft that can carry the required payload with minimal power allowed, makes a very low wing loading the obvious choice. Thus, while wing loading is very important, it is a simple requirement to fulfill, simply choose the right wing area.

4.2.4. Total Pressure Distribution

The total pressure distribution effects the coefficients of lift, drag, and pitching moments. We choose to prioritize this very close to wing loading because they are very similar. The pressure distribution over the fuselage on this slow flying aircraft is of much less importance than the pressure distribution over the wing. The pressure distribution on the wing will be a major determinant of wing loading. The use of the KK-AERO program assisted in this area.

4.2.5. Wing Planforms

Wing planform will effect the pressure distribution and thus the flight characteristics, but it could not be rated amongst the most important due to the limited manufacturing abilities. Restrictions applied due to bounded building experience even though an elliptical wing would give greater performance, it would be cost prohibitive.

4.2.6. Power Loading

Power loading gave good comparisons to similar aircraft, considering the engine sizing of the aircraft dimensions. With out the selection of an engine at this point in the design an average power of the allowable engines was used for formulation. The power loading has a lower value for aircraft with greater power to weight ratios.

4.2.7. Dimensions

The overall dimensions are quite easy to approach. An aircraft that meets a general standard of proportionality must be fabricated. This proportionality was based on a historical datum. Dimensioning was given a low rating because radio controlled aircraft typically do not share the same proportions of their full size counterparts.

4.2.8. Payload Configuration

The payload configuration was the least important since there were only three combinations that could have been used. Being this limited with all three choices almost equal in this quality, it was found to be the least important consideration.

4.3 Preliminary Wing Loading

The wing loading is found as the weight divided by the reference wing area. The reference wing area is a function of the fuselage shape. With a 55 pound loaded weight and optimized wing area according to our figures of 1296 square inches, a wing loading of 0.424 pounds per square in was obtained. This converts to 6.111 pounds per square foot, which is very reasonable in comparison with typical wing loading of sailplanes, which is 6 pounds per square foot.

4.4 Preliminary Power Loading

The power loading is the aircraft weight divided by the horsepower. Radio controlled aircraft engines are power rated in Watts. The direct conversion of watts to horsepower was used to provide the power rating of the engines for further analysis. The conversion is one Watt is equal to 0.00134102 horsepower. Propeller efficiency is also factored into this process. It is multiplied by the power of the engine to produce a thrust power. According to our calculations, the power loading of this aircraft comes to 32.5 pounds per horsepower. We found this to be quite high in comparison to typical powered sailplanes, usually having a power loading of 25 pounds per horsepower. In the case of power loading a higher value is found to mean that a smaller engine is in use.

4.5 Assumptions of the Preliminary Phase

Assumptions during the preliminary phase included standard day conditions. This assumption was used, as the conditions during the flight for competition cannot be guaranteed as they depend on the weather. As an aircraft built will suffer a wide range of conditions, a standard day provides an average of the conditions. The use of a maximum weight of 55 pounds was assumed, as this will not be exceeded due to the competition and ASA regulations. A propeller efficiency of 0.9 was assumed for computation sake. The propeller might not actually have such a good value, but a range of propeller efficiencies from 0.85 to 0.92 has been observed for current propellers. 0.9 was arbitrarily pick as a midrange representative of the propeller to be used. Flight speed of 35 feet per second was assumed from previous model aircraft and outside interviews with radio control pilots, engine power rating is assumed to be 700 watts. The electric engines were found to have wattage range of 400 to 1000 watts, since our exact engine had not been selected by this point a mid range value was selected.

4.6 Comparison of the Preliminary Phase with conceptual phase

The preliminary phase differs from the conceptual phase as it took a more in depth look at particulars in sizing and shaping of the main components of the aircraft and took a less look into general configuration. A more scientific approach was used for reasoning. The trade study's parameters were more interrelated to each other in the preliminary stage than those of the conceptual phase. This required the team to work more closely with each other. Data and feedback were constantly being exchanged between the team members to acquire pleasing results on a percentage basis to each standpoint of the figures of merit. The result based on this more technical approach provides a higher comfort level among the team than the more arbitrarily results obtained in the conceptual phase.

5. Detail Design Phase

Different materials, carbon-carbon composite, monokote, prefabricated parts, lexan, glue, rubber bands, balsa, aluminum, resin, lock nuts, and epoxy were considered to minimize weight and increase performance. In a final attempt to tweak the aircraft to the maximum performance, more parameters were studied in the areas of control sizing, twist dihedral, sweep, incidence, winglets, spars, and rib spacing. The structure of the boom, suspension, tires, steering controls, the specific placement of servos, electronic speed controls, motors, batteries, wires doors, and hatches were also of consideration. This design phase is the last of the optimizations, and the beginning of the construction. Plans were drawn up to builders' specifications, and final performance values were theoretically solved for and are to be expected.

5.1 Detailed Component Selection

Components used in this aircraft were selected on individual circumstances and reasoning. Weight and volume were two primary concerns for the items located inside the wings and fuselage, power also played a role for the servos, engines and batteries. Landing gear was selected on the basis of ground clearance and tip over. Tires were selected to distribute weight and have a large diameter to ease the ground roll.

5.2 Detailed Systems Architecture

The architectural layout is designed for function. The primary focus for the layout is the structural members and how they are fastened to each other. The systems are then set into areas that enable them to work around the structural members. Systems are also positioned to locations that give them the most effectiveness for control, and access. The systems are also arranged to help with the center of gravity position in both loaded and unloaded configurations.

5.3 Detailed Final Performance Data

With the competition of all dimensional data, performance data is obtained through formulas on to give an expected value for selected performance characteristics. These expected values hold meaning to validate the theory behind the formulas and to show how assumed values could adversely effect the precision compared to observed values. The parameters solved for are the take off distance, handling characteristics, g load and mission performances. Weight and balance worksheet is also constructed for the loaded and unloaded regimes of flight.

5.3.1 Take Off

The ground roll distance is found through integrating the velocity divided by the acceleration. The ground roll can be found through equation 5.1 through 5.3

$$Sg = 1/2g \int_{Vi}^{Vf} \frac{d(V^2)}{Kt + KaV^2} \qquad (eqn.5.1)$$

Where

$$Kt = \left(\frac{T}{W}\right) - \mu \qquad (eqn.5.2)$$

$$Ka = \frac{\rho}{2(W/S)}\left(\mu Cl - Cdo - KC^2 l\right) \qquad (eqn5.3)$$

According to our calculations, a ground roll of 168 feet is expected in a no wind situation. Takeoff into the wind is common practice and is a benefit to the aircraft. Takeoffs into the wind will shorten the ground roll distance.

5.3.2 Handling Qualities

Velocities, coefficient of lift, and drag are variables that change under certain configuration. Each of these is found for both the minimum drag and the minimum power at straight and level flight. The formulas for these variables are shown in equation 5.4 through 5.9. The maximum cruise speed was found to be 44 feet/sec.

$$V_{\min drag} = \sqrt{\frac{2W}{\rho S}\sqrt{\frac{K}{C_{Do}}}} \qquad (eqn.5.4)$$

$$C_{L\min drag} = \sqrt{\frac{C_{Do}}{K}} \qquad (eqn.5.5)$$

$$D_{\min drag} = qS(C_{Do} + C_{Do}) \qquad (eqn.5.6)$$

$$V_{\min power} = \sqrt{\frac{2W}{\rho S}\sqrt{\frac{K}{3C_{Do}}}} \qquad (eqn.5.7)$$

$$C_{L\min power} = \sqrt{\frac{3C_{Do}}{K}} \qquad (eqn.5.8)$$

$$D_{\min power} = qS(C_{Do} + 3C_{Do}) \qquad (eqn.5.9)$$

G = (T-D)/W (eqn. 5.10)

$$\dot{\psi} = \frac{W\sqrt{n^2-1}}{(W/g)V} \qquad (eqn.5.11)$$

The climb gradient of this aircraft is found to be 17.3 ft/sec. This is obtained through the equation 5.10. The rate of turn was found using the equation 5.11.The rate of turn is found to be 0.7 rad/sec

5.3.3 <u>Capability of g-Load</u>

With a factor of safety of 1.2, a g-load of 2.7 can be applied to this aircraft. This load factor is the maximum lift divides by weight. Thus as the aircraft is to be 45 pounds the maximum lift is 121.5 pounds found through a critical angle of attack of a flap down configuration and at top velocity. The maximum g-load is a function of the cross sectional area of the spars, Their depth and width. This structural problem yields special attention to the spars and attachment to the spars of the body.

5.3.4 <u>Balanced Field Length</u>

As any twin engine this aircraft has a balanced field length. The balanced field length is the distance of remaining runway required to bring the aircraft safely to a stop in the event of an engine loss at the most critical moment in the takeoff run. There are no brakes on the aircraft so it must rely on friction to come to a stop. Through the equation below, it is found that the balanced field length is 106 feet.

$$BFL = \frac{0.863}{1+2.3G}\left(\frac{W/S}{\rho g C_{Lc\lim b}} + h_{obsticle}\right)\left(\frac{1}{T_{av}/W - U} + 2.7\right) + \left(\frac{655}{\sqrt{\rho/\rho_{SL}}}\right) \qquad (eqn.5.12)$$

5.3.5 <u>Predicted Mission Performance</u>

During the flight for the competition certain flight performances are estimated to be observed. Takeoff in less than 200 ft. will be observed as the ground roll expected is 168 feet. a healthy climb rate of should have the aircraft up to 100 ft. in altitude in short order of 5.7 sec. With a maximum cruise speed of 44feet per second and a distance of 2000 feet to travel per circuit it is estimated that the aircraft will take approximately 45.5 seconds to complete a circuit unloaded, and 57 seconds. loaded. If a 360-degree turn is also required at mid circuit, the unloaded time will extend to 1.15 minutes per circuit and the loaded configuration will extend to 1.3 minute per circuit. Due to the ground ops the loading and unloading of the aircraft will take a minimal amount of time at 45 seconds each. In total each flight attempt is estimated to take about 7 minutes and 10 seconds to complete.

5.3.6 Weight and Balance Worksheets

The weight and balance is one of the most critical aspects to the handling and performance of the flight. The center of gravity is found by multiplying the weight stations by a moment arm defined and constant. This summation produces moments and is ideally set to help keep the aircraft from tending to pitch undesirably. By keeping the center of gravity located between the fore and aft limits of the static margin insure stability in flight. The center of gravity is located 8 inches in front of the main tires in a static condition and a loaded condition moves the center of gravity back by 4 inches, making the aircraft less stable longitudinally and increasing the stall speed. These values are found to be acceptable due to the static margin range that is 12 to 3 inches from the rear tires. These values have been found through equation 5.13.

$$K_n = (h_n - h) \qquad (eqn.5.13)$$

5.3.7 Assumptions in the Performance Predictions

For the performance of the aircraft, assumptions need to be made. The group considered ideal conditions consisting of an absence of wind conditions. By having no wind conditions, the performance of the aircraft will not be as efficient as it would be for into wind conditions. There is no guarantee that wind will be present, hence with no wind assumption is err is on the side of safety. Standard day conditions are also an assumption based of the reliability of the weather. By using standard day conditions the performance is expected to vary with changes with the rise and fall of the barometer and mercury. Using the small angle approximations helped to keep the angle of attack from exceeding critical limits. The acceleration due to gravity is considered constant with altitude as the range in altitude is significantly small and can be neglected.

5.4 Detailed Rated Aircraft cost

Our final design dimensions and components selected give us a refined rated aircraft cost. Applying these values to the previously expressed code written for the rated cost provides the following shown in Figure 5.1.

	Final Design
RAC (X $1000)	**17.631**
Mew	33
REP	6.25
MFHR	247.8
WBS wing	99.8
WBS fus	68
WBS emp	20
WBS fs	40
WBS ps	20
Empty weight (LB)	33
# Eng	2
Bat weight (LB)	5
Span (ft)	9.255
Chord (ft)	1.72
# Wing con suf	4
Max length (ft)	6.8
Inact vert	0
Act vert	2
Hor surf	0
Vee tail	0
Servo/esc	8
# Prop	2

Figure 5.1 Final design rated aircraft cost

Note that there is no horizontal surface in this analysis, this is a result of the canard exceeding the 25% of the wing span The canard is 0.2807 % of the wingspan. The wingspan Is 7.23 feet, while the canard is 2.03 feet, thus it's length was added to the wingspan and it's controlling surface was accounted for as another controlling surface on the wing. As a result there are no horizontal surfaces accounted for in the cost rate analysis.

5.5 Comparison of Detailed to Conceptual and Preliminary Phases

The detailed design phase is the most involved phase of the three. A finalized product is the result in this phase making the calculations even more critical than the previous two phases. The attributes are considered at a higher level of criticism, and at more involved analysis of performance is achieved at the detailed design phase. Discussions made in the detailed design phase are made on statistical information and rely less on personal opinion, or outside influence.

Every decision made in the design phase is based on numerical data to optimize the performance and scoring ability of the aircraft.

5.6 Detailed Drawing package

The drawing package includes a 3-view drawing of "Black Magic" in sufficient detail as shown in figure 5.2. The detail indicates aircraft size and configuration. Included in this package are drawings of primary structure component size and location: the payload size, location, and restraint method is shown. The location of the propulsive and flight control systems components can also be noted in figures 5.3 and 5.4.

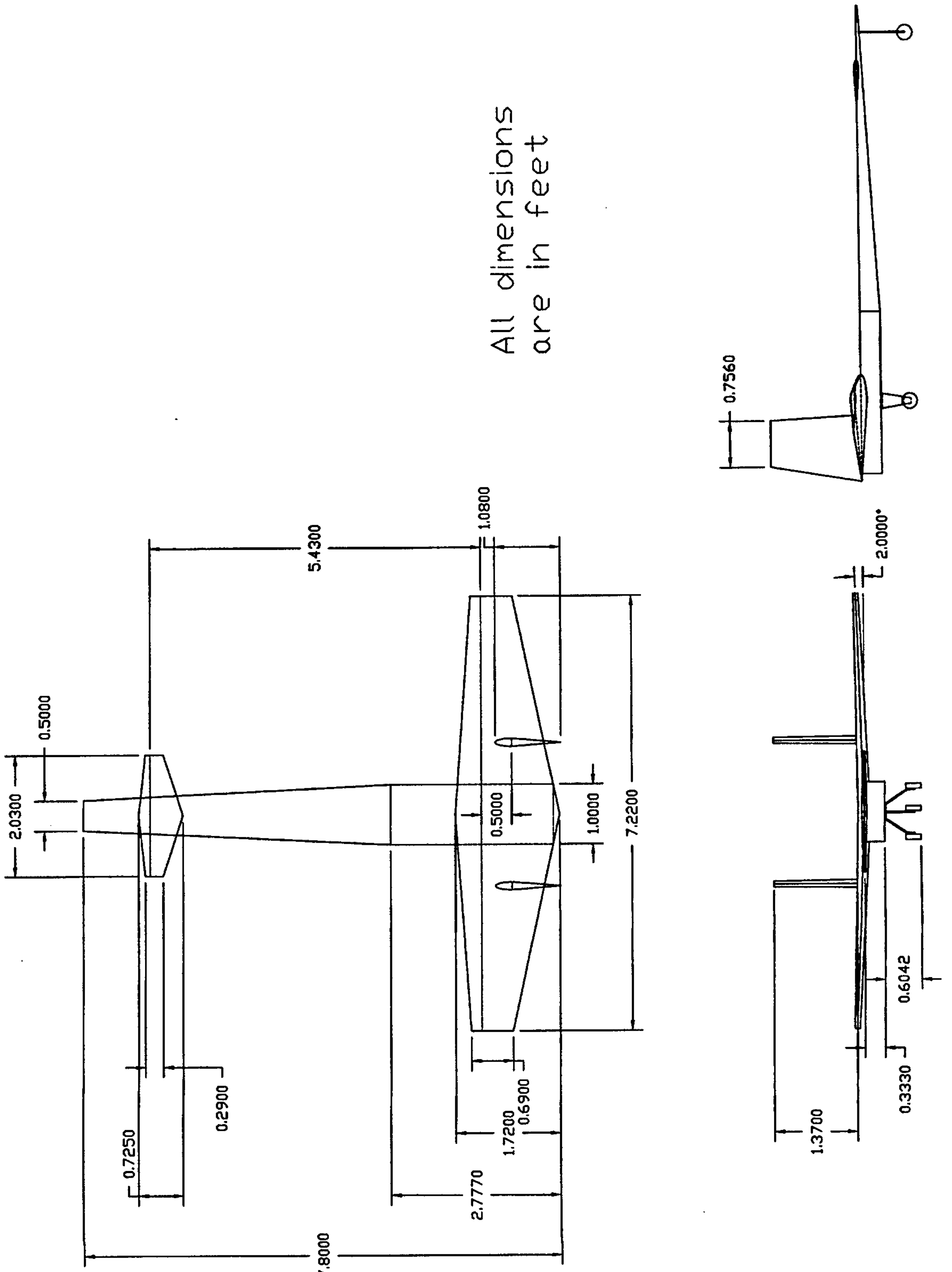

Figure 5.2 (3 view)

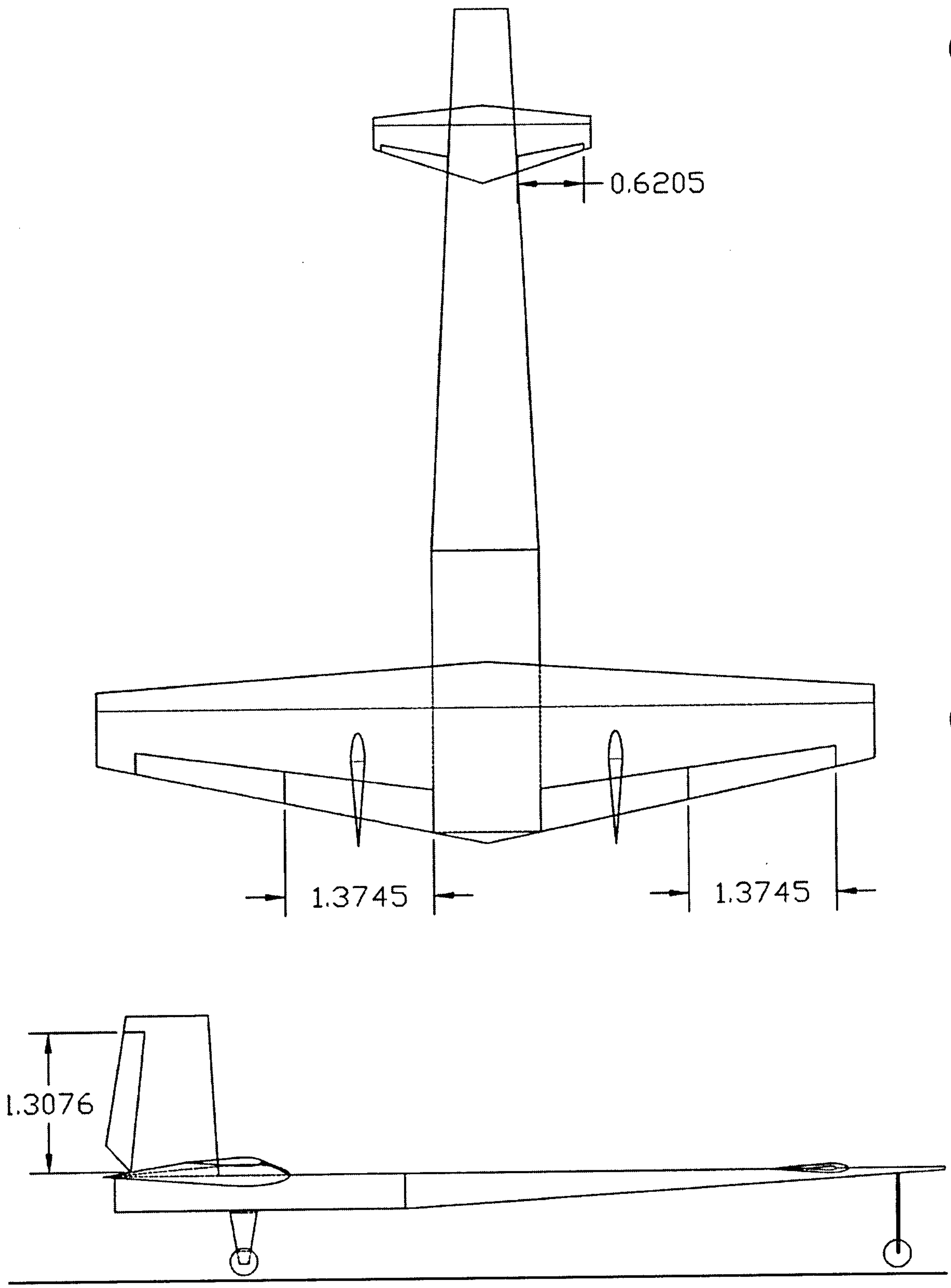

Figure 5.3 (control surfaces)

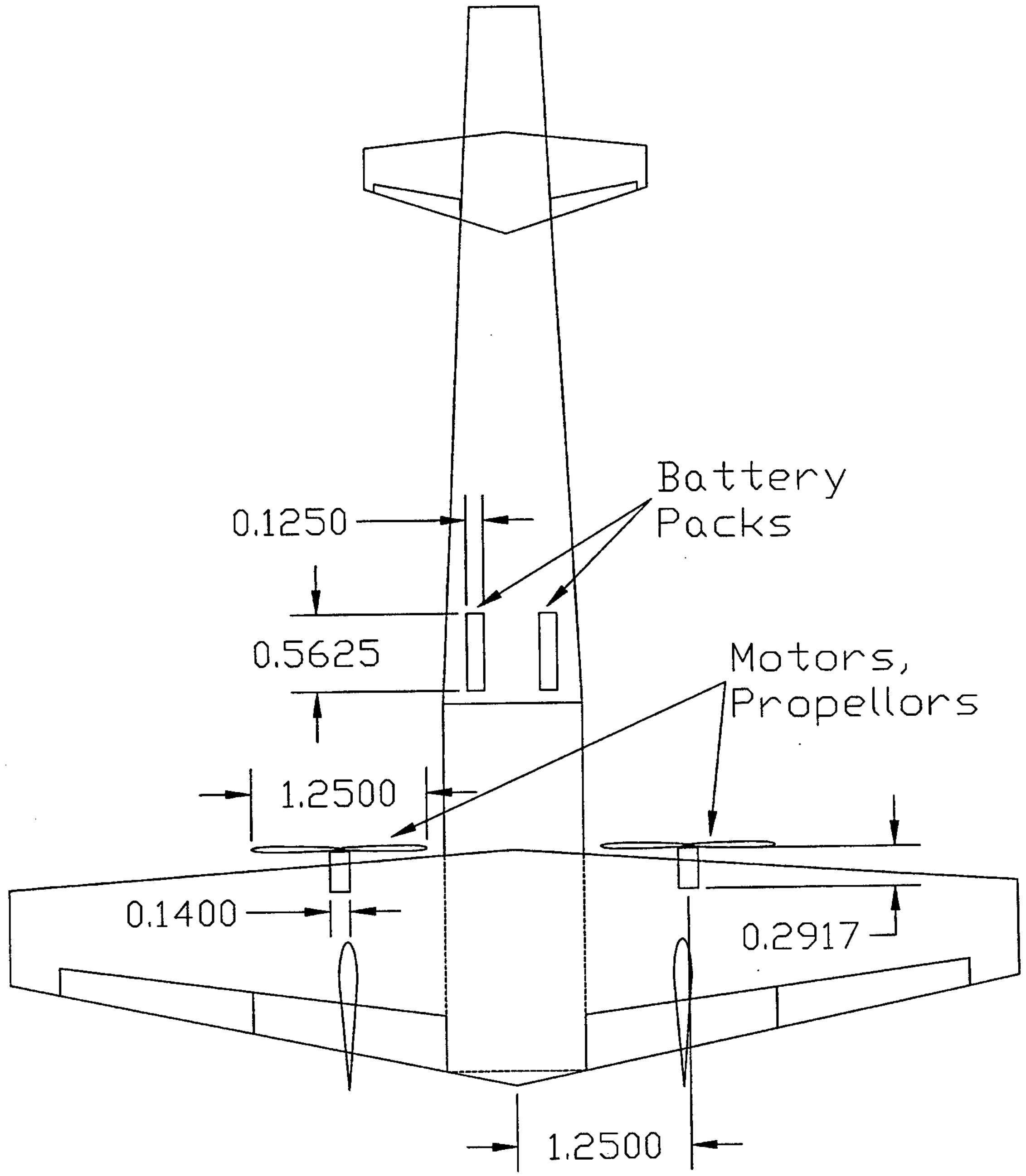

Figure 5.4 (component layout)

complicated assembly, but we thought that this would again be easier than fiberglass because wood is much less sensitive to small assembly errors. From our analysis of these two methods, we determined wood would suit our purposes best.

6.2 Selection of Processes for Major Components and Assemblies

For the major parts of the aircraft we selected different construction techniques. The fuselage had to be strong enough to support the flight loads of the wing and landing gear. For this we selected a standard assembly using plywood bulkheads, and balsa sheeting. Our design plans to have reinforcing plywood at the attachment points of the landing gear, wing, and tail boom. Wing construction follows the same technique used since the Wright brothers, ribs and spar with thin covering. For our wing, balsa ribs will be cut from templates made with AutoCAD. The spar will be constructed using a shear web with a balsa shear web and hard wood strips capping the shear web. The tail surfaces are quite easy to make. They use a simple built-up structure of balsa spars. The canard is constructed in a manner similar to the tail.

6.3 Final Design Assembly

The manufacturing process was divided up into several phases. A date was set for a deadline for each part completion, this is in the table later. These phases must be completed in an order that does not permit the omission of any important items, i.e. wings are not skinned until servos are mounted, etc. Final manufacturing process included the use of computers for complex shapes. The aircraft would be constructed of wood, with several items purchased due to the complexity of fabrication and competition requirement. Electronics would be purchased, but the crew would perform the assembly of the control linkages. The team would then cover the completed frame with all installed components. Bolt on fasteners were used where appropriate, such as the landing gear and mounting the motors.

6.4 Manufacturing Figures of Merit

The skill level of the team members is not necessarily top notch, several models were built roughly so that the members can have practice with the methods used. Inclusion of all team members was desired in this process so participation was encouraged. The members with the most experience in this field were deemed Teachers to the rest. Quality control management was assigned to the team members who were most familiar with the details involved with the layout and planing and drawings. The figures of merit obtained from our analysis of manufacturing methods have been given a ranking of importance displayed in Figure 6.1.

6. Manufacturing Plan and Process

The manufacturing process had to be well planned before it could be done. Analysis of available methods was used to determine which process to implement. Everything from the construction, environment, materials used, and skill level of process, to cost was considered.

6.1 Manufacture Process and Selection of Materials

The manufacturing process was largely determined by the construction materials to be used. For this project the materials were limited by our skills more than any other factor. Our fundraising limiting us also to economical construction technique desired. We gave consideration to the two main model aircraft construction methods: traditional wood construction and composite construction. Composite construction might have afforded us a better strength to weight ratio, but our lack of skills may have required one or two practice assemblies. Traditional wood construction made construction more practical though it cost us in performance. We made a detailed analysis of the both wood and composite construction. From this we determined that wood was the best option for our building type. Our scheduling timeline has been setup to give us goals to meet during the design, building, and eventual testing and competition.

6.1.1 Composite Construction

We gave composite construction considerable thought. Our research came up with several findings. Composite materials offer a very good strength to weight ratio. However, composite construction would have required specialized construction techniques such as the use of resins, foam, and fiberglass, due to the nature of these materials simple mistakes could have significantly detrimental results. For example, we discovered from our research that when doing a fiberglass cloth lay-up on a wing core, unevenly applying the resin could result in bubbles under the cloth and nullifying the strength advantages of composite material. Furthermore, we discovered the importance of alignment when doing a lay-up. If the cloth is pointed in the wrong direction or even has a small amount of waviness, the strength of the lay-up would be significantly reduced.

6.1.2 Wood Construction

The advantages and disadvantages of wood construction were more apparent than those of composite were. Wood construction is much more simple than composite. The pieces are simply cut and assembled. Nothing further is required. As long as care is taken to ensure that the glue joints are strong, one can assume the structure will perform as desired. Composite construction combines the compressive strength of foam with the tensile strength of fiberglass to create a very strong structure. With wood, we have a material that is stronger than foam in compression but weaker than fiberglass in tension. We knew, however, that wood could be made very strong if the structure was designed correctly. To make a strong wood structure requires a very

Attribute	Weight %
Availability of supplies	45
Cost	20
Availability of tools	15
Skill level	10
Organized & clean workspace	5
Computer assistance	5
Total percentage	100

Figure 6.1 Figures of Merit-Manufacturing

A skill matrix was constructed to define the amount of team members who were comfortable with areas of construction. This was used to help determine the method of manufacturing. This is shown in Fig. 6.2.

Required Skills	Wood Working Skills	Skills with Lathe & Drill	Machining	Sanding	Balsa/Plywood Framing	Carbon/Fiberglass Reinforcement	Wiring	Application of Monokote
Number of Personnel	6	3	2	3	4	5	2	4

Fig 6.2 Skills Matrix

6.5 Analytic Methods of Manufacture

We analyzed several different plans and procedures from RC kit aircraft to obtain construction methods. This starts with ensuring all the required parts are made or purchased and developing a method of assembly. We researched the methods used in labs and machine shops to determine fabrication methods. We looked at educational materials such as the classic video "Building the Rutan composites" to familiarize ourselves with composite construction. From the labs and videos we learned that a clean and organized workspace would be critical for efficient assembly. We also determined that a certain skill level is required to manufacture certain products such as lay ups. Having all the tools available and functional was noted to be of prime importance to the ease of production. We investigated computer programs to assist us with fabrication and assembly of complex parts. One method we considered implementing was using

a computer program such as AutoCAD to give use a scaled template for ribs and bulkheads. This has the potential to save us much time and effort in complicated measurements. We found that cost of materials and skill level required for some manufacturing exceeded our limits and thus were omitted.

6.6 Time Line of Manufacture

The manufacturing timeline is used to maintain dead lines and provide the members with a task so that there is no question as to what to do next. The proposed schedule is provided in Figure 6.3.

	March 7			March 10							March 17							March 24							
Tasks	T	F	S	S	M	T	W	T	F	S	S	M	T	W	T	F	S	S	M	T	W	T	F	S	S
Wings	X	X	X	X	X	X	X	X	X	X	X														
*Spars	X	X																							
*Ribs	X	X	X	X																					
*Assembled					X	X	X	X	X	X															
*Motor Mounts											X														
Fuselage												X	X	X	X	X	X	X							
*Cargo Bay												X	X												
*Canard Boom												X	X	X	X										
*Assembled													X	X	X	X	X	X							
Canard	X	X	X	X	X	X	X																		
*Spars	X	X																							
*Ribs	X	X	X																						
*Assembled				X	X	X	X																		
Vertical Stabilizers							X	X	X	X	X	X	X	X											
*Spars							X	X																	
*Ribs							X	X	X																
*Assembled										X	X	X	X	X											
Systems Integration																		X							
*Wiring of Motor, Servos, Batteries, Fuses & Receiver																		X							
*Mounting Motors and Servos																		X							
*Securing of Batteries and Receiver																		X							
*Attaching Propellers																		X							
Control Surfaces (Construct, Attach, & Connect Linkage)																			X	X	X	X	X		
*Ailerons																			X	X	X				
*Flaps																			X	X	X				
*Rudders																					X	X	X		
*Elevators																					X	X	X		
Landing Gear (Construct & Attach)																								X	X
Covering (Sizing & Application)																									X

Figure 6.3 Proposed Manufacturing Schedule

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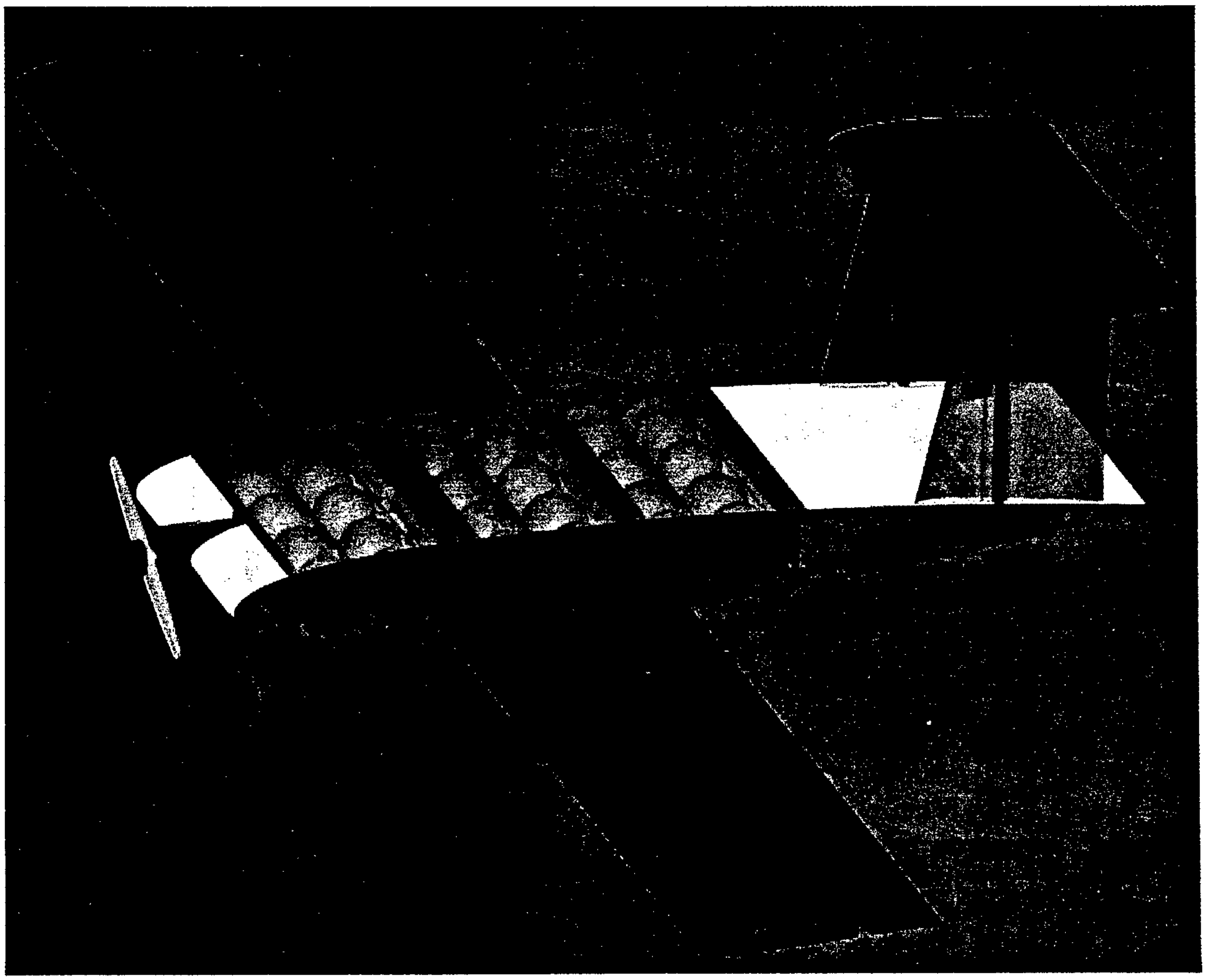
AIAA Student Design/Build/Fly Competition

Better Is the Enemy of Good



By:

The University of Texas at Austin
2001/2002 Design/Build/Fly Team

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1.0 EXECUTIVE SUMMARY

Students at The University of Texas at Austin's Department of Aerospace Engineering and Engineering Mechanics have designed and built an electric RC airplane in order to compete in this year's Cessna/ONR Student Design Build Fly Competition. This years mission profile involves flying both loaded and unloaded sorties around a designated course for a total of 6 laps. The goal of the aircraft is to carry the greatest number of softballs in the least amount of time while minimizing the Rated Aircraft Cost. The aircraft was designed in three phases: conceptual design, preliminary design and detail design.

The design process began in conceptual design in which the team eliminated general configurations from consideration. Then, in preliminary design, simplifications and approximations were made to certain design parameters to allow the aircraft to be mathematically modeled in the optimization code named DBF2002. DBF2002 was then used to size the configuration selected in conceptual design. The program analyzed approximately 1.12 billion different airplane combinations to determine the optimal configuration. Next, the detail design process created an airplane that matched the specifications of the optimal configuration. Finally, a manufacturing plan was created to build the airplane.

1.1 Conceptual Design

The first step in conceptual design was to select figures of merit that would be used to judge the airplane configurations. The figures of merit selected were required skill, required time, reliability, stability and control, rated aircraft cost effects, and speed. Next, design concepts such as, monoplane, biplane, tandem wing, and flying wing were scored based on the figures of merit. The design parameters investigated for the design concepts were the wing planform, landing gear configuration, and empennage configuration. The conceptual design selected was a monoplane with rectangular wing planform, a tricycle landing gear, and a conventional empennage.

1.2 Preliminary Design

The preliminary design started with creating figures of merit in order to judge the airplanes. The figures of merit selected were component weight, required time, required skill, and speed. The design parameters investigated were spar cross-section and material, wing/empennage structure, fuselage shape, fuselage shear web materials, landing gear cross-sections and fuselage side plate materials. The figures of merit were used to simplify these parameters.

Approximations were made on thrust estimation, battery selection, flight plan, drag, wind, servo performance, and commercial landing gear wheels, all of which were fed into DBF2002, a computer program written in MATLAB. The program iterated airplane characteristics such as airfoil C_l, number of balls carried, inner chord, outer chord, wingspan, number of motors, number of cells and amps. The

program then simulated the mission and determined the airplanes rated aircraft cost and flight score. The program investigated 1.12 billion different possible combinations of each airplane characteristics to find the airplane with the highest final score.

1.3 Detail Design

The detail design starts with the output from DBF2002. Using the program's output, a propulsion system was chosen to match the thrust and pitch speed requirements. Two off the shelf computer programs named Motocalc and Electricalc were used to select the motor and propeller configuration. Structural components were built and tested to ensure sufficient strength, the landing gear was modeled in ABAQUS to determine the structural integrity of the landing gear, the Flaperons and elevators were sized and the airfoil for the wing was selected. Finally, aircraft performance was predicted and the final configuration was drawn in Autocad.

2.0 MANAGEMENT SUMMARY

In the summer of 2001, the 2002 UT DBF team began to make preparations for this year's competition. By the beginning of the 2001 Fall Semester, the team was formed and new freshmen joined the team to aid in building an RC plane to compete in the 2002 Cessna/ONR Student Design Build Fly Competition. The team consisted of 6 seniors, 6 underclassman, all of whom are aerospace engineering majors, and one faculty advisor.

To keep the team motivated a Project Manager was chosen to oversee the project, and the team divided into three autonomous groups. The groups were Aerodynamics, Propulsion, and Structures. Table 2.1 shows the organizational structure of the team.

Table 2.1: Organizational Structure of the Team

Project Manager: Miranda Murdock		
Aerodynamics ***Leader*:** Paul Bauman	**Propulsion** ***Leader:*** Apoorva Bhopale	**Structures** ***Leader:*** Chris Moore
Kevin Mackenzie Dave Hughling Chris Moore	Clint Kam Miranda Murdock	Brandon Rogers Prasanna Weerakoon Bill Tandy Khoi Duong Sebastian Munoz

The team leaders and project manager developed a production schedule with major and minor tasks. Table 2.2 is the original timeline with both the planned start and finish dates and the actual finish dates for the tasks. Whenever a deadline could not be met, the timeline was slightly altered and the team attempted to complete other tasks ahead of schedule.

Under the direction of the team leaders, each group was given assigned tasks to complete over a certain amount of time. The workload was split among the group members according to each member's abilities as evenly as possible. Each building task was assigned to at least two people so that safety requirements were met and the task could quickly be completed. An optimization code was written by Chris Moore and Apoorva Bhopale in order to design the optimum aircraft given certain design parameters and assumptions and a stability code was written by Paul Bauman in order to verify the static and dynamic stability of the final aircraft design.

To verify the program's accuracy, a test aircraft was built using the design parameters selected by the program. This plane was also to be used as a test aircraft to allow the pilot to familiarize himself with the handling characteristics of the aircraft, while allowing other team members to focus on building the

competition plane, which would incorporate improvements and insights learned during the construction and design of the test aircraft.

Table 2.2: Project Timeline.

TASK	START DATE	FINISH DATE	ACTUAL DATE COMPLETE
Start-Up Tasks	**7/1/01**	**7/7/01**	7/9/01
Assemble Design Team	7/1/01	7/7/01	7/5/01
Get Budget Approval	7/1/01	7/7/01	7/9/01
Letter of Intent	7/7/01	7/7/01	7/9/01
Conceptual Design	**7/1/01**	**8/21/01**	8/21/01
Read and Discuss Contest	7/1/01	7/7/01	7/7/01
Configuration Analysis	7/8/01	7/21/01	7/20/01
Design Parameter Analysis	7/21/01	8/9/01	8/3/01
Conceptual Design Written	8/9/01	8/21/01	8/21/01
Preliminary Design	**8/21/01**	**12/31/01**	1/14/01
Design Parameter Analysis	8/21/01	9/14/01	9/16/01
Program Assumptions	9/14/01	10/1/01	10/2/01
Optimization Code Development	10/1/01	12/31/01	1/14/02
Preliminary Design Written	12/20/01	12/31/01	1/10/02
Detail Design	**12/31/01**	**2/1/02**	2/4/02
Final Program Run	12/31/01	1/2/02	1/16/02
Propulsion Systems Selected	1/3/02	1/10/02	1/22/02
Structural Testing / Modelling	1/3/02	1/8/02	1/8/02
Aerodynamic Analysis	1/3/02	1/18/02	1/21/02
Performance Calculations	1/18/02	1/20/02	1/22/02
Detail Design Written	1/20/02	2/1/02	2/4/02
Manufacturing Plan	**2/1/02**	**2/18/02**	2/18/02
Final Airplane Construction	2/1/02	2/16/02	2/17/02
Manufacturing Plan Written	2/14/02	2/18/02	2/20/02
Initial Test Flight	**2/18/02**	**2/18/02**	2/18/02
Compile Paper	**3/1/02**	**3/8/02**	3/10/02
Flight Tests	**2/21/02**	**4/19/02**	-
Contest	**4/26/02**	**4/28/02**	4/28/02

3.0 CONCEPTUAL DESIGN

The conceptual design process involved determining the general aircraft configuration and several design parameters. Then several aircraft parameters were analyzed to determine the overall shape of the aircraft for preliminary design. Each general design was then scored by figures of merit and the winning design proceeded to preliminary design.

3.1 Figures of Merit

The figures of merit were selected this year to minimize the design risks and maximize the overall score of the aircraft. In order to maximize the score while accounting for potential design setbacks or shortcomings, the figures of merit used were skill and time required to build, the design reliability, the inherent stability of the design, the Rated Aircraft Cost effect of the parameter, and the overall effect the parameter/design has on the speed of the aircraft. The complete conceptual design figure of merit table is shown in Table 3.2.

3.1.1 Required Skill

An important factor in the feasibility of each component's configuration was the ability of the team to build the configuration. Complicated configuration designs would increase the skill required, and material cost (due to mistakes) to build the component because many of the members were new and inexperienced. This figure of merit was given a weight of 2.

3.1.2 Required Time

As college students, time for aircraft construction and schoolwork must be balanced; therefore, the time a given configuration would take to implement was important. It would be unwise to choose a configuration that could potentially require more time to test and develop than available before the contest. Complicated configuration designs would increase the time required to build the component because many of the members were new and inexperienced. This figure of merit was given a weight of 1.

3.1.3 Reliability

Reliability of the configuration was a concern because in previous years, the DBF planes have crashed the week before the competition. Also a concern was the failure of the plane at the competition itself. The reliability of a configuration or a parameter is a measure of its complexity and difficulty/strain on the pilot. This figure of merit was given a weight of 2.

3.1.4 Stability and Control

The inherent stability of a design concept/parameter was important because certain designs are able to be more stable than others, lending to safer flight in Kansas' wind gusts. Due to the severe wind

conditions in Kansas, a high degree of control was desirable for the aircraft. This figure of merit was given a weight of 3.

3.1.5 Effect on Rated Aircraft Cost

Each design parameter was evaluated by the team to have a certain effect on the overall Rated Aircraft Cost of the aircraft for a given payload. The lower the figure of merit score for a given parameter, the more that parameter would raise the Rated Aircraft Cost of a given design; that parameter being the only difference between aircraft concepts. This figure of merit was given a weight of 4.

3.1.6 Speed

Examination of the mission scoring revealed that the flight score any given design could achieve was dependent on the design's total flight time. For a given thrust, one way to reduce the total flight time of the mission would be to reduce the aircraft's drag. This figure of merit was given a weight of 4.

3.2 Assumptions Made

In preparing a design to fly in the DBF competition, the team first had to discern the environment in which the plane would fly. Immediately, the team knew, from previous experience, that Kansas is very windy. Reference [1] contains weather data from the past 53 years and shows winds averaging 14mph in the month of April. Therefore, the team had to assume that the plane would be flying in windy conditions and must be considered in the design process.

The team has also learned from experience that there is no design that does not crash at least once. In fact, with the likelihood of a strong wind during the competition, a crash seems very probable. The team, therefore, assumes that the plane will crash at some point in testing and/or the competition and must, therefore, make a design that will be both modular and repairable.

Several assumptions were also made in the calculation of the Rated Aircraft Cost for each configuration. Ideally, a given aircraft design would maximize the amount of payload relative to the total aircraft weight. Thus it was assumed that the payload weight fractions from previous competitions would be similar to this year's aircraft. It was also assumed that each configuration would carry the maximum payload of 24 softballs. Also, based on an RC rule of thumb, it was further assumed that 50 watts of battery power would be required per pound of total aircraft weight. It was also assumed that last years batteries, which performed very well, would be used and that the motor configuration would run at 35 amps so that the amp load would not be too close to 40 amps, and yet still be large enough to allow for a low total battery weight. This allowed for the team to calculate the needed battery weight for the aircraft based on the loaded weight. The maximum wing loading was estimated using steady flight assumptions ($L = W$, $T = D$) for the landing approach phase, resulting in the equation:

$$\frac{W}{S} = \frac{1}{2}\rho(1.2V_S)^2 C_{L,max} \qquad (3.1)$$

where W/S is the wing loading, rho is the density at sea level, V_s is the stall velocity of the aircraft, and $C_{L,max}$ is the maximum lift coefficient. The factor of 1.2 is a safety factor included to prevent stall on landing. The pilot requested that the stall speed be no greater than 25 mph so that his approach speed would not be too great. $C_{L,max}$ was assumed from previous team's efforts to be 1.3. From (3.1) a wing loading of 3 lbf/ft^2 was then used for each configuration. Using the wing loading found from (3.1), it was then possible to find the necessary wing area for a given configuration. It was further assumed, based on previous years, that an Aspect Ratio of 7 would be an ideal balance between increasing structural weight and aerodynamic efficiency. It was also necessary, in order to calculate a Rated Aircraft Cost for each configuration, to assume that each configuration would have a fuselage with a fineness ratio of 8, in order to keep down the parasite drag on the fuselage. For 24 softballs, each with a diameter of 4 inches and stacked two abreast leaving 1 inch for fuselage skin, a fineness ratio of 8 means a fuselage length of six feet would be needed. Finally, it was assumed that each configuration would have one propeller and one motor

3.2 General Configurations

The first step in designing an airplane is determining the general shape of the airplane. Four different airplanes configurations were chosen for analysis. These configurations are described below.

3.2.1 Monoplane

The monoplane configuration has one wing and a horizontal stabilizer. Its versatility in design as well as relative ease in manufacturability makes it very attractive. The reliability of this configuration was decided to be slightly less than a biplane because of the monoplane's generally higher wing loading, which requires faster landing velocities. The monoplane was considered by the team to have the most inherent stability and control, this style of aircraft being well documented in both design and performance giving the DBF team a large database from which to draw. The monoplane, was considered a relatively fast configuration owing to its relatively lower parasite drag than the biplane or the tandem.

Based on previous year's aircraft data, the historical payload weight fraction for monoplanes was estimated to be 0.37. This led to an empty weight of 16 pounds and thus a wing area of 8.5 ft^2. For a wing with an aspect ratio of 7, the wingspan would be 7.7 ft and the chord would be 1.1 ft. The battery weight for a 25.4 pound loaded monoplane would be 4 pounds. For a monoplane, it was assumed that there would be 1 rudder, 2 elevators, 2 flaperons, 6 servos, and 1 motor controller. The monoplane's estimated RAC was 11.828 and the breakdown can be seen in Table 3.1.

3.2.2 Biplane

The biplane configuration is very similar to that of the monoplane, with the exception of a second wing above the main wing. In previous Design, Build, Fly competitions this design has done very well owing in part to both its high reliability and its good controllability. The skill and time required to build the aircraft would be the longest of the four configurations because of the need to build two wings and two pylons on the wings. Because of the large amount of surface area, the parasite drag of the biplane was considered large and therefore the configuration was considered slow.

Based on previous year's aircraft data, the historical payload weight fraction for biplanes was estimated to be 0.345. This led to an empty weight of 18 pounds. Since the biplane has two wings, the wing loading was 6 and thus each wing for the biplane had a wing area of 4.6 ft^2. For a wing with an aspect ratio of 7, the wingspan would be 5.6 ft and the chord would be 0.8 ft. The battery weight for a 27.4 pound fully loaded biplane would be 4.3 pounds. For a biplane, it was assumed that there would be 1 rudder, 1 elevator, 2 flaperons, 5 servos, 1 motor controller, and two wing pylons (vertical surfaces) without control surfaces on them. The biplane's estimated RAC was 13.138.

3.2.3 Tandem

The tandem wing airplane consists of two similar sized wings, one in front and one in the rear of the airplane. The tandem wing's horizontal stabilizer creates lift and controls the aircraft's pitch. The forward wing's downwash can make control difficult for the pilot and reduce the effective lift coefficient of the rear wing if not properly designed. Previous DBF teams have had direct experience with the design and implementation of the tandem wing configuration without much success leading the team to give the tandem design a low reliability score. This configuration also required the construction of two wings and therefore needed more time and skill in building it. Similar to the biplane, the tandem configuration has a large surface area, and therefore a large parasite drag resulting in slower flight speeds for a set amount of thrust.

Based on last year's design, the payload weight fraction for our tandem was 0.35. This leads to an empty weight of 18.25 pounds. Since the tandem has two wings, the wing loading was approximately 6. Each wing on the tandem therefore had a wing area of 4.7 ft^2. For a wing with an aspect ratio of 7, the wingspan would be 5.7 ft and the chord approximately 0.82 ft. The battery weight for the tandem would need to be 4.3 pounds. For the tandem, it was assumed that there would be 1 rudder, 1 elevator, 2 flaperons, 5 servos, and 1 motor controller. The tandem's estimated RAC was 13.00.

3.2.4 Flying Wing

The flying wing is one of the most efficient designs for an airplane. Having fewer control surfaces makes its Rated Aircraft Cost lower. By having less surface area for a given amount of lift, the flying wing has substantially less skin friction drag and therefore the plane is able to fly at greater speeds given the same

amount of thrust as other configurations. Difficulties arise, however, in the manufacturing of the plane. The airfoil requires special attention since the stability of the aircraft depends a great deal on the airfoil shape. Also, the flying wing is less suited for payloads that have large volumes compared to the other types of aircraft. Previous UT DBF teams have, however, had some success with the flying wing configuration, so this plane was considered.

Though there were not many flying wing aircraft on which to draw the payload weight fraction, the average found was 0.39. This leads to an empty weight of 14.7 pounds. The flying wing, therefore, needed a wing area of 8 ft^2, a wingspan of 7.5 ft and a chord of 1.1 ft. The battery weight for a 24.1 pound aircraft would be 3.75 pounds. For the flying wing, it was assumed that there would be 2 rudders, 0 elevators, 2 flaperons, 5 servos, and 1 motor controller. The flying wing's estimated RAC was 10.991. Table 3.1 shows the Rated Aircraft Cost breakdown for each general configuration.

Table 3.1: Configuration Rated Aircraft Cost Comparision.

Description	Monoplane	Biplane	Tandem	Flying Wing
MEW	**16**	**18**	**18.25**	**14.7**
REP	**4.000**	**4.300**	**4.300**	**3.750**
# of Engines	1	1	1	1
Total Battery Weight (lb)	4	4.3	4.3	3.75
MFHR	**211.4**	**244.4**	**236.32**	**194.8**
Wings	**76.4**	**114.4**	**116.32**	**74.8**
Wing Span (ft)	7.7	5.6	5.7	7.5
Max Chord (ft)	1.1	0.8	0.82	1.1
# of Control Surfaces	2	2	2	2
# of Wings	1	2	2	1
Fuselage	**60**	**60**	**60**	**60**
Fuselage Length (ft)	6	6	6	6
Empenage	**30**	**30**	**20**	**20**
Vertical Surfaces w/no control	0	2	0	0
Vertical Surfaces w/ control	1	1	1	2
Horizontal Surfaces	2	1	1	0
Flight Systems	**35**	**30**	**30**	**30**
# of Servos	6	5	5	5
# of Motor Controllers	1	1	1	1
Propulsion Systems	**10**	**10**	**10**	**10**
# Engines	1	1	1	1
# Propellers	1	1	1	1
TOTAL RATED AIRCRAFT COST	**11.828**	**13.138**	**13.0014**	**10.991**

3.3 Design Parameters

For the initial stage of the conceptual design, several design parameters were investigated for their performance advantages and disadvantages. The design parameters that were chosen for further analysis were then considered in the numerical program.

3.3.2 Wing Planform

The team considered three distinct planform shapes for the wing(s). The wing planforms considered were rectangular, tapered, and elliptical. The rectangular platform offers easy construction that the team was comfortable with from previous year's experience. However, the rectangular planform is the least efficient of the three planforms due to large tip-losses. The tapered wing planform has the advantage of more efficient production of lift while remaining relatively easy to construct. Tapered wings can suffer difficulties at the wing tips due to low Reynolds numbers; loss of control can occur if the wing tips stall. Tapered wings, while producing more lift per unit area than the rectangular wing, do not have a significant Rated Aircraft Cost advantage over the rectangular wings because the manufacturing man-hours cost is calculated by adding eight dollars per foot of wing span and eight dollars per foot of maximum exposed chord, instead of charging per square-foot of wing area as in past years. The elliptical planform is the most efficient wing shape in that it minimizes the induced drag of the wing. Unfortunately the elliptical wing is also extremely difficult to construct.

3.3.4 Landing Gear Configuration

The team examined three landing gear configurations: tail dragger, tricycle, and retractable tricycle. The tail dragger configuration is inherently unstable, having the tendency to want to flip around such that the plane faces backwards. Neither tricycle configuration contains this instability; they were therefore both given scores of 1. A tail dragger would also be a short, lighter option (meaning less RAC) with less drag than a tricycle; however, the pilot felt uncomfortable with the tail dragger configuration. The tricycle configuration was simple and had been used by the design team in previous years, but would have relatively large amounts of drag even with fairings and would weigh more than the tail dragger. The retractable landing gear would have little to no drag; however, it would be the heaviest of the options and the most complicated and difficult to construct.

3.3.5 Empennage Configuration

The empennage configurations considered consisted of conventional, T-tail, canard, and V-tail. The conventional and T-tail are somewhat versatile, easily constructed, and well documented designs that have good stability and control characteristics. The T-Tail, with the extra structure required to place the horizontal tail on top of the vertical tail was more difficult and required more time to build. The V-tail, however, is much more difficult to build and not quite as controllable, but a little less expensive in Rated Aircraft Cost as the conventional configuration. The canard, as discussed earlier, presents aerodynamic

and control problems and has met with little success from previous UT DBF teams. It requires little more skill than the conventional tail to build, but still has the RAC disadvantage over the V-Tail. The canard design was also considered to be the slowest design because of the extra induced drag caused by the lift the canard produces. The conventional tail configuration was chosen based on its figure of merit score.

3.3.3 Engine Configuration

The engine configuration is a decision between a pusher, puller or multiengine pusher/puller. The pusher configuration is better to resist aircraft crashes. The puller is more conventional and is sometimes required by the geometry of the aircraft. The multiengine configuration has the advantage that if one engine fails, the plane could still possibly land safely. After much debate, it was decided to let the optimization code iterate between the possibilities.

Table 3.2: Conceptual Design Figures of Merit.

	Component	**Skill Required (x2)**	**Time Required (x1)**	**Reliability (x2)**	**Stability & Control (x3)**	**RAC effect (x4)**	**Speed (x4)**	**Final Score**
General Configuration	**Monoplane**	1.00	1.00	0.95	1.00	0.93	0.90	**14.27**
	Biplane	0.85	0.80	1.00	0.95	0.84	0.80	12.91
	Tandem	0.85	0.85	0.85	0.80	0.85	0.85	12.60
	Flying Wing	0.60	0.70	0.80	0.75	1.00	1.00	12.95
Wing Planform	**Rectangular**	1.00	1.00	1.00	1.00	1.00	0.85	**14.40**
	Tapered	0.90	0.80	1.00	0.95	0.95	0.95	14.05
	Elliptical	0.50	0.50	1.00	0.90	0.90	1.00	12.80
Landing Gear Configuration	Tail Dragger	0.90	1.00	1.00	0.90	1.00	0.80	13.70
	Tricycle	1.00	0.95	1.00	1.00	0.95	0.75	**13.75**
	Retractable	0.60	0.50	0.80	1.00	0.90	1.00	13.10
Empennage Configuration	**Conventional**	1.00	1.00	1.00	1.00	0.90	0.90	**14.20**
	T-Tail	0.90	0.95	0.90	0.90	0.90	0.95	13.75
	V-Tail	0.75	0.90	0.80	0.95	1.00	1.00	14.05
	Canard	0.95	1.00	0.80	0.80	0.90	0.85	13.10

3.5 Features that Produced Final Configuration

The winning general configuration was the monoplane. This was mainly due to the ease of construction and the stability and controllability of the design. While it did not have the best RAC or the best speed, it was competitive enough in those categories to warrant its selection based on its qualities in the others. The wing planform selected for the monoplane was the rectangular wing, again because it had better control characteristics and was simple and quick to build. The Tricycle gear was selected mainly due to the pilot's unease with the tail dragger and the unreliable nature of the retractable gear. Finally the conventional tail was chosen because, while it wasn't the fastest or the cheapest (RAC) solution, it did have the overall best features.

4.0 PRELIMINARY DESIGN

Once the general configuration was decided upon in the conceptual design phase, a numerical program, DBF2002, was written using MATLAB in order to model the configuration's flight score given various aircraft dimensions. The program was written after the preliminary design phase was complete so that all of the decisions and assumptions made in the preliminary design could be written into the program. The end goal of the preliminary design was therefore to provide analysis on several design parameters so that the program could optimize the aircraft's general dimensions efficiently.

The general program flow chart is shown in Figure 4.1. The program was first given the contest parameters such as the takeoff field length and maximum battery weight. It was then given plane parameters such as the properties of the materials used and of the commercially bought parts. The program worked by iterating through various potential values for the aircraft's dimensions important to the performance of the aircraft such as, but not limited to, wingspan, chord, C_L, the number of softballs carried, the number of battery cells used, and the amps delivered to the motor(s).

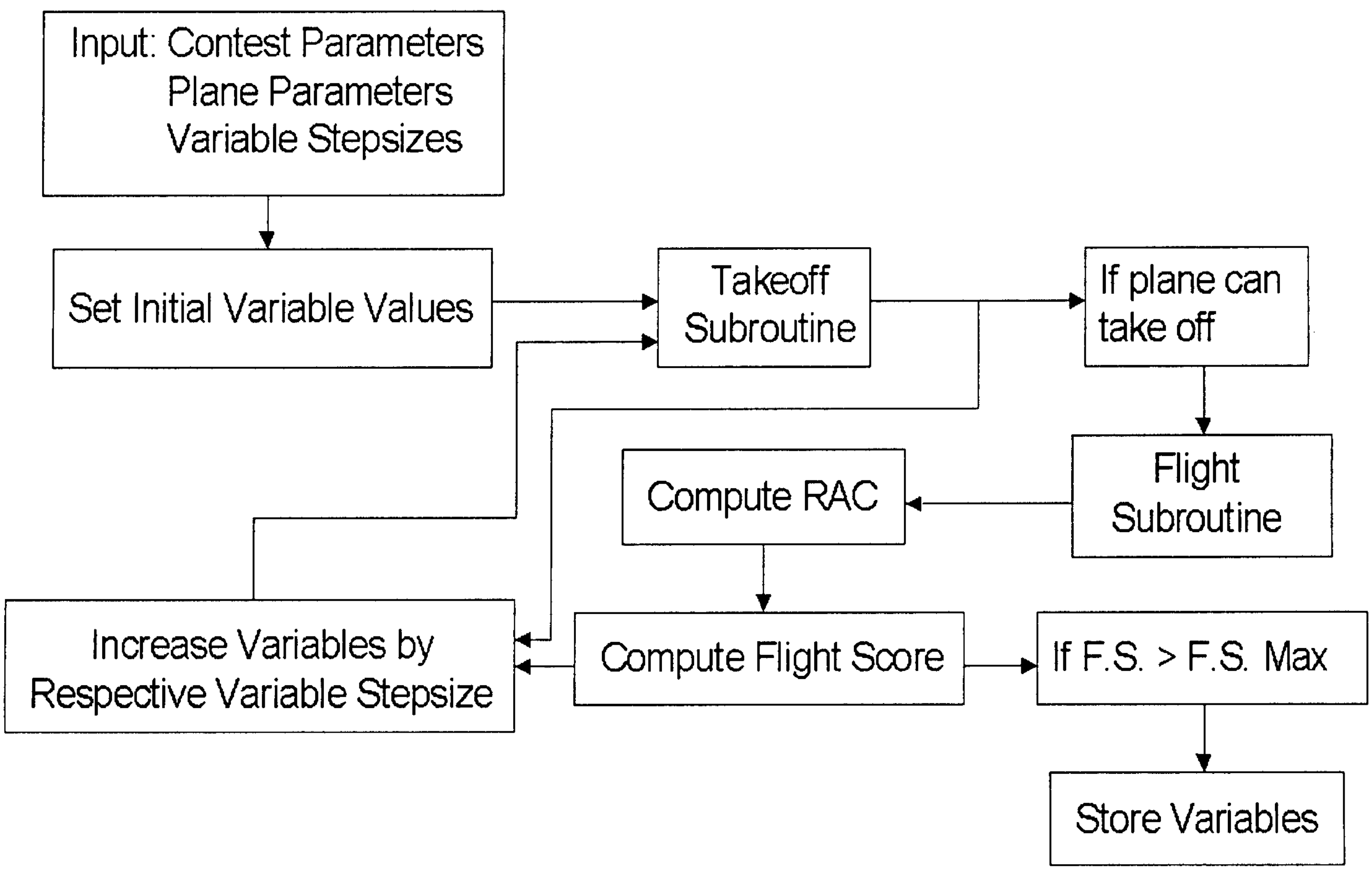


Figure 4.1: Optimization Program Flow Chart

With the values for these dimensions set, other aircraft performance variables were then calculated by the program, such as the thrust produced by the motor/propeller configuration, the total drag coefficient of the aircraft, the mass of the aircraft, and the Rated Aircraft Cost of the given aircraft. In order for the mass to be accurately calculated, the materials used in the aircraft construction needed to be pre-determined. To calculate the RAC for a given plane, various assumptions were made during the preliminary phase such as the number of servos.

The program then determined whether the plane would successfully takeoff in the minimum required distance, which we assumed to be 150 feet to provide a safety buffer. If the configuration took off, then the program calculated the time it would take the aircraft, with the assumed dimensions, to fly the course by use of incremental time-steps and $a = \mathcal{F}/m$. The flight score of the aircraft was then calculated by using the total score formula for the contest assuming a unit paper score.

4.1 Figures of Merit

Figures of merit were selected for the preliminary design phase in order to eliminate certain design parameters and allow for more efficient selection of the best scoring airplane by the program. The complete preliminary design figure of merit table is shown in Table 4.3.

4.1.1 Component Weight

In order to compare the weight of each component's various possible configurations, the worst-case loading for each component was considered. The minimum weight for a given configuration that could withstand the worst-case load was then compared against other configurations of the same component. This figure of merit was given a weight of 3.

4.1.2 Required Time

Since the contest dictates a tight schedule, an important factor in the feasibility of each component's configuration was the time required to build the configuration. Complicated configuration designs would increase the time required to build the component because many of the members were new and inexperienced. This figure of merit was given a weight of 1.

4.1.3 Required Skill

Another important factor in the feasibility of each component's configuration was the ability of the team to build the configuration. Complicated configuration designs would increase the skill, and material cost (due to mistakes) required to build the component because many of the members were new and inexperienced. This figure of merit was given a weight of 2.

4.1.4 Speed

As stated, the flight score for any given is dependent on the design's total flight time. For a given amount of thrust, one way to reduce the total flight time of the mission would be to reduce the aircraft's drag. This figure of merit was given a weight of 2.

4.2 Design Parameters Investigated

4.2.1 Spar Cross Section and Material

In order for the program to predict the weight of the spar, both the spar material and the spar geometry had to be chosen. The worst-case load on the spar's structural design was determined to be the "Wingtip test" stated in the competition rules. In this test point loads at each wingtip support the fully loaded plane, which was assumed to weigh 24 pounds (9.4 lbs of payload divided by a payload fraction of 0.4). All spar designs considered must be at least strong enough to withstand this test. Also, the estimated maximum torque with full flaps down was considered as a constraint when necessary; however it was in general less limiting than the bending case. The torque on the spar was estimated to be 20 ft-lbs with full flaps down.

The spar materials investigated included 2014-T6 aluminum, balsa wood, fiberglass/balsa composite, carbon-fiber, and a carbon-fiber/aluminum composite used the previous year. The aluminum spar cross-sections considered were a hollow circular geometry and box beam geometry. The balsa wood and fiberglass-balsa composite spar cross-sections considered were a rectangular box beam and a solid rectangular beam. The only carbon-fiber cross-section considered was a hollow circular cross-section. Tapered cross-sections were examined and found to present only 3% to 5% reductions in weight (~0.1 oz) at the expense of increasing the construction difficulty and time of both the spar and the ribbed wings, since different size ribs would have to be made at each location along the wingspan. The tapered cross-sections were therefore discarded.

In order to find the minimum weight for a given material cross-section combination, the following flexure formula was used

$$\sigma_u = \frac{M_{max} * y}{I} \tag{4.1}$$

where σ_u is the ultimate stress, I is the moment of inertia and is determined by the spar geometry, y is the distance from the neutral axis, and M_{max} is the maximum moment the spar experiences from the load. Since the limiting load was a point load at the wing tip (equal to one-half of the plane's mass) and the maximum stress will occur at a height, y, equal to one-half of the spar's maximum height, (4.1) can be simplified into three different cases based on the spar's moment of inertia.

Also used to find the minimum weight were the following torsion formulas for (a) rectangular and (b) thin walled members:

$$\tau_{max} = \frac{T}{\alpha * h * w^2} \tag{4.2a}$$

$$\tau_{max} = \frac{T}{2 * t * A_m} \tag{4.2b}$$

where, in (4.2a), τ_{max} is the maximum shear strength of the material, T is the applied torque from the deflection of the flaperons, h is the height of the spar, w is the width of the spar, and α is a dimensionless constant that depends on $^h/_w$ obtained from Table 4.3 found in [2]. In (4.2b), A_m is the area enclosed by the median line, and t is the thickness of the spar.

For each of the materials, σ_u was found by three-point load tests until failure and, in aluminum's case, looking up the published ultimate stress value for 2014-T6 aluminum. Table 4.1 shows the various materials densities and ultimate stresses.

Table 4.1: Experimental Values for Material Properties.

	Al – 2014-T6	**Balsa Wood**	**Fiberglass/Balsa**	**Carbon-Fiber**	**Carbon/Aluminum**
σ_u (ksi)	60	1.7	3.1	7.5	6
τ_{max} (ksi)	4	0.3	~ 0.6	~ 2.5	4.5
ρ (lb/ft^3)	175	7.2	10.6	75	115

To determine the spar dimensions for each configuration that gave the minimum weight and supported the wingtip load, two approaches were used, one for rectangular cross sections and the other for the circular spar geometry. Both geometries were constrained by a maximum height/diameter allowed that was based on the quarter chord thickness of the wing. For preliminary design purposes, this thickness was assumed to be 1.5 inches. The hollow spars were also constrained by an estimated minimum thickness to avoid thin wall buckling. For the metal spars this was assumed to be 1/32th of an inch, for the carbon-fiber and the balsa wood it was assumed to be 1/16th of an inch. The last constraint on the spars was material availability. For the metal spars, another constraint is that the aluminum is only readily available in discrete diameters leading to certain spar geometries being unattainable.

For the rectangular geometries, the torsion formula (4.2a) was solved for the minimum width assuming the height of the spar was equal to the maximum allowed height. The maximum stress the spar would experience was then calculated from the flexure formula for the wingtip load. If this exceeded the ultimate stress value shown in Table 4.1, the width was increased so that the maximum stress in the spar was less

than the ultimate stress. The weight of the 4-foot long spar was then calculated from the dimensions found and is shown in Table 4.2.

For the hollow circular cross-section, the flexure formula was solved first. Assuming the wall thickness to be equal to the minimum thickness to avoid buckling, the minimum diameter required to satisfy the wingtip load was calculated. If this diameter exceeded the maximum allowed, the thickness was increased such that the diameter was equal to the maximum allowed. The maximum shear experienced in the spar was then found from formula (4.2b) based on the estimated pitching moments produced with full flaps down. If this shear exceeded known, representative values, then the wall thickness was further increased until the spar could withstand the required loading. The weight of a 4 ft long spar was then calculated.

Table 4.2: Spar Material/Cross-Section Minimum Weights.

Material	Cross-Section	Weight (oz)
Al	Hollow-Circular	3.82
Al	Hollow-Rectangular	4.5
Al/Carbon	Hollow-Circular	7.23
Balsa	Hollow-Rectangular	3.89
Balsa	Solid	2.95
Carbon	Hollow-Circular	4.22
Fiberglass/Balsa	Solid	3.3
Fiberglass/Balsa	Hollow-Rectangular	3.2

The estimated time and skill required for each spar design was discussed in a group meeting. The aluminum spars would require machine shop access and skills that the team did not have, so they were scored correspondingly low. The composite laminate spars, such as the aluminum/carbon and the fiberglass/balsa designs, required skills with composites that several, but not all, of the team had from previous years. The hollow rectangular balsa wood spar designs required more time and slightly more skill than the solid balsa spar configurations, and were therefore scored slightly lower in both categories. The spar materials and cross-sections were assumed to not effect the speed of the flight. As shown in Table 4.3, a solid balsa spar was selected based on its figure of merit score; however, after testing it was found that moderate torques caused unacceptable twisting of the spar and so the next best spar was chosen, the solid balsa spar with a fiberglass layer. The fiberglass was found to add sufficient resistance to twisting and prevented the balsa wood from warping.

4.2.2 Wing/Empennage Structure

Four wing/empennage structures were considered; a solid foam core with a monokote surface, foam ribs covered by monokote, balsa ribs with monokote, and foam ribs with birch sheeting. The solid foam core could be easily and quickly produced, but would weigh the most. The foam ribs would weigh the least,

but take the longest to make and have higher drag than the solid foam wing because the monokote tends to bow in-between the ribs. The balsa ribs, since they are denser than foam, would need to be thinner in order to make the wing lighter; however, the monokote was found to bow too much in-between the ribs as the thickness of the ribs decreased, and so the balsa rib wing was heavier and had more drag than the foam rib wing. The foam ribs with a thin birch sheeting instead of a monokote surface eliminated the bowed effect of the monokote, but added weight and significant time to the wing construction. The foam rib wing with monokote was chosen.

4.2.3 Fuselage Shape

The fuselage design was determined to be vital to success in this year's competition because the flight score for a given run is calculated by the number of laps flown plus the number of softballs carried, divided by the total time of the run. The fuselage design affects both the number of balls that a given aircraft configuration can carry and the total mission time for that aircraft due to potential loading advantages or disadvantages of a given fuselage. The fuselage shapes considered were an extended symmetric airfoil, a cylindrical tube, wing-mounted pods, and a cambered airfoil. The symmetric airfoil was not nearly as long as the cylindrical fuselage and, being an airfoil shape, it was extremely quick and easy for the team to manufacture. It was potentially heavier than the cylindrical fuselage, but previous years found that is was difficult to manufacture light cylindrical shapes. This difficulty is reflected in the low construction time and skill required figure of merit scores received by both the wing-mounted pods and the cylindrical fuselage. The wing-mounted pods and the cylindrical fuselage were considered to be slow because the rules state that the balls cannot be stacked, therefore the they would need to be either prohibitively long or very wide in order to carry 24 softballs. The drag, therefore, on the cylindrical shapes would be larger than the drag of an airfoil shaped fuselage. The cambered airfoil was considered slower than the symmetric airfoil because the nature of the airfoil was such that it would necessarily have a low aspect ratio (it would be long and thin) and therefore have a large induced drag coefficient. The extended airfoil section was chosen and the optimization program therefore only considered the airfoil fuselage; however the geometry of the airfoil fuselage was such that a pusher engine configuration could not be considered.

4.2.4 Fuselage Shear Web Materials

Once the fuselage shape was determined to be the symmetric airfoil, it was necessary to select the material to be used internally for the wing-carry through structure and to support the payload. The shear web materials considered were balsa wood, a fiberglass/balsa composite, and poplar. Since the fiberglass/balsa composite allowed the team to add strength were it was needed such as to the wing carry-through or to the front shear web were the front gear attaches to, it was the lightest of the three options. However, the fiberglass/balsa option did take both more time and skill than the other options, but the team was relatively experienced with fiberglass so it was not a big concern. The poplar, because of

its large ultimate strength, was constrained by buckling because the shear webs made from it could be made extremely thin and still theoretically support the bending load. The poplar shear webs, therefore, had to be made thicker to prevent buckling and were thus the heaviest option. The team also investigated a fiberglass/poplar shear web design, but the balsa was already being used for other parts of the plane and was readily available, whereas pieces of poplar in the sizes we required were difficult to obtain. The shear web material chosen was the fiberglass/balsa laminate.

4.2.5 Landing Gear Cross-Sections

The team considered three different landing gear extension cross sections: a hollow circular beam, a solid circular beam, and a solid rectangular beam. The hollow circular beam would be the lightest gear extension, but it would be wider than the comparable solid circular beam and therefore, even after adding fairings, the hollow beam would have slightly more drag. Also, previous teams have already machined a connector between the landing gear strut and the extension for a solid cross section; however, this connector would not work for a hollow cross section and so new connectors would need to be machined, a skill the team did not have. In order for the rectangular cross section to withstand loads from both the side and the front, depending of the angle of impact with the ground, it had to be made excessively thick, making it have both more drag and weight than the other two cross sections. It would also require machining in order to get the precise dimensions needed so as to minimize its weight. The final landing gear's pieces are labeled in the detail design drawing package and show that the team chose the solid circular cross section.

4.2.6 Fuselage Side Plate Materials

The final design parameter the team considered before the optimization program was written was the material for the fuselage side plates. Since the fuselage shape was a hollowed out airfoil with shear webs running across the width of the fuselage, side plates were needed in order to support the longitudinal loads of the structure. The main landing gear was to be attached to the side plates and thus the side plates had to be able to withstand a hard landing load. They also had to be stiff enough such that the load induced by elevator deflection would not significantly deflect the plates. The team considered balsa wood side plates, balsa wood with birch plating, a fiberglass/balsa composite, a carbon-fiber/balsa composite, and a poplar fuselage. The pure balsa design had the advantage of being quick and easy to construct; however, it had to be heavy in order to be stiff enough so that no control loss occurred. The balsa wood with birch plating option was lighter than the pure balsa design because the birch added the needed stiffness with less weight. However, it would require a good amount of time to construct the plates this way since the balsa and the birch would need to be cut into an airfoil shape and then the birch plating would need to be laminated onto the balsa. The fiberglass/balsa composite, as well as the carbon-fiber/balsa composite allowed the team to lightly stiffen the side plates. The carbon-fiber design would require more time to construct due to the team's relative inexperience with carbon, but made up for

the extra time in weight savings. The poplar concept was heavy because poplar is very strong, but also very dense meaning that the poplar side plates were constrained by buckling and therefore had to be made thicker than was necessary for just a bending load. Poplar was also difficult to obtain in the dimensions that were needed for the side plates. The team decided to make the plates out of carbon-fiber/balsa composite as shown below.

Table 4.3: Preliminary Design Figures of Merit.

	Component	Weight (x3)	Time Required (x1)	Skill Required (x2)	Speed (x2)	Final Score
Spar Configuration	Al (H,C)	0.77	0.70	0.60	0	4.22
	Al (H,R)	0.66	0.50	0.40	0	3.27
	Al/Carbon (H,C)	0.41	0.75	0.70	0	3.37
	Balsa (H,R)	0.76	0.85	0.95	0	5.03
	Balsa (S,R)	1.00	1.00	1.00	0	6.00
	Carbon (H,C)	0.70	0.50	0.45	0	3.50
	Fiberglass/Balsa (H,R)	0.89	0.70	0.85	0	5.08
	Fiberglass/Balsa (S,R)	0.92	0.90	0.90	0	**5.47**
Wing / Empennage Structure	Foam core w/monokote	0.74	1.00	1.00	1.00	7.22
	Foam ribs w/monokote	1.00	0.80	0.95	0.90	**7.50**
	Balsa ribs w/monokote	0.89	0.90	0.90	0.85	7.07
	Foam ribs w/birch sheeting	0.80	0.70	0.80	1.00	6.70
Fuselage Shape	**Symmetric Airfoil**	0.9	1	1	1	**7.70**
	Cylindrical	1	0.60	0.70	0.8	6.60
	Wing mounted pods	0.95	0.40	0.50	0.85	5.95
	Cambered Airfoil	0.9	0.95	0.95	0.9	7.35
Fuselage Shear Web Materials	Balsa Wood	0.88	1	1	0	5.64
	Fiberglass/Balsa	1	0.9	0.9	0	**5.70**
	Poplar	0.85	1	1	0	5.55
Landing Gear Cross-Sections	Hollow Circular Beam	1	0.9	0.9	0.98	7.66
	Solid Circular Beam	0.95	1	1	1	**7.85**
	Solid Rectangular Beam	0.9	0.9	0.8	0.9	7.00
Fuselage Side Plate Materials	Balsa Wood	0.81	1	1	0	5.43
	Balsa w/Birch Plating	0.9	0.8	0.9	0	5.30
	Fiberglass/Balsa	0.97	0.85	0.85	0	5.46
	Carbon-Fiber/Balsa Wood	1	0.8	0.85	0	**5.50**
	Poplar	0.78	0.9	1	0	5.24

4.3 Program Assumptions Made

The assumptions made in writing the optimization code are more numerical/physical in nature than the assumptions used in the conceptual design that were based on previous knowledge, experience, and intuition.

4.3.1 Thrust Estimation

Previous teams used a commercial program, Electricalc, to predict the performance of a given motor/propeller combination for a given number of battery cells and amps. This year, Electricalc was again used and further verified with more extensive static-thrust tests, in which Electricalc generally predicted thrusts below the measured values by 5% to 10%. After extensive analysis of Electricalc's output, static thrust versus pitch velocity curves for various motor/propeller configurations with different amounts of power input were obtained as shown in Figure 4.2. It should be noted that the curve fit's were only accurate above 450 watts of input power; however, this was not considered to be a problem because most aircraft that were competitive had input power levels around 400 or more watts.

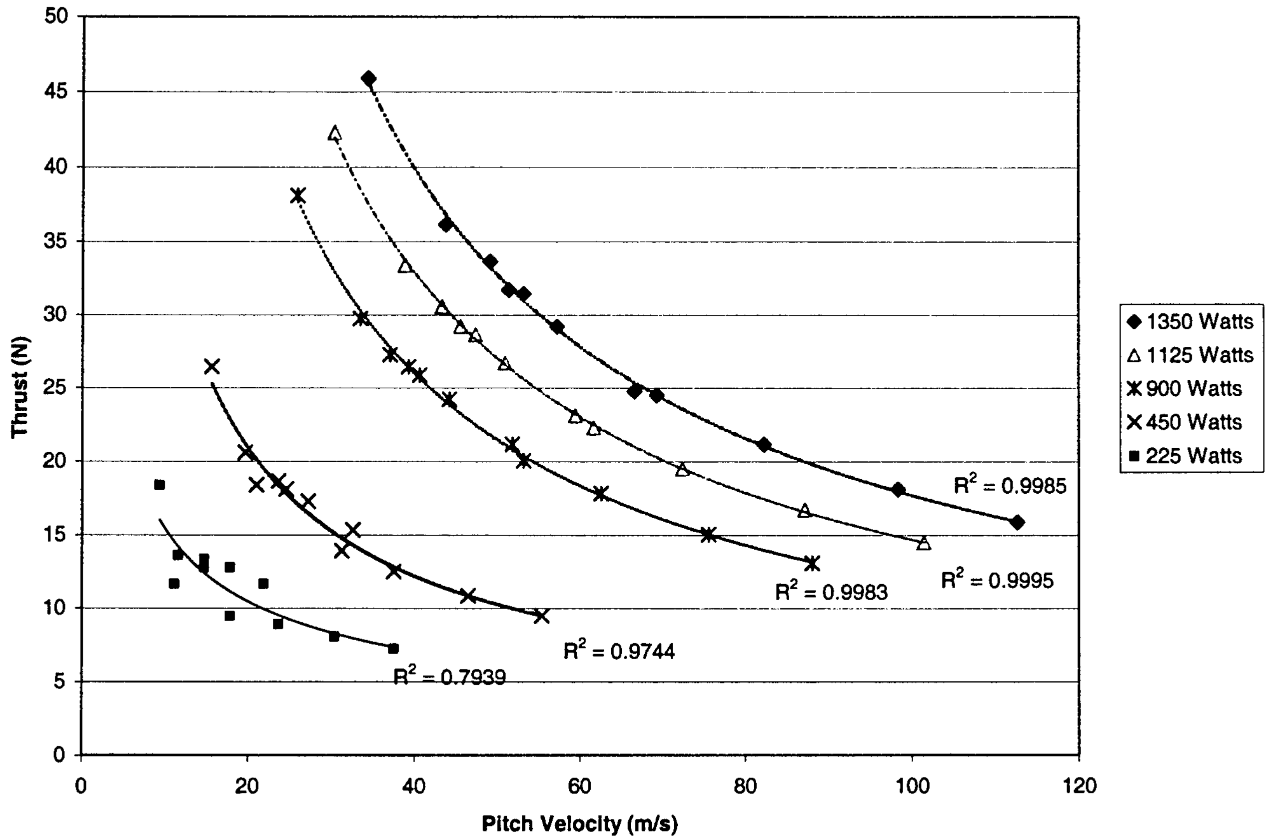


Figure 4.2: Static Thrust vs. Propeller Pitch Velocity Curves.

Further analysis of the static thrust as a function of both input power and pitch velocity yielded a relation for the thrust versus pitch velocity and watts of the form

$$T_0(W, V_p) = (C_1 + C_2 * W) * (V_p)^{(C_3 * W^2 * C_4 * W + C_5)} \tag{4.3}$$

where T_0 is the static thrust, C_i are the constants determined by the curve fit, W is the input power in watts, and V_p is the pitch velocity of the propeller. The pitch velocity corresponds to the velocity at which,

according to Electricalc, the thrust deviates from a linear relation with the aircraft velocity. Up to the pitch velocity the thrust as a function of velocity was assumed to be

$$T(V) = T_0 * \left(1 - \frac{V}{2 * V_p}\right) \tag{4.4}$$

where T is the thrust for a given velocity, and V is the velocity. To ensure that the plane did not approach the non-linear region of the thrust versus velocity curve the pitch velocity was assumed to equal 2.1 times the stall velocity of the aircraft. This assumption was found to keep the maximum velocity of the aircraft under the pitch velocity of the propeller by approximately 20 mph ensuring that, even with gusts, the thrust would not fall off rapidly.

4.3.2 Battery Selection

The team selected the batteries that the plane would use before writing the optimization program so that the weight per battery cell, internal resistance, and total milliamp-hours could be accounted for in the program. The batteries were constrained by the contest rules, which state that all propulsion of the airplane is to be powered by no more than 5 lbs of Nickel Cadmium (Ni-Cad) batteries. The second assumption made in selecting the batteries was that they must last for a minimum of 4 minutes of full throttle flight time. Another assumption was that the team would not manufacture its own custom gearing due to our lack of manufacturing capability, therefore the gearing will be limited to those that we could purchase.

The amp-hour rating of the battery determines the time duration of the battery's output at a certain load. For example, if a 5 amp-hour battery is being drained at 5 amps, the battery will last 1 hour. In order to maximize the energy density of the battery pack, the batteries have to last just long enough to complete the task. Batteries with lower capacities can be placed in parallel to get the right duration of flight. For example, if two 2.5 amp-hour batteries are placed in a parallel circuit, they will have the total net capacity of 5 amp-hours. The internal resistance of the battery determines the amount of watts the battery loses while outputting the power. To maximize the airplane's performance, the battery pack should contain the greatest amount of energy within the five-pound limit and have the lowest possible internal resistance.

In order to find the best Ni-Cad battery suited for the competition, the propulsion team investigated 177 different Ni-Cad batteries to find the most energy dense Ni-Cad batteries available on market. The weight was normalized for all the batteries, to ease comparisons. Figure 4.3 shows some of the best batteries and their energy output as a function of drain rate in amps.

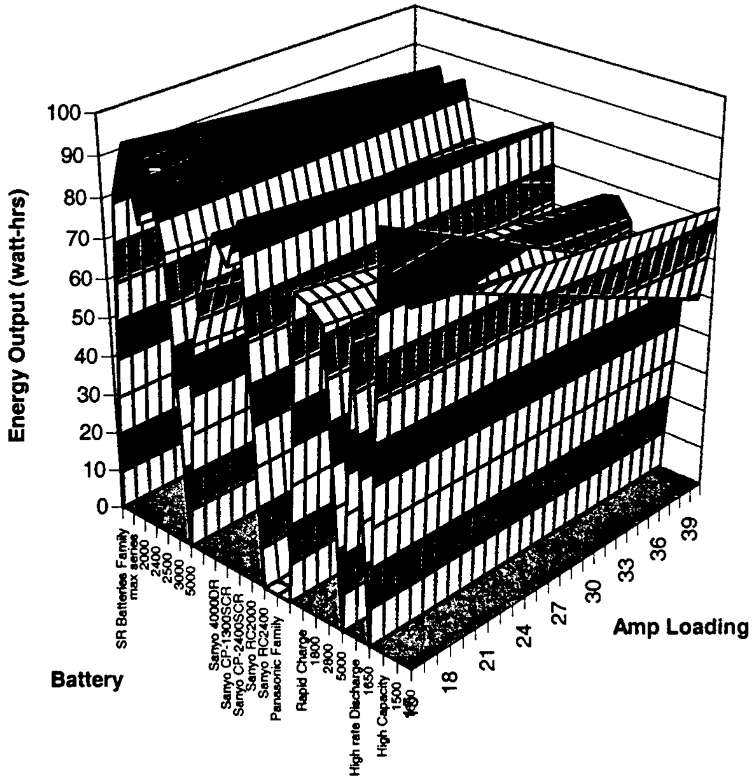


Figure 4.3: Battery Energy Output vs. Amp Load for Various Batteries.

According to the results, the Panasonic High Capacity 1500 mah battery had the highest energy output. However, as the amp loading increased, the energy output greatly decreased. The SR batteries 2400 mah battery had the second highest energy output initially, but as the amp loading increased, it maintained its high-energy output. Since the motor configuration would most likely need more than 20 amps, this battery was chosen. The information of the SR 2400 battery is listed in the table below.

Table 4.4: Battery Statistics Input into Program.

SR Batteries 2400 max	Statistics
mAh	2400
$^{Weight}/_{cell}$ after wrapping (oz)	2.1
Internal Resistance (mΩ)	3.8

4.3.3 Flight Plan

The first aerodynamic assumption made in the program was the flight plan. Since the flight score was dependent on the total flight time, the flight plan was designed to minimize the time while limiting the pilot difficulties. This was accomplished by limiting the flight plan such that the aircraft flew at full throttle for the entire course except for just before the landing approach turn. After the second 360-degree loop was completed, the throttle was set to zero, allowing the plane to glide until the velocity was 1.2 times the stall velocity of the aircraft. The plane was then flown at half throttle until landing.

For calculating turn performance of the airplane, a quasi-steady, level turn was assumed. That is, the change in velocity in the direction of the flight path is assumed negligible as well as the component of thrust normal to the flight path and the turn takes place at constant altitude. Also, a coordinated turn is assumed. That is, the sideslip angle is zero so that the velocity is aligned with the flight path.

4.3.4 Drag

To estimate the drag of the aircraft, an empirical model was used; details of this model can be found in [3]. First, the model assumes a parabolic drag polar with constant coefficients of the form

$$\mathrm{C_D} = \mathrm{C_{Do}(M) + K(M)C_L^2} \tag{4.5}$$

where C_{Do} and K are assumed constant since the low speed of airplane removes the dependence on M. To calculate C_{Do}, the equivalent parasite area method was used. That is, the contribution of drag from friction was summed according to

$$C_{Do} = C_{Df} = \sum_k \frac{f_k}{S_W} \tag{4.6}$$

where f_k is the equivalent parasite for each component of the airplane; f_k is determined using the average skin friction coefficient, an interference factor accounting for imperfect connections between components (e.g. wing connection to the fuselage), and a form factor that accounts for thickness effects. The induced drag coefficient, K, was calculated using the standard form

$$\mathrm{K} = \frac{1}{\pi \mathrm{A_W e}} \tag{4.7}$$

where A_W is the aspect ratio of the wing and e is Oswald's efficiency factor.

4.3.5 Wind

In order to truly optimize the aircraft, the wind velocity had to be accounted for in the program. The average wind speed in Wichita, Kansas during the month of April is 16 mph and so it was decided that the program would use a wind velocity of 11 mph in line with the runway. This velocity was assumed in order to account for the uncertainty in the direction of the velocity vector. The team decided to assume that, on average, the velocity would point 45 degrees relative to the plane's takeoff direction and thus the wind

velocity parallel to the plane's flight path would be $16/\sqrt{2}$ or 11 mph. The program modeled the flight path for each aircraft configuration assuming the wind to be solely parallel to the runway. The cross winds were neglected because it was assumed that their effect on the overall flight time would be negligible and that the cross winds would merely be trimmed out by the control system.

4.3.6 Servos

It was assumed that each aircraft configuration would use 6 servos: 1 for the front gear, 2 for the flaperons, 2 for the rudders, and 1 for the elevator. For the Futaba servos that the team uses, the weight per servo was 2.4 oz. The team considered only using 4 servos and using only the rudder for control on the ground (eliminating the front gear servo); however, the pilot was uncomfortable piloting the plane on the ground with just the rudder. After some debate it was determined that the weight savings and thus RAC savings were not sufficient to justify increasing the size of the rudders and inconveniencing the pilot.

4.3.7 Commercial Landing Gear/Wheels

The team decided early on that it did not have the necessary skills to construct the landing gear strut or the wheels themselves. The team also decided that it would be easier to replace a landing gear strut in the case of a hard landing than having to repair the fuselage or the landing gear connection with the fuselage. To ensure that the strut bent before the fuselage or the gear connection broke, the struts that were selected were light, but sturdy enough to take most landings. The team selected 3-inch diameter rubber wheels that weigh 1.7 oz. apiece. The front strut selected weighs 2.2 oz. and main gear struts that weigh 3.2 oz. apiece. The landing gear strut is labeled in the detailed design drawing package.

5.0 DETAIL DESIGN

5.1 Program Output

Once preliminary design had chosen several design parameters and made certain assumptions, the optimization code was written taking these simplifications and assumptions into account. The program was then run in order to choose the optimal dimensions and parameter values. The program output is shown in Table 5.1.

Table 5.1: Program Output Parameters.

GENERAL PARARMETERS	
Empty Aircraft Weight	13.6 lb
Total Flight Time (per mission)	300 sec
Cruise Velocity	60 mph
Number of Balls	24
WING PARAMETERS	
Chord	1.15 ft
Wing Span	10 ft
Taper Ratio	1
Spar Height	1.5 in
Spar Thickness	0.5 in
LANDING GEAR PARAMETERS	
Main Gear Extension Diameter	.25 in
Front Gear Extension Diameter	.25 in
Gear Height (from motor center)	9 in
PROPULSION PARAMETERS	
# of Motors	1
Batteries in Parallel / Series	All in Series
Static Thrust	6.29 lbf
# of Cells	24
Amps	40
Pitch Velocity, V_p	80 mph
Battery Usage (amp-hours)	2.1
AERODYNAMIC PARAMETERS	
$C_{L,CRUISE}$	0.4
$C_{L,TO}$	0.67
$C_{L,LANDING}$	1.2
C_{do}	0.031
K_{wing}	0.055
$K_{fuselage}$	0.108

5.2 Propulsion System

The output of the optimizer program suggested that the optimal propulsion configuration for the airplane would have 1 motor. The optimal static thrust was calculated to be 6.3 lbf (100 oz) where the static thrust is thrust the propulsion system would produce at zero airspeed. The optimal pitch speed of the propeller was determined to be 80 mph where the pitch speed of the propeller is the speed at which the thrust provided by the system is half of the static thrust. In most cases, the pitch speed is also approximately the speed at which the thrust becomes a nonlinear function of velocity. The optimal number of batteries was determined to be 24 batteries from DBF2002. The optimizer program also suggested that as the amp loading reaches maximum loading (40 amps), the score increases. The final configuration drained 2.1 amp hours from the 2.4 amp hour battery. According to the program, the batteries have approximately 27 seconds of reserve flight time allowing the pilot flexibility in flying.

The rules of the competition require the motors be brushed Astroflight or Graupner motors. The team investigated 55 different Astroflight motor/gearbox combinations. The team also investigated 89 different Graupner motors. These motors with their respective gearbox were examined in an off the shelf computer program named Motocalc.

Propellers with pitch and diameters between 1 and 50 inches were entered into Motocalc. Next, the battery specifications were entered into Motocalc. The choices that resulted in final motor temperature less than 250 ^{0}F were set aside for further examination. These motors were then analyzed to determine if they could provide a minimum 85% of ideal static thrust and pitch velocity.

The resulting choices were then analyzed to ensure that the pitch/diameter ratio did not exceed .8. A propeller with a pitch to diameter ratio larger than .7 is initially stalled. Since the expected head wind at Wichita is estimated to be roughly 11 mph, a propeller could be stalled up to this airspeed. Ideally, the pitch speed of a propeller increases with the stall speed of the propeller. Therefore, a propulsion system with a propeller stall speed slightly below the head wind speed would result in the fastest airplane configuration. Table 5.2 shows properties of the motors that were able to provide 85% of the ideal pitch speed and ideal thrust while maintaining a motor temperature less than 250 ^{0}F.

Table 5.2: Motor / Propeller Configuration Statistics.

	Motor	Gearing	Diameter (in)	Pitch (in)	Amps	Motor Weight (oz)	Thrust (oz)	Pitch Speed (mph)	Motor Temp (F)
Astroflight Motors	C60P #661	1	13	10	27.5	22	91	73	210
	FAI60 #660	1	10	5	37.2	19.5	99	73	241
	FAI60G #660G	1.63	12	9	34	19.5	99	82	232
Graupner Motors	Graupner Ultra 1800-5 Neo 18V #3311 '94	1	10	6	39.7	18.6	101	84	231
	Graupner Ultra 2000-7 24V #6332	1	12	9	31	22.6	94	80	193
	Graupner Ultra 3300-7H 20V #6362	1	11	6	35.4	18.7	104	72	235
	Graupner Ultra 3450-7 28V #6345	1	13	9	32.7	25.8	118	77	224
	Graupner Ultra 3500-8 30V #6346	1	14	11	30.8	28.1	103	74	195

A figure of merit matrix was created to determine the optimal motor. The best motor for the airplane will meet or exceed both the pitch speed and the static thrust required by the optimizer program. Some other important considerations are the propeller diameter, motor weight and engine temperature. Table 5.3 shows each Figure of Merit and its multiplier value.

Table 5.3: Figure of Merit Weighting.

Figure of Merit	Multiplier
Static Thrust	5
Pitch Speed	5
Propeller Diameter	3
Motor Weight	2
Engine Temperature	1

The static thrust and the pitch speed of the airplane were determined to be most important because the optimizer program sized the airplane using these critical assumptions. For this reason, a multiplier of 5 was placed for these figures of merit. A motor that exceeds the expected values of 100 oz static thrust and 80mph pitch speed would be the best motor. Dividing the motor value by the largest values normalized values for the static thrust and pitch speed.

The propeller diameter determines the length of the nose gear. A larger propeller results in a larger chance of ground strike by the propeller; these ground strikes can damage the propellers performance over time. The ground strikes can, however, be avoided by increasing the length of the nose gear. Increasing the length of the nose gear, however, increases the bending moment on the nose gear. A larger bending moment requires the nose gear to be built stronger which, in the end, will result in a heavier nose gear. Furthermore, a larger propeller dramatically increases its weight. Both of the effects result in added weight on the airplane as a whole. For this reason, a multiplier of 3 was placed on the diameter of the propeller. A value of .05 was subtracted from the score for every inch that the propeller is greater than the smallest propeller.

The motor weight significantly affects the weight of the airplane thereby decreasing the payload capacity of the airplane and also increasing the airplanes Rated Aircraft Cost. For this reason, a multiplier of 2 was placed on the engine weight. The motor weight score was calculated by dividing the lightest motor weight by the weight of the motor being scored.

The engine temperature affects the motor's lifetime and performance and also poses a hazard to the surrounding structure of the airplane. A large portion of the airplane is constructed using Monokote and Styrofoam. Although 250 ^{0}F is below the melting point of both materials, the temperature effects over time could be serious. Therefore, a motor with a lower running temperature is preferred. For this reason a multiplier of 1 was placed on the engine temperature. Dividing the coolest motor running temperature by the motor temperature gives the motor temperature score. The final figure of merit matrix is shown in table 5.4.

Table 5.4: Propulsion Figure of Merit Scores.

Component	Static Thrust (x5)	Pitch Velocity (x5)	Propeller Size (x3)	Engine Weight (x2)	Motor Temperature (x1)	Final Score
Astroflight C60P #661	0.77	0.87	0.85	0.85	0.92	13.36
Astroflight FAI60 #660	0.84	0.87	1.00	0.95	0.80	14.25
Astroflight FAI60G #660G	0.84	0.98	0.90	0.95	0.83	14.52
Graupner Ultra 1800-5 Neo 18V #3311'94	0.86	1.00	1.00	1.00	0.84	15.12
Graupner Ultra 2000-7 24V #6332	0.80	0.95	0.90	0.82	1.00	14.09
Graupner Ultra 3300-7H 20V #6362	0.88	0.86	0.95	0.99	0.82	14.35
Graupner Ultra 3450-7 28V #6345	1.00	0.92	0.85	0.72	0.86	14.44
Graupner Ultra 3500-8 30V #6346	0.87	0.88	0.80	0.66	0.99	13.48

The Graupner Ultra 1800-5 Neo 18V #3311'94 was determined to be the best motor for the airplane. After further analysis, however, this motor was discredited because the Graupner Ultra 1800-5 Neo 18V #3311'94 contains Neodymium magnets. These magnets are extremely powerful for their size enabling the motor to be lightweight. Unfortunately, these magnets are also very sensitive to heat. Therefore, this motor will not be chosen because of the relatively high operating temperature of the motor. The next best motor was the Astroflight FAI60G #660 motor with 1.63 gearing. The motor, however, was found to be discontinued, therefore, this motor could not be chosen. The third best motor was the Graupner Ultra 3450-7 28V #6345. This motor was chosen because it had the best score in the FOM matrix of those available.

5.3 Structural Systems

5.3.1 Structural Testing

In order to verify the structural strength of the built spar, a three-point load test was conducted on the fiberglass-laminated spar. In order to make the wing spar 54 inches, two pieces of balsa had to be glued together because the balsa could only be ordered with a maximum length of 48 inches. It was discovered during testing that a significant shear weakness existed at the square joint connection point between two pieces; proving to be the source of early failure of the spar. To alleviate the stress concentration, the connection was changed from a square connection to a 45^0 cut and then epoxied together. This shifted the distribution of the shear stress and strengthened the connection, allowing the spar design to hold the expected load from the mechanics of materials solution.

5.3.2 Landing Gear Analysis

In order to predict the structural integrity of the landing gear for this year's airplane, a model was constructed in ABAQUS and a static analysis was performed. The static analysis modeled a 'hard' landing using a point load applied at the landing gear wheel location. The angle of the load applied relative to the landing gear leg was set to 15^0, the maximum angle that could occur before the wing would strike the ground.

The model created used a balsa plate that represented a portion of the fuselage's side plate, which stretched between two shear webs since the load would be concentrated in this region. The steel leg was fixed to the plate in the region of contact. This assumption is valid since epoxy was used to fix the leg to the plate; the epoxy is stiff enough that any displacement (short of failure) will be very small. The plate uses a shell element model while the landing gear leg uses a Timeshenko beam model (since shear deformation may not be neglected under the extreme loading). Figure 5.1 illustrates the setup of the analysis while Figure 5.2 shows the results.

Figure 5.2 is a field plot of the Mises stress that developed in the plate under a 70 lb. loading oriented 15^0. Mises stress was used since it is a well-accepted criterion for predicting failure. As can be seen from the plot, the stress is significant at the joint of the steel rod and the balsa plate. The magnitude of the Mises stress indicates that severe failure will occur in the plate since the Mises yield stress of balsa was calculated to be 1.1 ksi. It was concluded, based on this simulation, that in order for the balsa to be able to withstand a 'hard' landing, a laminate would be needed to strengthen the balsa. Fiberglass was chosen for its strength and lightweight as well as its relative ease of application. Coupled with the added strength of the carbon fiber on the outer skin, the fiber glass should provide adequate strength for 'hard' landings.

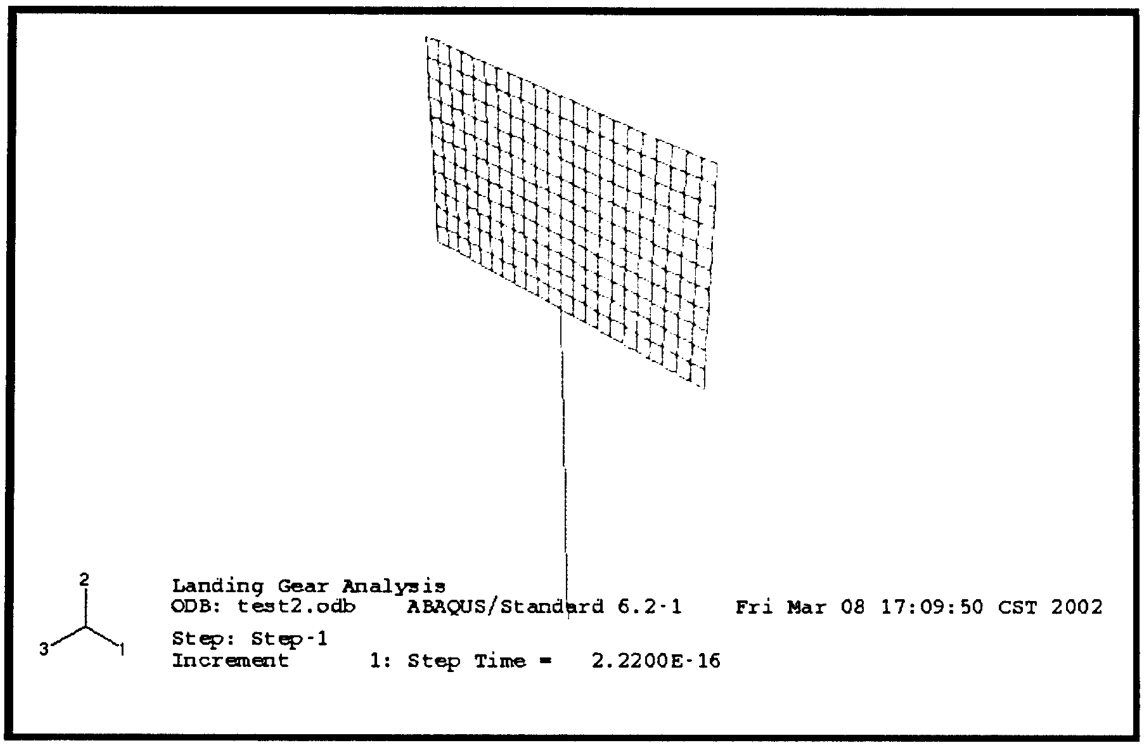


Figure 5.1: Landing Gear Mesh

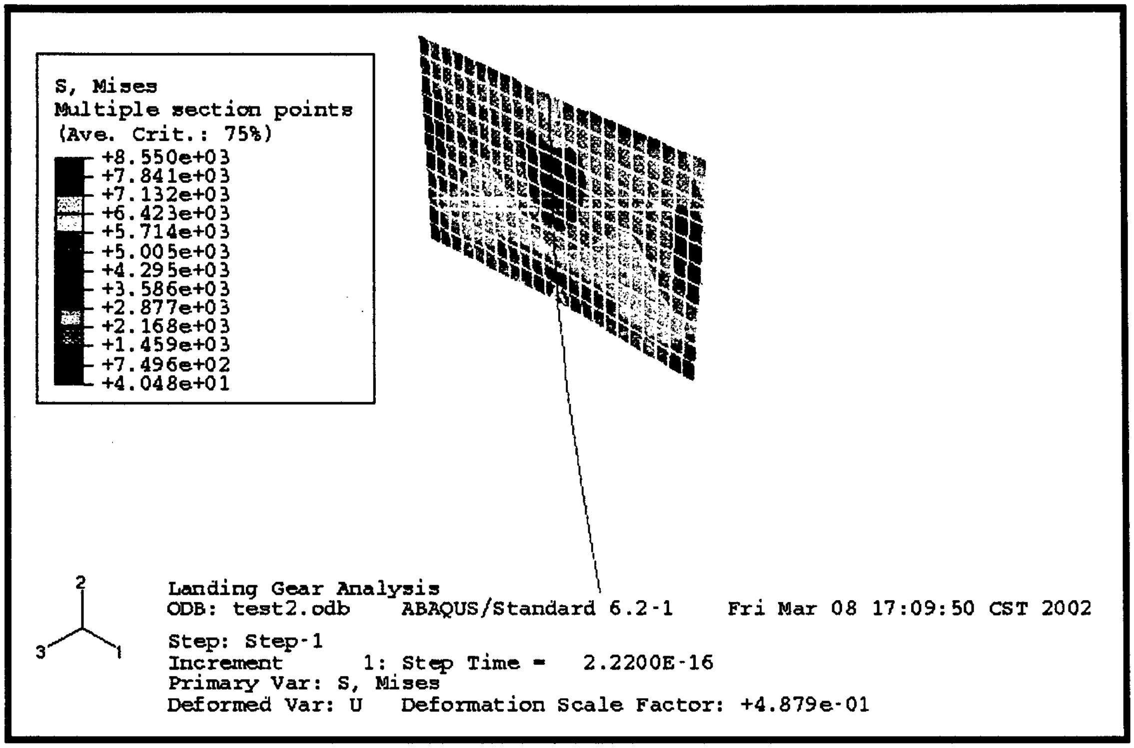


Figure 5.2: Stress Distribution of Balsa Plate Under 70 lb. Vertical Loading

5.4 Aeronautics System

5.4.1 Deviation from Originally Selected Tail

The original tail configuration consisted of a conventional tail; that is, the horizontal tail emanating from sides of the fuselage and the vertical tail centered on the fuselage. It was later discovered that the tail size dictated by the program resulted in a horizontal tail that was more than 25% of the wing span, which means the tail would be considered a wing, making the airplane much more expensive in the cost model. To size down the horizontal tail in this configuration would result in an inadequate control surface. It was then decided to change the configuration of the tail to 'quasi-T-tail' configuration; that is, a two vertical tail configuration with the horizontal tail on top. The drawings more clearly illustrate this configuration.

Two vertical tails were used so that balsa could be used as a spar material where as a single tail configuration would require a stronger material which would increase the weight. Also, the angular deflection of the vertical tail will be significantly decreased in a crosswind so that the horizontal tail will still be effective. Furthermore, the sturdy part of the fuselage is in the side plates, so the tails could easily be relocated without significant redesign of the fuselage.

5.4.2 Fuselage-Wing Fairing

A fairing on the wing-fuselage connection was introduced to avoid significant interference effects. To begin sizing the fairing, the wing airfoil was increased in scale so as to remain within the side plate dimensions and then linearly tapered to the wing size. The fairing was also constrained in the span-wise direction so that the width of the fuselage did not exceed 18" as it would be considered a wing and dramatically increase the RAC of the aircraft. The fairing would be relatively easy to construct from Styrofoam given the teams experience with hotwire cutting. Once the faring was cut, it could be laminated with Monokote and become permanently affixed to the fuselage sideplate. The wing could then connect through the fairing and Monokote would seal the seam between the connection. The 3-view drawing more clearly illustrates the fairing.

5.4.3 Control System

To size the elevator, it was desired to maximize ease of control for the pilot, especially with the expected wind speeds in Kansas. Hence, when in steady level flight, no elevator deflection should be required to maintain level flight. This condition can be achieved by examining the equations of motion.

Dividing the steady, level flight equations of motion by the dynamic pressure and substituting in the stability derivatives for the aerodynamic coefficients yields

$$\begin{aligned} &C_T - C_{D0} - KC_L^2 - (W/\bar{q}S_W)\gamma = 0 \\ &C_{Lo} + C_{L\alpha}\alpha + C_{L\delta_E}\delta_E - W/\bar{q}S_W = 0 \\ &C_m = C_{mo} + C_{m\alpha}\alpha + C_{m\delta_E}\delta_E = 0 \end{aligned}$$

For level flight, $\gamma = 0$ and the equations can be solved for α and δ_E:

$$\alpha = \frac{C_{m\delta_E}\delta_E(C_L - C_{Lo}) + C_{L\delta_E}C_{mo}}{C_{L\alpha}C_{m\delta_E} - C_{m\alpha}C_{L\delta_E}}$$

$$\delta_E = \frac{-C_{L\alpha}C_{mo} - C_{m\alpha}(C_L - C_{Lo})}{C_{L\alpha}C_{m\delta_E} - C_{m\alpha}C_{L\delta_E}}$$

where: α = angle of attack (radians)

δ_E = elevator deflection angle (radians)

From these equations, the deflection of the elevator could be calculated for each configuration and, hence, the plane could be designed for ease of control while still maintaining a statically stable aircraft.

The rudder for the aircraft was constrained by the fuselage and the horizontal tail. Sufficient clearance for the horizontal tail was required; this determined the limit on the upper dimension of the rudder. The lower limit was determined by an estimate of the boundary layer growth over the fuselage. That is, it was desired to keep the rudder out of the boundary to suppress any adverse effects that would come about from the rudder interfering with the boundary layer. Thus, the lower limit of the rudder was set to the edge of the boundary layer.

5.4.4 Airfoil Selection

With the sizing parameters complete from DBF2002, the airfoil for the wing had to be selected. Not only must the airfoil of the wing satisfy the stability requirements, but must also generate the required lift of the airplane. Furthermore, the previous year's team experienced a great deal of trouble cutting the airfoil trailing edge, so the trailing edge must also be sufficiently thick so the shape could be cut with some uniformity.

To select an airfoil, a search was done through the UIUC online catalog of airfoils [4]. Searching through the database and comparing to airfoil shapes, the list was narrowed to two airfoils for comparison: the Clary YSM and the SD7037. Both were moderately thin airfoils that could potentially produce enough lift, but still allowed for ease of cutting the trailing edge. Figures 5.3 and 5.4 below illustrate properties of each airfoil.

While the drag characteristics of the SD7037 are slightly better, the lift characteristics of the Clark-YSM are far superior. Not only is the stall angle larger for the Clark-YSM, but the lift slope is also greater. Hence, the airplane can achieve the required lift with a smaller angle of attack. While the angle of attack may seem insignificant when comparing the drag of the two airfoils, the effect on the rest of the airplane will be quite significant as the fuselage is an airfoil shape. Hence, the lower the angle of attack of the fuselage, the less chance there is for severe separation over the fuselage, which may lead to deep stall. Based on these characteristics, the Clark-YSM was chosen for this year's wing airfoil.

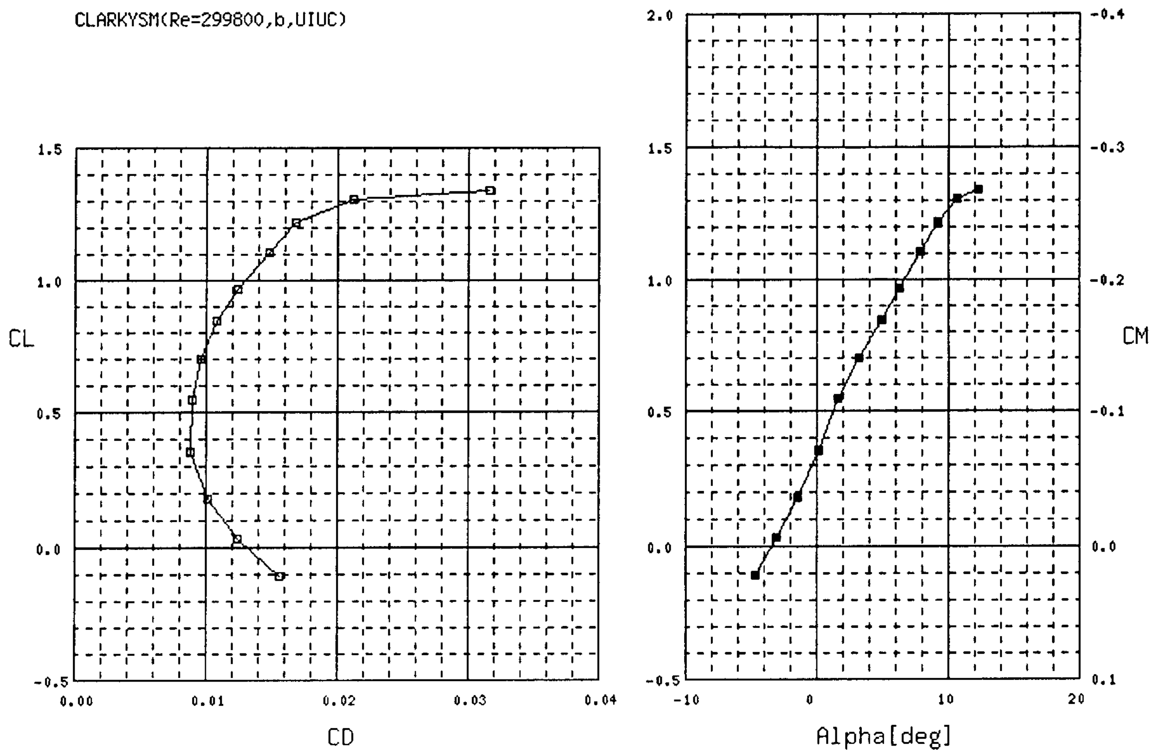


Figure 5.3: Clary-YSM [4]

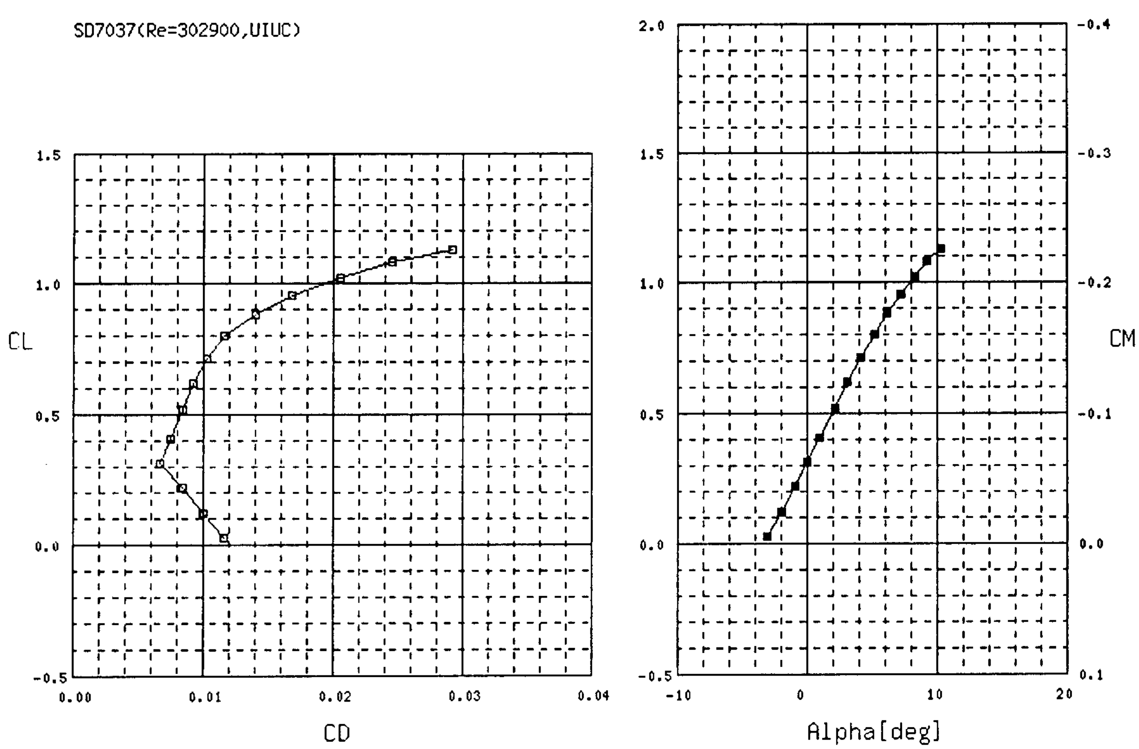


Figure 5.4: SD7037 [4]

It should be noted to obtain the lift coefficient necessary for landing would require a very high angle of attack. To compensate for the lift deficiency at landing, the ailerons will also serve as flaps (flaperons) to achieve the necessary lift coefficient so that the airplane can approach at a lower velocity for landing. Separate flaps were not considered as they would increase the cost of airplane to dramatically to justify their existence.

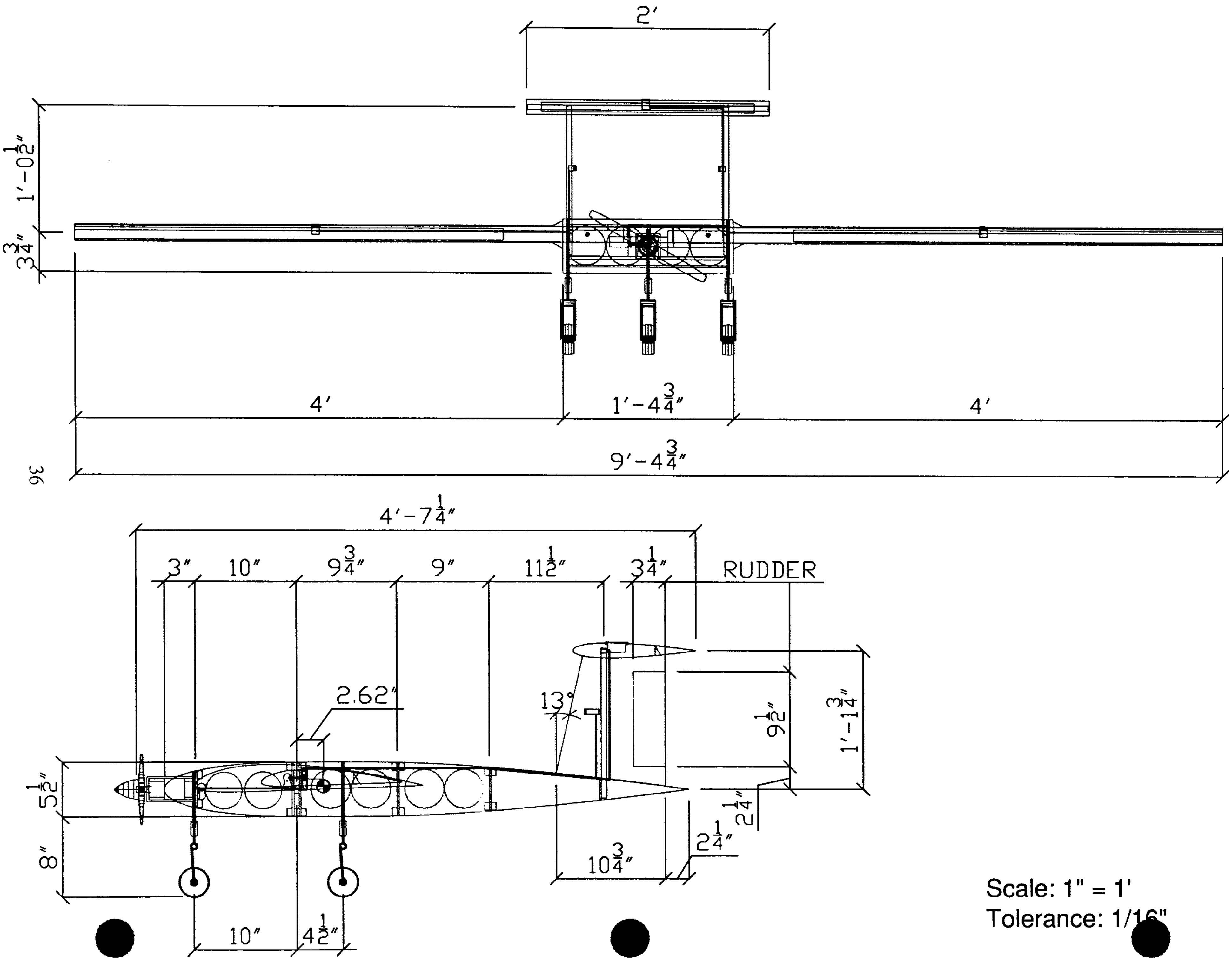
5.5 Performance Calculations

5.5.1 Final Configuration

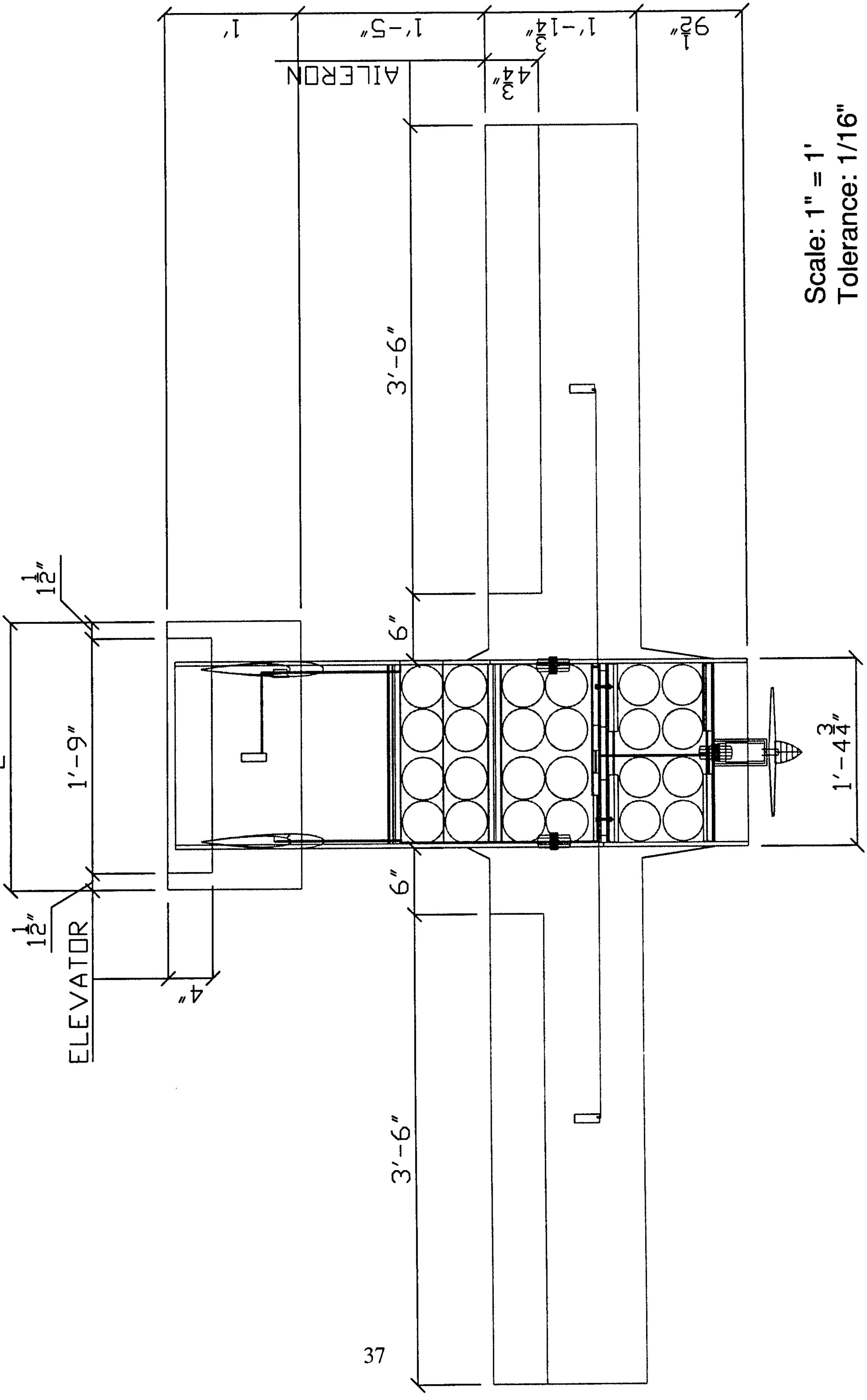
The final aircraft configuration is shown in Table 5.5 and in the following drawing package.

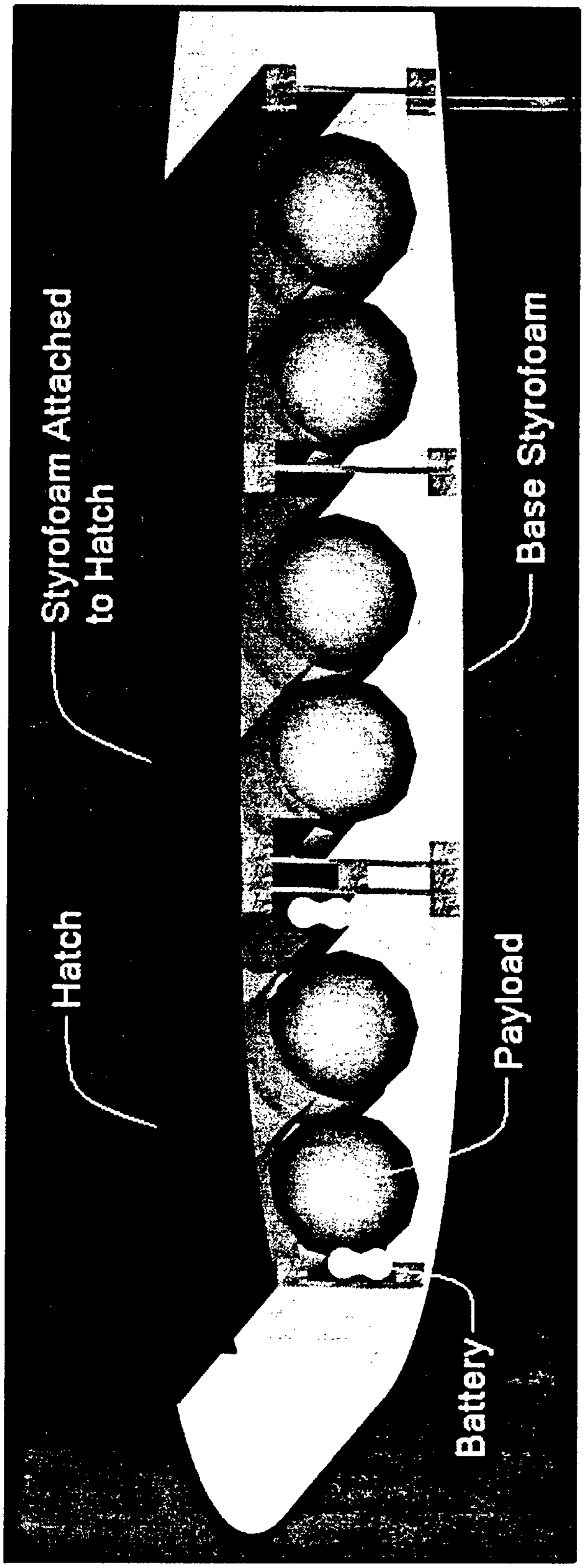
Table 5.5: Final Aircraft Data.

DIMENSIONS	
Wing Span	10 ft
Chord Length	1.15 ft
Wing Area	11.5 ft^2
Flaperon Area (total)	2.625 ft^2
Horizontal Tail Span	2 ft
Horizontal Tail Chord	1 ft
Horizontal Tail Area	2 ft^2
Elevator Area	.58 ft^2
Vertical Tail Height (quarter chord)	1 ft
Vertical Tail Chord (MAC)	.785 ft
Vertical Tail Area (total)	1.57 ft^2
Rudder Area (total)	.42 ft^2
Aspect Ratio	8.7
AIRFOIL DATA	
Wing Airfoil	Clark Y
Fuselage Airfoil	Compound NACA 0012
Horizontal Tail Airfoil	NACA 0012
Vertical Tail Airfoil	NACA 0009
MISCELLANEOUS	
Center of Gravity - empty (from quarter chord)	1.14
Center of Gravity - loaded (from quarter chord)	2.62
Soft Ball Capacity	24
Empty Weight	14.2 lb
Loaded Weight	23.6 lb
Motor	Graupner 3450-7
Propeller	13 x 9
Static Thrust	118 oz
Pitch Velocity	78 mph
Rated Aircraft Cost	10.633
Predicted Flight Time per Mission	300 sec
Total Predicted Flight Score	18.0

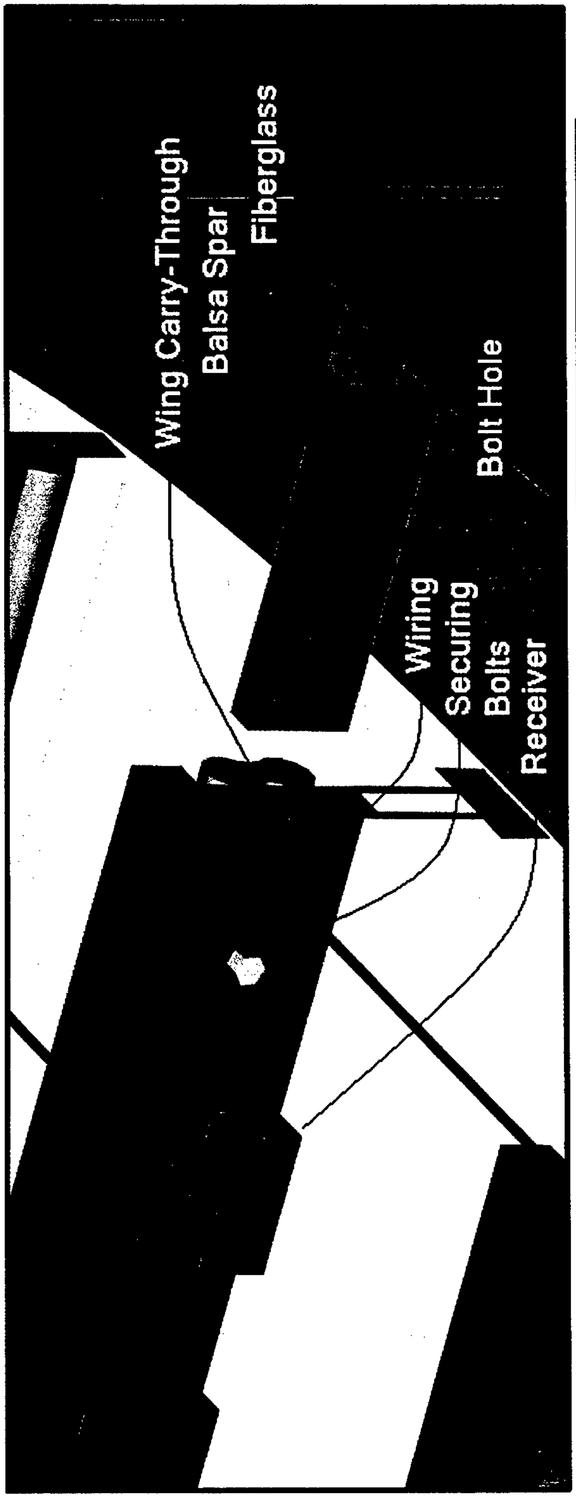


Scale: 1" = 1'
Tolerance: 1/16"

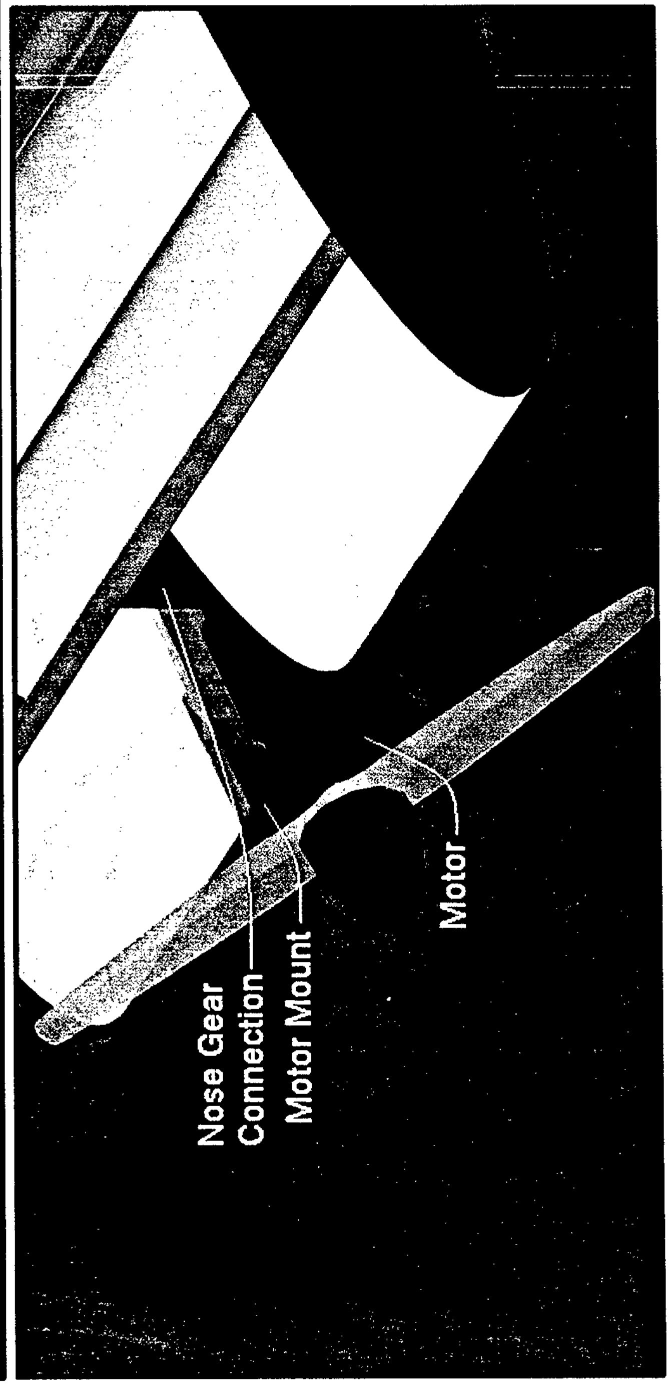

AILERON
1'
1'-5"
1'-1 3/4"
9 1/2"
4 3/4"
3'-6"
6"
6"
3'-6"
2'
1'-9"
1 1/2"
1 1/2"
ELEVATOR
4"
1'-4 3/4"
Scale: 1" = 1'
Tolerance: 1/16"


Hatch
Styrofoam Attached to Hatch
Payload
Base Styrofoam
Battery

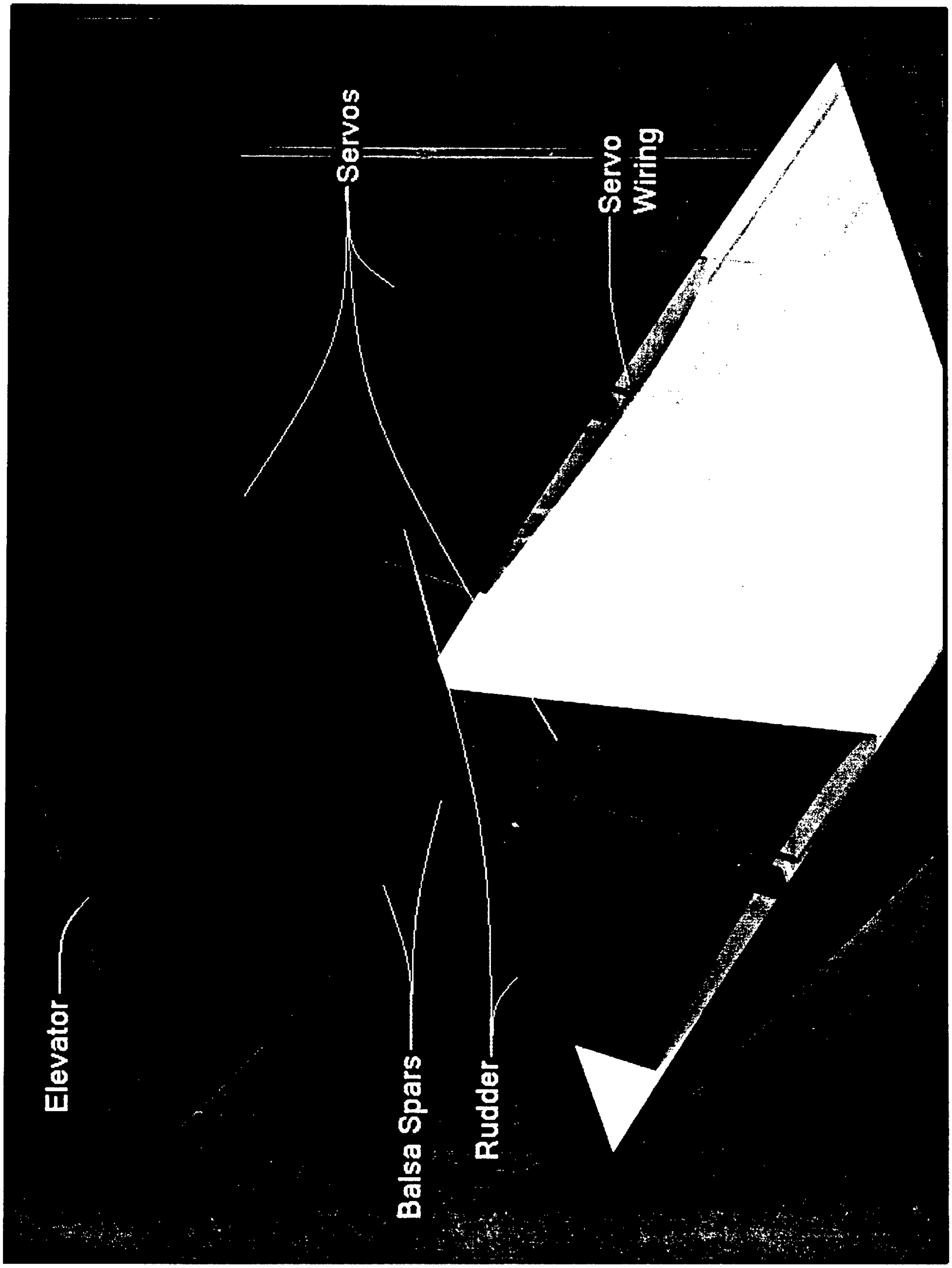

Landing Gear Extension
Collar
Fuselage Side Plate
Wing Fillet
Spring
Landing Gear Strut
Wheel



Wing Carry-Through
Balsa Spar
Fiberglass
Bolt Hole
Wiring
Securing
Bolts
Receiver



Nose Gear
Connection
Motor Mount
Motor


Elevator
Balsa Spars
Rudder
Servos
Servo Wiring

5.5.2 Weight and Balance Sheet

In order to estimate the CG position of the plane, a weight and balance spreadsheet was created shown in Table 5.6. The total empty weight of the plane calculated from the individual parts was 14.1 pounds, in good agreement with the scale measurement for the entire plane, which was 14.2 pounds. The extra weight is most likely due to uncertainties in the weights measured. The empty aircraft CG position found was 1.14 inches behind the quarter chord of the wing and the loaded aircraft CG position was 2.62 inches behind. The CG of the aircraft moves forward towards the quarter chord of the aircraft when the balls (the payload) are removed. This has two advantages; first, it increases the static margin of the aircraft when it is lighter, thereby increasing the stability of the plane, and second it increases the maneuverability of the aircraft when it is loaded, allowing the aircraft to still be controllable.

Table 5.6: Weight and Balance Spreadsheet.

	Weight [oz]	Position [in]
Wing and Spar	25.0	0.250
Wing Servos and Wiring	4.8	1.900
Vertical Tail	6.0	30.400
Vertical Tail Servos and Wiring	4.3	23.000
Horizontal Tail	5.0	30.400
Horizontal Tail Servo and Wiring	2.6	24.000
Front Gear	10.0	-10.250
Front Gear Servo and Wiring	1.4	-9.000
Main Gear	18.0	4.500
Motor	26.3	-12.250
Motor Controller	1.3	-9.500
Propeller	2.3	-13.000
Longeron/Motor Mount	0.9	-6.500
Radio/Battery	4.4	0.625
Fuselage	28.3	6.500
Fuselage Side Plates	24.0	8.000
Hatch	9.0	4.000
Front Batteries	26.0	-9.500
Rear Batteries	26.0	-0.625
Payload	152.0	4.800
EMPTY TOTAL WEIGHT (lb)	14.10	
UN-LOADED CG (in)	1.14	
LOADED TOTAL WEIGHT (lb)	23.60	
LOADED CG (in)	2.62	

5.5.3 RAC Calculation

An aircraft cost model is given in the contest rules in order to scale the aircraft. As stated in the rules the Rated Aircraft Cost, or RAC, is calculated by the following:

$$RAC = \frac{(100 * MEW + 1500 * REP + 20 * MFHR)}{1000}$$

where *MEW* is the Manufacturers Empty Weight, *REP* is the Rated Engine Power, and *MFHR* is the Manufacturing Man Hours; all of which are defined in the contest rules. Table 5.7 shows the calculated RAC for the final aircraft design.

Table 5.7: RAC breakdown.

Description	Input
MEW	14.2
REP	3.250
# of Engines	1
Total Battery Weight (lb)	3.25
MFHR	216.9
Wings	**95.2**
Wing Span (ft)	10
Max Chord (ft)	1.15
# of Control Surfaces	2
Fuselage	**46.7**
Fuselage Length (ft)	4.67
Empennage	**30**
Vertical Surfaces w/no control	0
Vertical Surfaces w/ control	2
Horizontal Surfaces	1
Flight Systems	**35**
# of Servos	6
# of Motor Controllers	1
Propulsion Systems	**10**
# Engines	1
# Propellers	1
TOTAL RATED AIRCRAFT COST	10.633

5.5.4 Take-Off Performance

From the output of DBF2002, the predicted take-off distance of the loaded airplane is 150 ft. in 0 mph winds. To achieve this take-off performance, the pilot will perform a small rotation of the airplane at 150 ft., which will produce the lift necessary for the airplane take-off. The flaperons will not be necessary at take-off, only for landing so the plane can approach at an acceptable velocity. This case is the limiting one for take-off distance since any wind speed will decrease the take-off distance as well as the decrease in mass for the unloaded portion of the mission. Also, the designed 150 ft. take-off distance produces a safety factor accounting for uncertainties in the drag calculation since the required take-off distance for the contest is 200 ft.

5.5.5 Turning Flight

Since it was decided that the wing tip test at the competition would be the most severe load encountered and the wing tip test is approximated as a 2.5 g load, the turning performance of the airplane was designed for a 2.5 g loading in the turns. The program takes the 2.5 g loading into account when sizing the spar necessary to withstand the structural load. To confirm this assumption is valid, the lift distribution over the wing was approximated as

$$L = \int_0^L p_o \cos\left(\frac{\pi x}{2L}\right) dx$$

p_o was calculated using the yield stress of the spar material and Bernoulli-Euler beam theory. Evaluating for the loaded airplane weight and the given lift distribution yields a theoretical g loading of 2.89. Hence, the assumption of a 2.5 g turn is acceptable in the program.

5.5.6 Expected Endurance

From the DBF2002 output, the expected flight time is approximately 300 seconds, which includes 40 seconds of ground time, 30 seconds of glide time, and the rest powered. The selected batteries provide the power required as well an estimated 27 extra seconds of full throttle yielding a very nice safety factor for deviations from the flight plan that will likely be encountered in the high wind environment.

5.5.7 Flight Testing

The first flight test of the airplane was met with little success. Upon lift-off, a severe gust of wind caused the airplane to stall near the ground and coupled with pilot error, caused the airplane to crash shortly after take-off. However, even with the crash, some performance characteristics could be found. First, the take-off distance in the 20 mph winds was well within expectations. Second, following the stall, the airplane nosed down implying it is statically stable, which qualitatively verified the stability calculations.

Due to inclement weather and schedule conflicts with the pilot, subsequent flight tests at the time of the report have yet to be done. The team is confident, however, that the airplane will perform according to design specifications once the pilot has had ample opportunity to practice.

5.6 Handling Qualities

5.6.1 Longitudinal Static Stability

The previous year's DBF team experienced severe stability problems with the aircraft. Thus, it was imperative that improvements be made in stability predictions of the aircraft. To perform stability calculations, a MATLAB code SS1 was written. This code took parameters from the DBF2002 output coupled with selected aerodynamic properties to calculate stability parameters.

The most important parameter in predicting the aerodynamic response, longitudinally, is the location of the aircraft aerodynamic center (X_{AC}) relative to its center of gravity (X_{CG}); the difference of X_{AC} and X_{CG} is termed the static margin. In order for the airplane to be statically stable, the static margin must be positive. Several geometric and aerodynamic properties must be taken into account in order to calculate the static margin.

First, the geometry of the airplane must be considered; that is, the location of the horizontal tail with respect to the wing. The horizontal and vertical dimensions of the horizontal tail aerodynamic center, L_h and H_h respectively, relative to the wing aerodynamic center are used to calculate the downwash from the wings and its effect on the horizontal tail. The downwash model is an empirical one and is given in [3]. Once the downwash components were calculated, these effects modified the lift stability derivatives of the wing and horizontal tail to give lift stability derivatives of the airplane. Also modified are the moment stability derivatives of the wing and horizontal tail to give the moment stability derivatives of the airplane.

Second, the size of the tail needed to be considered. The tail was sized according to the volume coefficient method. It was desired for the handling qualities to be at least somewhat similar to the pilot's own airplane. Thus, the volume coefficient for the horizontal tail was matched to the volume coefficient of the pilot's RC horizontal tail. With the volume coefficient and the distances relative to X_{CG}, the horizontal tail area could be calculated. Once the area was calculated, the chord length and span could be manipulated to size the elevator as desired (see 5.4.3).

With all these quantities, X_{AC} could be calculated according to [3]. Using this method to calculate X_{AC}, the static margin was calculated as .35. Hence, the airplane is statically stable. While this value does seem very high, the team decided a sluggish response would be desired with the considerable winds expected in Kansas in April.

5.6.2 Lateral Static Stability

As before with longitudinal static stability, the airplane must also be stable laterally; that is, the plane must return to its original operating condition when perturbed by a gust of wind from the side or any sideslip angle. To ensure lateral static stability, the total yawing moment stability derivative, $C_{n\beta}$, must be positive. The two major contributions to yawing stability stem from the fuselage and the vertical tails. The fuselage tends to be destabilizing while the vertical tail is stabilizing. Thus, the vertical tail must be sized such that lateral stability is maintained.

As before, the volume coefficient method was used to size the area of the vertical tail and the volume coefficient was matched to the volume coefficient of the pilot's RC vertical tail to attempt to mimic the behavior of the pilot's airplane. The chord and span dimensions were then limited by the placement of the horizontal tail up on the vertical tails while still keeping an acceptable rudder size.

Using empirical relations for the contribution of the fuselage and vertical tail to $C_{n\beta}$ from [5], the value of $C_{n\beta}$ was determined to be .22. Thus, the airplane is laterally stable.

5.6.3 Longitudinal Dynamic Stability

With the airplane now sized to maintain static stability, both longitudinally and laterally, the dynamic performance of the airplane needed to be predicted. Using steady, level flight assumptions, the stability derivatives were approximated using relations from [3]. Table 5.8 shows the calculated non-zero stability derivatives for the aircraft.

Table 5.8: Final Design Longitudinal Derivatives.

Longitudinal Derivatives			
Derivative	Value	Derivative	Value
$C_{L\alpha}$	5.74	C_{Lq}	6.01
$C_{L\dot{\alpha}}$	2.17	C_{mq}	-17.1
$C_{L\delta_E}$	.613	$C_{x\alpha}$	.121
$C_{D\alpha}$	.219	$C_{z\alpha}$	-5.78
$C_{m\alpha}$	-1.41	$C_{z\dot{\alpha}}$	-2.17
$C_{m\dot{\alpha}}$	-6.18	C_{zq}	-6.01
$C_{m\delta_E}$	-1.74	$C_{z\delta_E}$	-.613

Table 5.9: Final Design Eigenvalues.

	Eigenvalues
Short Period Mode	$-10.58 \pm 7.93i$
Phugoid Mode	$-.017 \pm .488i$

Correspondingly, the eigenvalues for the response of the airplane in steady, level flight are shown in Table 5.9. Both modes display convergence. The short period mode is highly damped and will be unobservable in flight.

One of the major difficulties in determining the stability derivatives was the calculation of the moments of inertia for the airplane. The facilities to conduct a dynamic measurement to estimate the inertia characteristics of the aircraft were insufficient at the time. The team resorted to AUTOCAD. AUTOCAD provides inertia calculations based on a unit mass. The moments were found for the balsa, Styrofoam, and softballs using AUTOCAD and then later scaled by the appropriate density and translated to the proper coordinate system. Assumed with this calculation is that the materials are homogeneous. While this is not physically true, it was determined that the assumption should provide reasonable results in estimating the dynamic performance of the airplane. The moments of inertia for the motor and landing gear were later approximated using the standard calculations.

5.6.4 Lateral Dynamic Stability

Again, with the airplane sized according to static stability considerations, the dynamic performance of the airplane needed to be estimated. Correlations from [5] were used to approximate the lateral stability derivatives. Table 5.10 below summarizes the lateral stability derivatives of the airplane.

Table 5.10: Final Design Lateral Derivatives.

Lateral Stability Derviatives			
Derivative	Value	Derivative	Value
$C_{y\beta}$	-.367	C_{yr}	.646
$C_{n\beta}$	.22	C_{nr}	-.371
$C_{l\beta}$	0	C_{lr}	.314
C_{yp}	-.02635	$C_{y\delta_r}$	.112
C_{np}	-.0425	$C_{n\delta_r}$	-.241
C_{lp}	-1.01	$C_{l\delta_r}$	.031

Also shown are the calculated eigenvalues for the response of the aircraft. The roll mode is highly convergent and the dutch roll mode is also convergent. The spiral mode displays divergence, but this is common amongst aerospace vehicles and should not present difficulties to an experience pilot.

Table 5.11: Final Design Lateral Eigenvalues.

	Eigenvalues
Spiral Mode	.0843
Roll Mode	-2.682
Dutch Roll Mode	-.723 ± .954

6.0 MANUFACTURING PLAN

The design and construction of the UT-DBF airplane was highly influenced by the experiences of previous years. The team, however, was still relatively new and still lacked machine shop skills so processes that required such skills were avoided if possible.

6.1 Figures of Merit

Figures of merit for each process included performance advantages inherent to the process, ease of construction, time required to construct, and cost. Some processes, like the construction of the wings, or the assembly of the empennage did not have alternative designs that were investigated.

6.1.1 Performance

Depending on the construction method of a particular component, it may enhance the component's performance or detract from it. Obviously, better performance is desired; this figure of merit was given a weight of 3.

6.1.2 Construction Skill Required

The inexperience of the team necessitated simple, easy construction techniques that could be accomplished without the use of machine shop tools. This figure of merit was given a weight of 2.

6.1.3 Construction Time Required

An important factor in the feasibility of each component's construction method was the time required to implement the method. Complicated methods would increase the time required to build the component. This figure of merit was given a weight of 2.

6.1.4 Cost

The team began the year with a strict budget in place that could not be exceeded. Because of the expensive nature of the propulsion systems, the construction methods needed to be as cost effective as possible in order to avoid running over budget. This figure of merit was given a weight of 1.

6.2 Manufacturing Processes Investigated

6.2.1 Wing Spar Core

The dimensions of the wing spar were such that no commercial piece of balsa could be bought and used without modification. The first choice was to use two separate pieces of balsa epoxied together. The only advantage to using the two separate pieces was the balsa needed had already been purchased the

previous year and was readily available. The single piece of balsa had to be cut down in order to fit the dimensions, but was stronger and was easier to build.

6.2.2 Spar Laminate Method

The spar design called for the spar to be laminated with fiberglass in order to provide torsional stiffness. The laminate could either be applied and then sealed in a vacuum bag, a process that requires skill and time, but yields lighter, stiffer spars. Alternatively, the laminate could be applied and allowed to cure without the use of the vacuum bag.

6.2.3 Hatch Construction

The two methods considered for the fabrication of the hatch were to make a template and cut Styrofoam shapes, or just simply overlay the carbon composite onto the fuselage. The template method would lighten the hatch, but increase the time and skill required because of the extra time needed to make the templates. The overlay method would not be as light because of the uneven distribution of epoxy that would result in the composite, but it would be easier.

6.2.4 Hatch Opening

Two different ways for the hatch to open were investigated: a standard pivot way in which the hatch would pivot along the width of the fuselage and a "Shuttle" like opening in which the hatch would fold open in two sections, each pivoting on a side plate. The shuttle method would be heavier and be more difficult to build; whereas the standard pivot (shown in the detail design drawing package) would be quick and relatively easy to construct.

6.2.5 Motor Mounts

Two different motor mount designs were considered: worm clamps that would secure the motor and an actual mount made of Liteply. Since the Graupner motor requires the body of the motor to be able to turn freely, the worm clamps were not effective in securing the motor tightly enough against vibrations. The Liteply motor mount allowed for the motor body to rotate and was skill relatively light and easy to construct compared to other potential options such as a milled aluminum mount.

Table 6.1: Manufacturing Plan Figures of Merit.

	Option	Performance (x3)	Skill Required(x2)	Time Required (x2)	Cost (x1)	Final Score
Core Spar	Two Balsa pieces	0.80	0.90	0.95	1.00	7.10
	Single piece	1.00	1.00	1.00	0.80	**7.80**
Spar Laminate	**Fiberglass**	0.90	1.00	1.00	0.00	**6.70**
	Vacuum Bag Fiberglass	1.00	0.80	0.80	0.00	6.20
Hatch Construction	**Make Template**	1.00	0.95	0.95	0.00	**6.80**
	Overlay on Fuselage	0.90	1.00	1.00	0.00	6.70
Hatch Opening	**Standard Pivot**	1.00	1.00	1.00	1.00	**8.00**
	"Shuttle" Opening	0.90	0.85	0.80	0.95	6.95
Motor Mounts	Worm Clamps	0.50	1.00	1.00	1.00	6.50
	Liteply Box	1.00	0.75	0.85	0.80	**7.00**

6.3 Final Manufacturing Process

6.3.1 Fuselage

The airplane fuselage was the first component to be constructed. During the 2000/2001 competition, a symmetrical NACA airfoil was used as the shape with a balsa shear web. This proved to be relatively easy to construct and incredibly strong for its weight. The shear web would provide strength for the bending moment caused by a edge loading. To provide for a bending moment by tip to tail load, a side plate would be constructed. A more traditional cylindrical or elliptical style fuselage was also considered, but dismissed for difficulty in construction and lack of accessibility.

This year's airplane used NACA0012 airfoil that was extended slightly in the middle to provide sufficient volume for the cargo. Also, its length is long enough to allow for the tail of the aircraft to be mounted directly onto the fuselage (without the need for an extension). The initial design of the fuselage was created on the computer using AutoCAD. Two different configurations for the softballs were considered. The optimization program had determined that the maximum twenty-four balls were required for a competitive score. Thus, 2 designs were considered: a three wide by eight-row, and a four wide by six-row configuration. Further analysis showed the four by six arrangement was better. Even with the rules limiting the fuselage to a maximum of eighteen inches, the four-ball arrangement allowed for a one-inch fillet between the wing edge and the fuselage. Also, the three-ball arrangement would require a longer fuselage. Although a lengthier fuselage that would be beneficial for the rear control surfaces, it would also greatly increase the rated aircraft cost. Lastly, the three-ball arrangement would require more structure to the rear of the airplane increasing the weight.

After the complete fuselage had been modeled on the computer, templates were designed and printed. The template designs were then glued to a sheet of formica with 3M Super77 spray adhesive and cut

using a scroll saw. The templates were positioned on either side of a block of Styrofoam, and a hot wire dragged along the templates to cut the different shapes. After all Styrofoam sections of the fuselage were created, the one half inch by one half inch balsa sticks that contributed to the shear web were cut using the scroll saw. The balsa sticks were then glued into their corresponding slots in the foam using epoxy. Also, the three-eighths inch thick balsa sheets for the shear web were measured, cut, and epoxied into place. To lower the weight, large holes were cut in the Styrofoam that didn't need the strength or shape.

The next task for the construction of the fuselage was the side plates. Since these required significant strength, carbon fiber laminated onto a three eighths inch thick balsa was considered ideal. Unfortunately, balsa sheets long enough for the airplane's fuselage could not be found. To minimize the risk of the sheet failing under the load, two balsa sheets were cut in such a way that their seam was along one of the fuselage's shear webs. Using AutoCAD, paper templates were created with the shape of the fuselage. The designs were traced onto the balsa sheet side plates, and the plates were then cut with the scroll saw. Two acceptable methods for laminating the carbon onto the side plates were available. The first was laying the carbon fabric with epoxy onto the plates and then applying sufficient distributed weight to guarantee a smooth surface. Although easy to do and efficient, more epoxy had to be used to ensure a fully wet surface, resulting in higher weight. The second method, which was used for the final design of the airplane, was the use of a vacuum bag. The carbon was epoxied onto the balsa sheet and then a plastic bag was wrapped around it a vacuum created. The vacuum bag created a smooth surface for the side plate and reduced excess epoxy. The vacuum pump remained on overnight while the epoxy cured to ensure all excess epoxy was removed from the plates. Once it had cured completely, the openings for the wing spar and servo wiring were measured and cut from the template using a dremmel.

It was determined the strongest method to attach the landing gear and Styrofoam fuselage sections to the side plates was the use of fiberglass. Fiberglass was epoxied across all sections of the side plate that would be in contact with the shear webs of the foam. A layer of fiberglass was epoxied below the steel rods of the landing gear and then an epoxy fillet was created with microfibers along the rod and then a second fiberglass sheet overlaid the gear. After placing the fiberglass, the Styrofoam fuselage was positioned onto the curing side plate. A foam insert ensured proper alignment between the wing box and the slot for the wing in the side plate. Weights and clamps were used to ensure complete contact. After a minimum of six hours, the weights and clamps were removed and the other side plate was attached the same way.

The last major component of the fuselage was the hatch to access the payload. Again, templates were printed from the computer and Styrofoam was cut to match the shape of the hatch. Packing tape was taped across all surfaces of the Styrofoam; this allowed carbon fiber to be laid across the curved surface without bonding to the Styrofoam mold. After having epoxy spread evenly across the carbon fiber, the top

section of the mold was weighted on top. A vacuum bag was not used since a good seal would not have been attainable. Again a minimum of six hours passed before the molds were removed. After curing was complete, excess carbon fibers were cut from the hatch. The fuselage could not be fully completed until the vertical tails were attached with their wiring. Grooves were cut from the Styrofoam between the tail and the radio receiver to conceal the wires. Also the engine mount and nose gear were attached to the front. The front of the fuselage was cut to round it with the wing's fillet. Finally a layer monokote was applied over the fuselage using a hot iron and heating gun as a smooth skin.

6.3.2 Wing

The airplane's wing spars were chosen to be made from one half inch thick by one and one half inch tall balsa sticks. Since no sticks were available the required length of fifty six inches, two sticks had to be epoxied together. The connecting location of the two balsa sticks were cut at a forty-five degree angle to minimize shear forces. Between four inches off the tip of the balsa stick to sixteen inches off the tip of the balsa stick two oz. fiberglass was laminated on all sides. The remaining length of the stick (with an additional one inch overlap over the two oz. fiberglass) seven tenths oz. fiberglass was laminated. During the curing process the sticks will be placed in a vacuum bag to ensure the maximum strength will be obtained from the fiberglass. The two oz. inner fiberglass increases the strength for the wing's bending moment and the remaining seven tenths fiberglass will reduce wing twisting and bowing.

Three designs were considered for the wing construction. The fastest and easiest option was to keep the wings solid foam but this option was also the heaviest. The second option was to create a solid foam wing, and cut small blocks from the wing to reduce weight. This increased the difficulty slightly over the first option, but was still relatively easy. Lastly, the wings could be constructed using an array of ribs that were each individually glued into place along the spar. Although the lightest, this method was considered too time consuming and difficult to balance out its advantages. The second option was decided as the best for the airplane. Templates for the wing were designed in AutoCAD and printed. Similar to the fuselage, the templates were cut out of formica and the wing shape was cut out of the Styrofoam. The templates also provided a space for the balsa spar to be placed inside of the wing. The full foam wings were cut out and then the servo wiring was run inside of the wing. Tape was used to cover the wiring from the epoxy used to hold the spar onto the wing. The control surfaces were also measured and cut from the wing using the hotwire.

To reduce weight, the final design of the plane had Styrofoam blocks cut from the wing. Monokote was then applied over the entire wing to ensure a smooth aerodynamic surface. The Monokote also covered the blocks of Styrofoam that had been removed. The control surface for the wing had a thirty degree angle cut along the edge to allow for the needed deflection; Monokote was applied here as well. A hinge

made of Monokote was used to attach the back control surface onto the wing; this created a strong hinge for the flaperon to rotate about. Finally, the servo was imbedded into place and the arm attached.

6.3.3 Empennage

The first component of the tail to be built was the two vertical surfaces. In AutoCAD, the geometry for the surface was finalized. Templates were then designed for not only the upper and lower ends of the vertical surface but also the shape of the fuselage beneath it. Again, similar to the fuselage sections and wings, the vertical surface was cut from Styrofoam. The templates of the fuselage were used to cut the bottom of the vertical to match the curve of the fuselage. The vertical surfaces consisted of two separate pieces of Styrofoam that had a one-quarter by one half-inch slot cut out down the center of each. When placing the two sheets together, a one half by one half-inch slot remained for the balsa spar. Servo placement was chosen and wiring was run along the inside of the tail. Next, the rudder was measured and cut from the vertical surface. Like the ailerons/flaps, an angle was cut out rudder for required deflection. Moving on, the horizontal surface needed to be completed next.

Templates were constructed similarly to the rudder with the one half by one half-inch spar. The top of the vertical surface was also cut to form the shape of the horizontal surface; this ensured a snug fit to reduce drag. The elevator was also measured and cut as necessary. Balsa sticks were measured and cut for both vertical surfaces and the horizontal surface. Once all components were completed, they were epoxied into place on the fuselage (with accuracy maintained using a level). Wiring for the elevator was strung along the inside the horizontal surface, and then down the side of the vertical surface in a trench that was cut. All servo wires converged to the fuselage and were hidden beneath Monokote. Now, the empennage could be completed with Monokote.

6.3.4 Landing Gear

Due to unusual design of the fuselage, landing gear choices were limited. The mid-section of the fuselage was relatively weak, and even if the gear were positioned at the shear web, it would very doubtful that the fuselage could support even a relatively mild landing. The only realistic placement for the main gear was along the side plates. A C-shaped landing gear would be impractical, because it would either need to be long enough to span across the entire undercarriage of the fuselage or be supported by the weak mid-section. A steel rod laminated with fiberglass across the entire height of the side plate, however, would be very strong and can easily be attached to the rod coming from the actual wheels. Another benefit of the steel rod would that it would bend before actually breaking the side plate or shearing from the side plate. This results in quicker repairs if a hard landing should occur. The steel rod method was chosen and its construction was discussed in the fuselage section more in-depth. Aluminum sleeves were machined to match the steel rods protruding from the fuselage to serve as connecting rods to the wheels.

The nose gear was much more complicated than that of the main gear because it had to be able to rotate. A steel pipe with three eighths inch inner diameter and one half-inch outer diameter was cut four inches long. Styrofoam from the front of the fuselage was removed to reveal the balsa shear web behind it. Fiberglass was used to reinforce the balsa sheet and then the steel pipe was epoxied onto it. An epoxy fillet with microfiber was used to strengthen the connection and a second fiberglass sheet was positioned again over the landing gear component. Nylon spacers were fitted inside the steel pipe so a tight fit was created for the landing gear rod but was still allowed to rotate. The landing gear steel rod had stoppers screwed below and above to maintain the gear position relative to the airplane.

6.3.5 Motor Mounts

To manufacture the motor mounts, appropriate dimensions were measured for the Liteply plates and then cut. Next, holes were drilled in the front plate to accommodate the shaft and securing screws for the motor. Then, the three plates were epoxied together at the edges and filleted with micro-fiber. Finally, the Liteply plates were epoxied to the front balsa plate of the fuselage to which the front landing gear also attaches.

6.4 Manufacturing Time Table

In order to meet the initial timeline's projections, the team leaders set a construction timetable. Table 6.2 shows the manufacturing timeline, which lists the major components of the aircraft and their associated construction time periods.

Table 6.2: Manufacturing Timeline.

TASK	START DATE	FINISH DATE	ACTUAL DATE COMPLETE
Fuselage	**2/1/02**	**2/13/02**	**2/15/02**
Make Templates	2/1/02	2/2/02	2/2/02
Build Wing Carry-Through	2/2/02	2/2/02	2/2/02
Cut Stryofoam	2/3/02	2/5/02	2/5/02
Assemble Stryofoam Sections	2/5/02	2/5/02	2/6/02
Construct Side Plates	2/6/02	2/7/02	2/7/02
Attach Side Plates / Main Gear	2/8/02	2/9/02	2/9/02
Build Motor Mount	2/9/02	2/9/02	2/10/02
Build Front Gear	2/9/02	2/9/02	2/11/02
Build Hatch	2/10/02	2/11/02	2/13/02
Monokote and Finish Fuselage	2/11/02	2/13/02	2/15/02
Wings	**2/1/02**	**2/8/02**	**2/8/02**
Make Templates	2/1/02	2/1/02	2/1/02
Make Wing Spar	2/1/02	2/2/02	2/2/02
Cut Stryofoam	2/2/02	2/2/02	2/2/02
Cut Control Surfaces	2/3/02	2/3/02	2/3/02
Add Servos / Wiring	2/4/02	2/5/02	2/6/02
Attach Wing Spar	2/5/02	2/6/02	2/6/02
Monokote and Finish Wings	2/7/02	2/8/02	2/8/02
Empennage	**2/8/02**	**2/16/02**	**2/17/02**
Make Templates	2/8/02	2/9/02	2/9/02
Cut Stryofoam	2/9/02	2/10/02	2/10/02
Cut Control Surfaces	2/10/02	2/10/02	2/11/02
Add Servos / Wiring	2/11/02	2/12/02	2/13/02
Make / Attach Empennage Spars	2/12/02	2/13/02	2/15/02
Monokote and Finish Empennage	2/14/02	2/16/02	2/17/02

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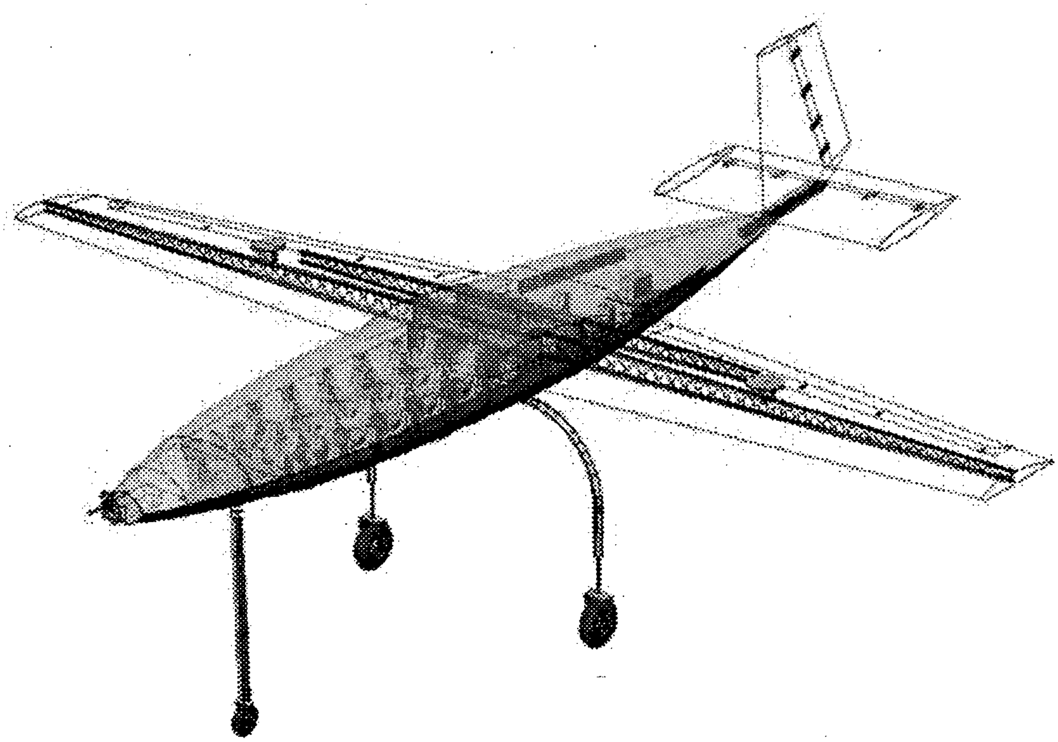
2001/ 2002 AIAA Foundation
Cessna / ONR Student Design Built Fly Competition

Design Report



"Rüzgar Sultan"
Middle East Technical University
March 2002

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1. EXECUTIVE SUMMARY

A student group of Middle East Technical University gathered to compete in 2002 Cessna/ONR Student Design Build Fly (DBF) competition. The competition involves designing and building a propeller driven, electric powered, remote controlled, unmanned aircraft that must carry softballs (10-24 softballs) around a predetermined course for three times – two empty and one loaded – in a minimum flight time. The designed aircraft must be capable of completing at least three missions.

The main goal of the design team was to design and build a performant aircraft to achieve the highest possible score during the competition.

This year's competition rules restrict each team to fly three missions, in sequence, during a single flight period that will be the lesser of 10 minutes or the time required to complete all three-mission tasks. The competition score is mainly based on the design team's written report score, rated aircraft cost and total flight score.

In this design project, engineering approaches are used to reach the final design configuration. The design cycle is composed of the following phases; conceptual design, preliminary design and final design.

1.1 Conceptual Design

To refrain from any time loss or rushing in the design process the design team is divided into five major sub-groups: Aerodynamic, Structure, Propulsion, Landing Gear, and Stability& Performance sub-groups. Each sub–group was assigned tasks to study their related concepts that are compatible with different configurations of other sub–groups. The figures of merit (FOM) were chosen and each candidate configuration was rated according to these figures of merit. The rated aircraft cost of each configuration is also considered. The aim of the conceptual design was to decide on the configurations that would bring success in the competition for analyzing further.

1.1.1 Alternative Design Parameters: During the conceptual design phase the alternatives were developed by changing the fuselage shape and configuration, wing configuration and the tail structure. A trade – off study has been performed for the tail structure among the conventional tail, T – tail, V – tail, and canard configuration. The advantages and the drawbacks of the wing planform being rectangular, elliptic or tapered has been considered. The mono-wing and bi-wing configurations were examined in the conceptual design phase. The fuselage structural configurations that were taken into consideration were the symmetrical and the airfoil imposed fuselage shapes to most efficient benefit from the internal volume of the fuselage. Since the motors to be used are restricted by the competition rules, the most efficient ones from the variety of Astro Flight and Graupner brushed motors were selected for further consideration without neglecting the motor can also be used in a canard configuration.

1.1.2 Results of the Conceptual Design: Different combinations of the designed aircraft having different wing planform, fuselage structural shape, tail configuration and manufacturing material were examined throughout the conceptual design phase. Whole election process has been based on the idea of carrying a maximum number of softballs possible (24 softballs). Four conceptual configurations were selected among the others for further analysis. All four alternative designs had the following characteristics in common: tapered wing planform, monoplane wing configuration, wing structure made-up of foam core and composite sheet covering, fuselage constructed from carbon fiber composite, tail manufactured from foam core and wood sheet covering, tricycle and faired landing gear configuration. The differences raised in the winglet, tail configuration and fuselage structure concepts for the four alternative designs.

1.2 Preliminary Design

The four alternative design concepts were examined in detail during the preliminary design phase. More intensive calculations were made on the configurations to carry the selection process on a more scientific platform. Each configuration was also studied in the name of aerodynamics to obtain the highest efficiency. The aim of the preliminary design phase was to decide on the optimum configuration for the designed aircraft to be considered in detailed design phase.

1.2.2 Results of the Preliminary Design: The candidates for the optimum configuration has been examined in detail and the one that owned the most advantageous characteristics had been chosen. The optimum configuration had been studied more deeply and engineering calculations were made on its performance. The results that were obtained by the end of the calculations of the preliminary design phase are summarized.

Wing Span	2.65 m (8.69 ft)
Wing Area	1.1 m^2 (11.829 ft^2)
Aspect Ratio	6
Taper Ratio	0.45
Wing Airfoil	SG 6042
Horizontal Tail Airfoil	64A0010
Vertical Tail Airfoil	64A0010
Tail Configuration	Conventional
Fuselage Length	2.706 m (8.862 ft)
Max Payload Capacity	4.464 kg (9.8 lb)
Empty Weight	10.5 kg (33 lb)
Horizontal Tail	
Span	0.650 m (2.12 ft)
Aspect Ratio	3.255
Vertical tail	
Span	0.310 m (1.015 ft)
Aspect Ratio	1.355

Power Plant Parameters	
Number of Motor(s)	1
Motor	Graupner Ultra 3450-7
Power	1160 kw (1.532 hp)
Cells	SR 2400 max
Number of Cells	36
Propeller	18x12 two-blade propeller
Landing Gear Specifications	
Configuration	Tricycle
Type	Faired

Table 1.1 Summary of the Results of the Preliminary Design

1.3 Final Design:

In the final phase of the design, the selected design configuration was improved by detailed studies and was prepared for manufacturing. The longitudinal, lateral static and dynamic stability analysis were performed and the response of the designed aircraft was calculated. Component selection and system architecture was also examined in detail. The aim of the final design phase was to investigate the stability and control characteristics of the designed aircraft, to revise the precalculated parameters by more detailed analysis and to prepare the final design configuration for construction.

1.3.1 Results of the Final Design : The end of the final design was also the end of the design that can be performed by paper work. Though the static margin of the aircraft seemed high, due to the interactions of the wing with the fuselage, the static margin is expected to reduce. The longitudinal stability behavior of loaded aircraft came out to be perfect. However for the unloaded case, the damping was not very good but this was considered to be critical since the careful respond of the pilot can come over this problem. The results of the lateral stability analysis were also satisfactory and the designed aircraft is expected to show a good lateral stability. By the end of the stability calculations, it was seen that the designed aircraft was capable of handling very harsh conditions of Wichita. The Dutch Roll characteristics of the designed aircraft were also good such that the aircraft was able to recover from Dutch Roll condition. All the calculations were based on the idea of carrying 24 balls. The stability and weight analysis has showed that the designed aircraft is capable of completing the mission safely with 24 balls. The total time required to accomplish the mission was reduced to 4:04 min. For the final airplane, if a written report score of **80** were taken, with the rated aircraft cost of **13.9013**, total score was predicted as **11306.47**. However this value would not mean anything, since the score formula is revised.

2. MANAGEMENT SUMMARY

2.1 Introduction

By the start of 2001 fall–semester, a group of Middle East Technical University students gathered together to participate and compete in this year's Cessna/ONR Student Design/Built/Fly competition. Our team worked on this project in a well-organized manner for the purpose of designing a successful aircraft in the light of past years' experience. The management architecture is constructed carefully in the beginning of design. The team consisted of eighteen students; twelve seniors and six juniors from aerospace engineering, and two aerospace engineering graduate advisors.

2.2. Architecture of the Team

It is known that the aircraft "design" is a long and tedious process and the structure of the team influences its outputs to a great extend. Therefore our team is composed of members with great enthusiasm to work together in a team. A team leader was elected to coordinate the tasks of the team.

The team is divided into three groups; namely, the design group, production and logistics group and sponsorship & treasurer group. Each group has a group leader to organize the group's tasks and workload among its members. The organizational structure of the team is shown in Table 2.1. The design group is divided into 5 main sub-groups, which are; aerodynamics, structure, propulsion, landing gear, stability & dynamics sub-groups. It is also decided that the integration between the groups to go in concurrence. The design group was responsible from the conceptual, preliminary and final design of the airplane. The production & logistic group was responsible from supplying all kinds of support to the design group or sub-groups, such as written documents, computer software, web searches and material. Also, the communication with the junior students is not forgotten for them to gain experience for their future studies. The sponsorship & treasurer group was responsible from finding financial support to the team to cover the manufacturing costs of the designed airplane, the travel costs for the team members and shipping cost of the UAV to the competition site.

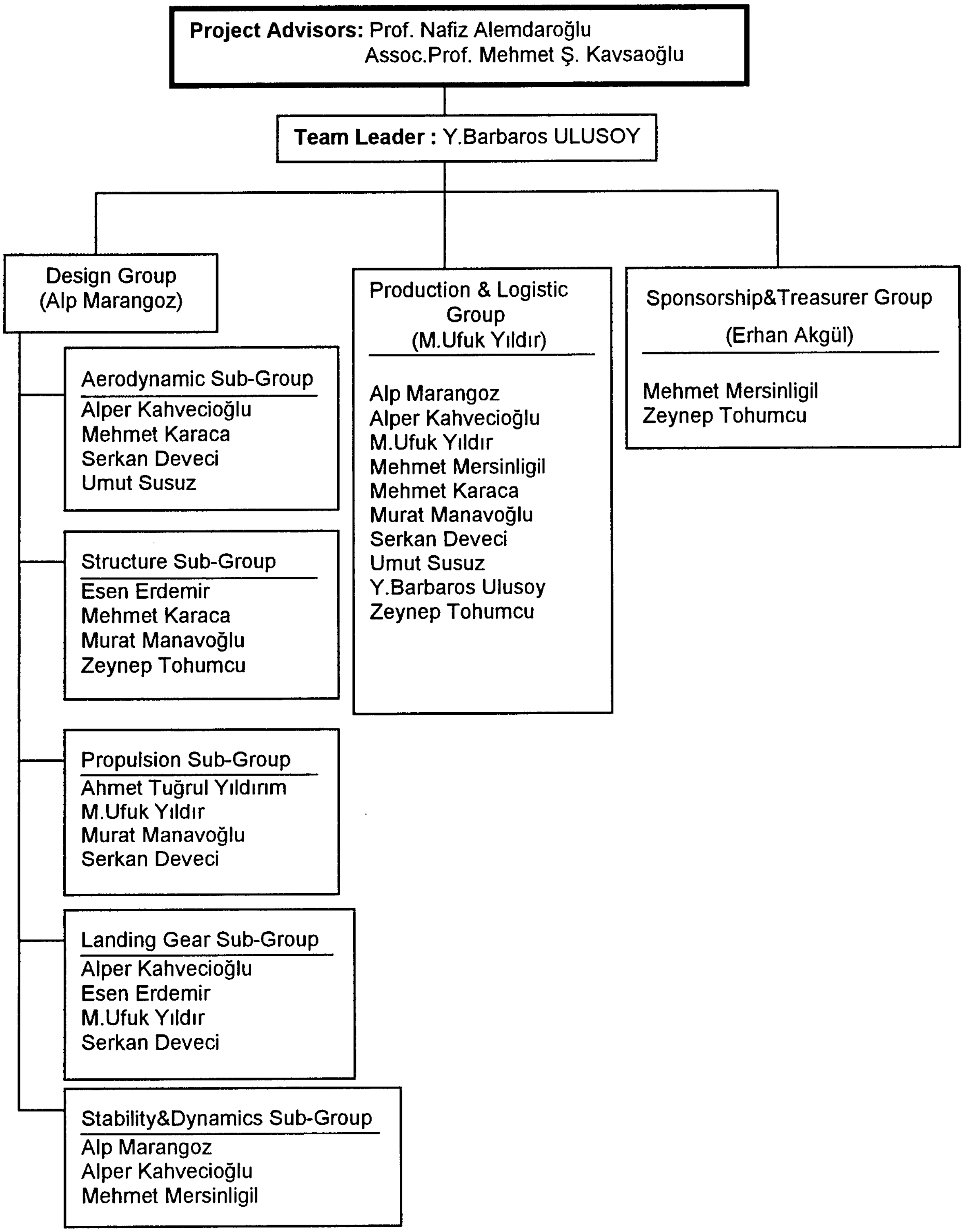


Table 2.1 Architecture of the team

The design group was composed of senior students taking the AE 451 Aeronautical Engineering Design course. This gave the team the advantage of further analyzing and modifying their design according to the knowledge acquired in this design course. The design group modified the design during their weekly study hours of the course under the supervision of the advisor.

The team leader had the responsibility of assuring that the milestones are met and controlling the whole team according to the planned timetable. In some phases of design, sub-groups worked together to prevent any delay in the planned time sequence.

During the regularly held weekly meetings of the team, the team leader assigned tasks to the production & logistic group leader following the request coming from the design group and the sponsorship & treasurer group. So the production & logistic group leader was the person who was responsible for any incomplete job. The production & logistic group leader assigned the given job to one of his group members but he personally followed the task until the job task was performed. In case the job was not completed, the same job was assigned to another member of the group who was also capable of performing the same job on the assigned time. So, there was an auto-control mechanism within the system. In addition, the team leader was personally responsible for the project advisors for any incomplete job. The project team held meetings with the advisors each week. In these meetings, the team groups notified the present and subsequent milestones of the group. Besides, the problems of the groups and the status of the project were discussed in the mood of a forum. Also, the project time chart was checked. The weekly team meetings, in which each group leader gave a report to the team leader and the weekly study hours, in which the team leader gave a report to the project advisor, became a successful method of controlling the progress of the project. The reason for the delayed jobs was mostly due to the lack of material.

The Table 2.2 Milestones Chart shows the planned and the actual timing for the major and minor components of the aircraft.

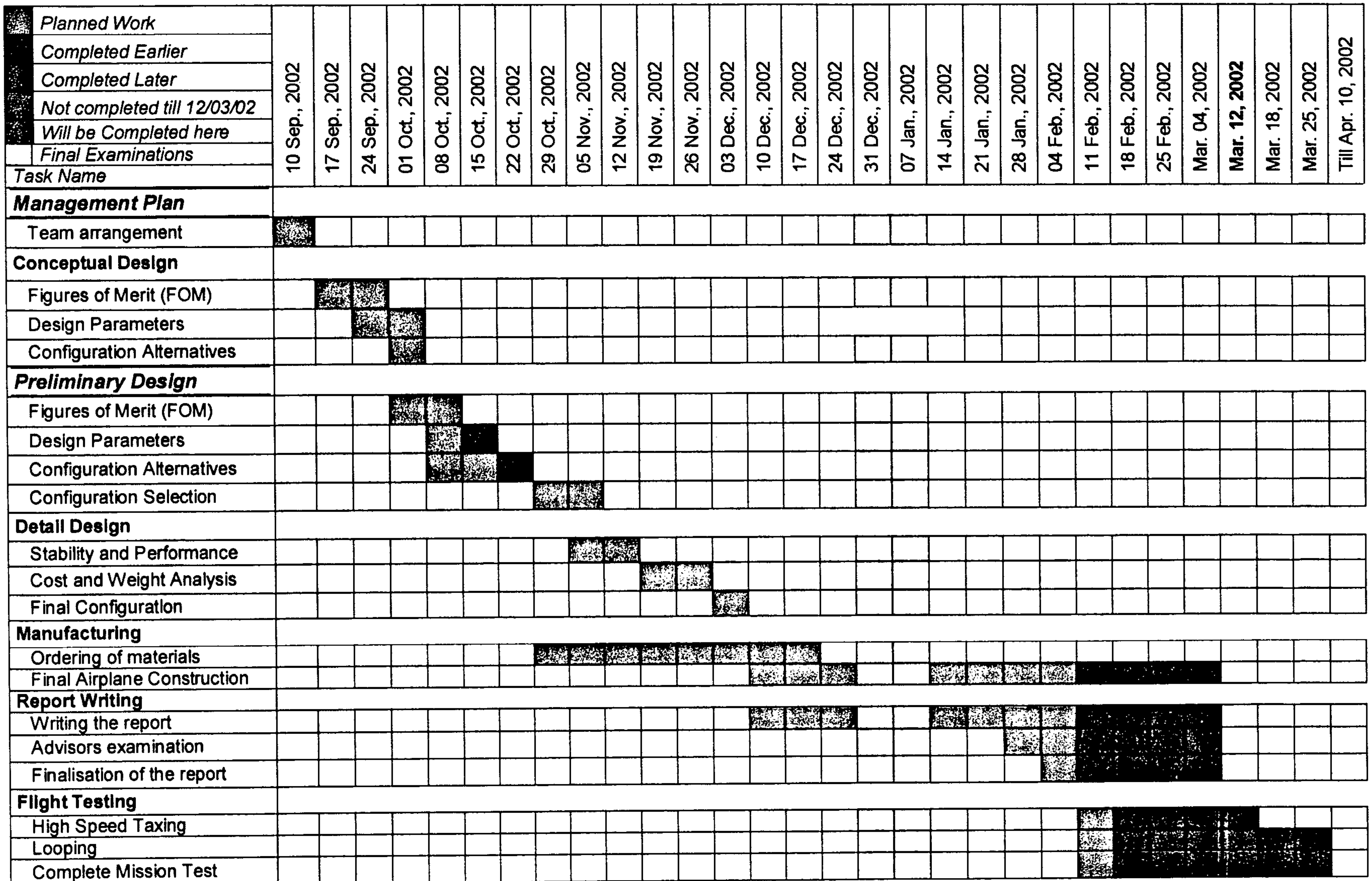


Table 2.2 Milestones chart

The success of the team depended not only on the successful organizational structure of the team but also on the team spirit to reach the final target. The participation and involvement of each team member is followed throughout the project. To show the involvement of each group member to the related tasks, a rating over 10 is given for each member as shown in Table 2.3. In this figure, "10" shows "maximum involvement" and "0" shows "no involvement".

	Alp Marangoz	Alper Kahvecioğlu	Erhan Akgül	Esen Erdemir	M.Ufuk Yildır	Mehmet Mersinligil	Mehmet Karaca	Murat Manavoğlu	Serkan Deveci	Umut Susuz	Y.Barbaros Ulusoy	Zeynep Tohumcu	Underclassmen
Conceptual Design													
Design Parameters	10	10	2	10	10	10	9	10	10	8	10	8	3
Figures of Merit	10	10	0	10	10	10	8	10	10	9	10	7	3
Fuselage Analysis	10	10	0	9	10	10	9	8	10	9	9	7	3
Wing Analysis	10	10	1	10	10	9	8	9	9	7	10	8	3
Empennage Analysis	10	10	0	10	10	9	7	9	9	8	9	7	3
Power Plant Analysis	10	10	0	8	10	10	10	9	9	9	10	4	3
Results obtained	10	10	3	10	10	10	10	10	10	10	10	4	3
Preliminary Design													
Fuselage Analysis	10	10	0	10	10	9	9	9	9	7	10	8	3
Wing Analysis	10	10	0	8	10	10	8	9	9	9	9	8	0
Empennage Analysis	10	10	0	9	10	10	8	9	9	9	9	8	3
Power Plant Analysis	10	10	0	9	10	9	8	9	9	7	10	8	0
Landing Gear Analysis	9	10	0	9	10	9	7	9	10	8	10	8	3
Results Obtained	10	10	0	10	10	10	10	10	10	10	10	10	2
Detail Design													
Stability and Control Analysis	10	10	0	9	10	9	8	9	8	7	9	10	3
Performance Data Acquisition	10	10	0	9	10	10	8	9	9	9	9	10	0
Comp. Selection & Sys. Arch.	10	10	0	8	10	9	10	8	9	8	9	8	0
Weight Calculation	10	10	0	9	10	10	9	9	10	10	10	9	2
Manufacturing													
Design Parameters	10	10	0	0	10	10	9	10	10	7	10	4	0
Figures of Merits	10	10	0	0	10	10	7	10	9	5	10	3	0
Results obtained	10	10	0	0	10	10	8	10	10	6	10	3	0
Manufacturing of the airplane	10	10	0	0	10	10	8	10	10	5	10	3	0
Documentation of Design	8	7	2	7	10	8	7	8	8	5	9	10	0
Final Report	7	4	0	10	4	9	7	9	10	3	10	0	0
Drafting Package	0	0	0	0	10	0	0	0	0	0	0	0	0

Table 2.3. Personnel Assignment: Showing the participation of the team members to the project. A "10" shows the maximum participation of the member and a "0" shows no participation.

The group members and their functions are summarized below in Table 2.4

MEMBER	FUNCTION AT THE TEAM
Y.Barbaros Ulusoy /AE/SR	Team leader. Organized the communication between the sub-teams. Made contact with AIAA. Contributed to the design of the aircraft. Participated in production and written report.
Alp Marangoz /AE/SR	Design group leader. Dynamics & Control sub-group member. Contributed to production. Participated in the written report.
M.Ufuk Yıldır /AE/SR	Production and Logistics group leader. Landing Gear and Propulsion sub-group member. Contributed to the design and made the drawings of the aircraft. Participated in manufacturing
Alper Kahvecioğlu /AE/SR	Aerodynamics, Landing Gear and Dynamics & Control sub-group member. Contributed to production and written report.
Mehmet Mersinligil /AE/SR	UAV Pilot. Propulsion and Dynamics & Control sub-group member. Contributed to production and written report. Sponsorship team member.
Serkan Deveci /AE/SR	Aerodynamics, Propulsion and Landing Gear sub-group member. Contributed to production. Participated in the written report.
Esen Erdemir/AE/SR	Structure and Landing Gear sub-group member. Contributed to written report.
Mehmet Karaca /AE/SR	Aerodynamics and Structure sub-group member. Participated in production and written report.
Murat Manavoğlu /AE/SR	Propulsion and Structure sub-group member. Contributed to production and written report.
Umut Susuz /AE/SR	Aerodynamics sub-group member. Participated in production
Zeynep Tohumcu /AE/SR	Structure sub-group member. Contributed to production. Sponsorship group member.
Erhan Akgül /AE/SR	Treasurer. Sponsorship group leader.
Ahmet T. Yıldırım /AE/SR	Propulsion sub-group member.

Table 2.4 Function of member in the team

3. CONCEPTUAL DESIGN

3.1 Introduction

The conceptual design phase includes the examination of the alternative configuration concepts and the selection of the most suitable configuration for the competition. Detailed analysis of each parameter affecting the design was performed, and the importance of each parameter was examined. At the end, the best configuration was chosen for the components of the airplane. In the succeeding sections, the figures of merit and the result of the analyses are supplied.

3.2 Figures of Merit (FOM)

Although there are many design parameters to consider, the figures of merit are introduced to reduce the categories to which more importance should be given, and to reduce the number of possible configurations for further analysis. The departure point is to consider what is expected from a UAV and what the DBF/2002 rules dictated us. Evaluation method of the competition is analysed to obtain the UAV that would get the highest score. This lead us to concentrate on a design that can carry the highest number of softballs within the shortest possible time period including the loading/unloading times with all the assigned laps flown and having the lowest possible rated aircraft cost. Since the figures of merit directly affect the optimum design of a UAV, an extra care was spent for their determination.

3.2.1 Manufacturability

One of the most restricting parameter of the design phase is the technology required to construct the UAV. Most of the team members were not experienced in building a model aircraft and had a lack of knowledge about constructing balsa or composite structures. Simple design would yield a shorter time for manufacturing. As the structure became more complex, the required manufacturing time increased. Since none of the team members were professional model makers, it would be a difficult task to build complex and complicated proposals. By each year, every team gains experience and therefore to be one of the bests, we needed to do a better job than last year's team and be in a step further. To do this, we had to search for better manufacturing technologies that would give us the chance of constructing an aircraft that would probably have a complex and complicated structure. Also, it can be mentioned that the manufacturability gives us the ability of repairing or producing the airplane in case of any damage.

3.2.2 Power Consumption

Since the rules restricted the weight of the batteries to be used, power consumption was an important consideration for the design team. On the other hand, the weight of the batteries used has an extra importance from rated aircraft cost point of view. Thus the design team's aim was to minimize the mission flight time while maximizing the number of balls to be carried (maximum of 24 balls). Since our aim is to complete the mission given in a complete and successful way, we have chosen to design of the

airplane for the maximum number of balls. As the weight of the aircraft increased, the power requirement for takeoff in the takeoff, distance given by the rules, increased. Therefore the weight of the UAV was an important factor to consider when determining the power consumption of the UAV. Also, it is known that as the aerodynamic efficiency of the aircraft increases, the drag acting on the aircraft reduces, decreasing the power required for the aircraft. For that reason, aerodynamic efficiency was considered as a selection parameter for the UAV.

3.2.3 Specific Strength

The structural strength of the aircraft is affected by many factors. The wings must be able to carry the entire weight of the aircraft, the fuselage must be strong enough to carry the payload, the landing gears must withstand high impacts, etc. All of these factors are considered in determining the materials and manufacturing techniques to be used. When performing the selection of the materials, their weight was considered as well to be a major factor, since the aim is to have a light aircraft.

3.2.4 Handling & Performance

The handling qualities and the performance of the parameters of the aircraft are of the most important factors considered for the design. The UAV must have good handling qualities and it should be controllable from the ground. It must also have short takeoff and landing distances as specified by the rules of the competition. Since the takeoff distance and the battery weight are restricted (200 ft and 5 lb), the designed aircraft should be a high performance one to satisfy these requirements. The aerodynamic performance of the aircraft must be optimized in order to carry the maximum number of softballs allowed with minimum power requirement. In addition, the aircraft must be easily controlled during its entire flight envelope either empty or loaded, and must withstand adverse atmospheric conditions, i.e. strong crosswinds.

3.2.5 Cost

Each team participating in the competition is responsible to fund the total cost of their UAV. "Cost" is an important figure of merit and must be minimized as much as possible. Therefore a trade-off analysis is required in making the choice of construction materials, as far as their cost and advantages are considered.

3.2.6 Stability

The stability of the aircraft has a direct effect on the aircraft performance and handling qualities. Thus designing a stable aircraft capable of maneuvering easily was the goal of the team. The UAV should easily be controlled while on the ground and in the flight, either empty or loaded along a predetermined course and under adverse wind conditions.

3.2.7 Transportation

There is one additional restriction for our team to consider in design; the restriction imposed for the transportation of the finished product. Since the aircraft will be transported overseas in a large box (container), the maximum sizes imposed by the airline transportation must be obeyed. Therefore, the aircraft needs to be easily assembled and disassembled and should satisfy the volume and size limitations set by the airlines.

3.3 Initial Phase Design Parameters

The rough estimates for the initial size of the aircraft, its components, the materials to be used, the wing and the tail geometries, the structure of the fuselage to carry the desired payload, the engine(s) and the landing gear to carry the total weight of the UAV, all considered through this section. The following design parameters are investigated for the initial design phase of the conceptual design.

3.3.1 Wing Planform

Although the wing planform does not have a direct effect on the weight and the rated aircraft cost of the airplane, it affects the performance of the aircraft.

There are not many options feasible for the wing planform of the designed aircraft. The options considered for the wing geometry were rectangular, elliptical and tapered planforms. Each of these options had their advantages and drawbacks. The elliptical planform is the one that has the best lift distribution and the minimum induced drag but it is very difficult to manufacture an elliptical wing and is also time consuming. Although the rectangular wing planform is more superior to other planforms from the manufacturability point of view, the rectangular wing geometry has shortages in aerodynamic efficiency. Therefore a tapered wing is considered. It is known that for a certain taper ratio and quarter chord sweep angle; a tapered wing can produce a lift distribution similar to that of an elliptic wing. The taper therefore shortens the required manufacturing time and provides a satisfactory lift distribution.

3.3.2 Wing Configuration

Wing configuration is another parameter that affects the performance of the airplane. The design team examined two different wing configurations, monoplane and bi-plane, for this mission with and without winglets.

When the winglets were considered, it was realized that they added to the rated aircraft cost with marginal improvement gained in the aircraft performance.

Bi-plane wing configuration should be considered when low structural weight is more important for the design than the aerodynamic efficiency or when low speed is required without complicated high lift devices or excessive wingspan. There is a reduction of 30% in induced drag for a bi-plane when compared to a monoplane of equal span. However, due to the interaction between the two wings, the lift per unit wing area is not as high for a biplane as it is for a monoplane having the same aspect ratio.

Moreover, a bi-plane configuration suffers from the rated aircraft cost a lot. Therefore a carefully manufactured monoplane configuration will provide the same lift force with less structural weight.

3.3.3 Wing Structure and Materials

Three types of construction were considered for the wing. These were namely; from wooden ribs, using a foam core with balsa wood sheet covering and using a foam core with composite sheet covering. The wooden rib construction is light and stiff, however, difficult to realize since manufacturing of it requires even more precision for a tapered wing. The manufacturability of the balsa wood sheet covering is easier than the composite sheet covering, but since the composite materials had a better specific strength, its use can be more justified. Although composite materials are more expensive and hard to manufacture, because of its improved structural strength advantage over the wood structure, composite manufacturing is chosen for further analysis. The design team decided that the facilities available were sufficient for construction and manufacturing of the composite structure.

3.3.4 Fuselage Structure, Shape and Material

Before designing the structure of the fuselage the design team has considered the evaluation criterion of the DBF contest. To get the highest score from a mission it is needed to carry the highest number of softballs within the minimum time with all the laps flown during a mission according to the DBF2002 competition rules. Therefore, it was decided to design the aircraft to carry the maximum number of softballs, which is 24. It was thought that if a problem was encountered, the number of softballs to be carried could always be reduced. Keeping in mind, the adverse and windy conditions of the competition field, designing for maximum number of balls and then reducing it seemed to be a good choice.

The fuselage is a very important part of the aircraft. It must possess the structural strength to withstand the forces induced on the assembly by the wings, tail structure, landing gear, payload and motor.

There are restrictions imposed by the rules of the competition about the layout of the balls. No payload might be carried in the wing unless a configuration of flying wing is used and no speed-loader or rapid-insertion container is allowed.

The mission score is very much affected by the mission time. The shorter the mission time is the higher the score. Since the mission time includes the loading and unloading of the cargo, we need to shorten the loading/unloading time as well. This can only be achieved by using an easily accessible cargo door.

For the efficiency of the fuselage it is important to have the smallest possible volume. This will provide a lighter fuselage. Small and aerodynamic fuselage must be considered since it reduces the drag considerably when the strong side winds are taken into account.

A rectangular fuselage can be manufactured very easily with reasonable cost; however, it produces high drag. Also a rectangular fuselage will have a high amount of wasted volume therefore the volume of the fuselage would not be used efficiently.

For decreasing the drag of the fuselage, an airfoil fitted body would have reasonably less drag and therefore the power consumption can be reduced significantly by reducing the drag. However, there may be problems during the manufacturing of such an airfoil fitted fuselage because of its complexity. This would increase the cost of manufacturing as well.

For manufacturing purposes, it would be advantageous to separate the fuselage into two axisymmetrical parts – an axisymmetrical body would be preferable due to its manufacturability.

The material selected for the fuselage is also very important especially from manufacturability, cost and specific strength, (strength per weight) points of view. We consider mostly the wood and carbon fiber composite for the production of the fuselage. Balsa wood is a light material that can be used but when the specific strength is considered, carbon fiber composite is much better. However the composite material would cost more when compared with the balsa wood. Also when ease of manufacture is considered, composite fuselage would be difficult in the lack of advanced tools. If good tooling capabilities are achieved, composite body production would even be easier than the wood body production.

3.3.5 Tail Structure

A tail consists of a horizontal and a vertical stabilizer located aft of the center of gravity by means of a tail boom (for canard configuration it is forward of the center of gravity) and is used to control and stabilize the aircraft in pitch and yaw.

For a conventional tail, the horizontal and vertical stabilizers are mounted to the tail boom structure with the horizontal stabilizer located at the bottom of vertical stabilizer. This classical configuration is very easy to manufacture, to analyze, and has been well researched and proven in most of the successful aircraft designs. Therefore it should be considered further.

For the T-tail configuration, the vertical stabilizer itself is joined to the tail boom and the horizontal stabilizer is mounted on top of it. Although T–tail has many drawbacks, its spin recovery characteristics are good due to the fact that the flow coming out of the horizontal tail surface does not intersect with the vertical tail surfaces. There is always uniform fresh airflow coming to the vertical tail surfaces since they are situated at the tips of the horizontal tail. These advantages are accompanied with an increase in the structural weight and a stall recovery problem. Also the manufacture of a T-tail is more cumbersome as the elevator control rods must be transferred through the vertical stabilizer.

A V-tail is something strange as there is no such difference between the horizontal and vertical stabilizers. It uses two surfaces aligned in a V-shape. V-tail provides extra stability advantages. However, the integration and the landing/takeoff criteria make this configuration difficult. Yet, it is decided to have further analysis for this configuration because of its advantages in reducing the material amount to be used to manufacture this configuration.

A canard configuration places the horizontal stabilizer forward of the wing. Because a canard surface generally produces lift, an aerodynamically efficient configuration may be obtained. Also the position of the motor (if single motor is used) can be changed to have a pusher aircraft. A canard configuration is generally shorter than a conventional airplane, thus reduces the rated aircraft cost. However, the static and dynamic stability characteristics and handling qualities need to be well understood by ourselves and we need to gain experience on this type of aircraft. A canard configuration has many advantages and it is worth to think on it.

A flying wing may be the best of all; however it is very difficult to manufacture a stable aircraft so the design team decided not to study it further.

3.3.6 Landing Gear

The landing gear is the most critical part of the aircraft since they are subjected to high loads. Therefore the aircraft and its parts like propeller(s) must be safe even when the aircraft makes unsuccessful landings.

The landing gear configuration has a direct effect on the ground handling characteristics of the aircraft. Therefore the landing gear configuration became an important issue as the team realized that multiple takeoff and landings are required during each mission segment. In order to satisfy the design mission, an efficient landing gear configuration needs to be designed. The main concern was made on the "tricycle" and "tail dragger" arrangements together with a "retractable" or "faired" configuration. They are discussed in detail below.

A retractable gear eliminates the drag coming from the landing gears while the airplane is in flight. However, multiple takeoffs and landings require greater power consumption to raise and lower the landing gear for takeoff and landing. Also, it is more difficult to manufacture a retractable landing gear that would be strong enough to support the weight of the aircraft. Allocating space for the retractable gear is another problem to be considered. A retractable gear would be heavier than a faired one.

A tail dragger design was considered because of the decreased drag on the landing gear during flight. A tail dragger design provides the necessary strength to support the airplane on landing and takeoff. However, tail dragger designs are more difficult to control on the ground.

The third landing gear design option considered was a standard tricycle landing gear. The tricycle landing gear can easily be manufactured to support a heavy airplane. This landing gear provides satisfactory ground control but cause an increase in the drag. The tricycle landing gear was selected for further analysis because it provided the desired ground control and the structural strength.

3.3.7 Power Plant

The power plant consists of the electrically powered motor(s), propeller(s) and the batteries. There are some requirements and restrictions given in DBF/2002 rules to be considered in the evaluation of the motor and the batteries.

The motor selection should be made after deciding on the number of motors to be used. The first idea was to use two motors to generate more thrust and lift. However, the power available is restricted by the 5 lb battery weight. From the past years' experience it is known that, if two motors are used, the batteries (number of the batteries corresponding to the maximum allowable battery weight) can feed the motors only about 5-6 minutes. This duration may not be enough for this year's competition since the mission is defined as approximately a 10-minute mission. Therefore, it is more important to have the power allowing the aircraft to complete a whole mission than to have high thrust. This led us to concentrate on the single engine alternative. If a good design were made, one engine would be sufficient to fly the whole mission. Using single engine will also be beneficial as the rated aircraft cost is taken into account, since an extra engine worth 375 $ times the total battery weight. In terms of the weight considerations, each extra engine costs about 800 grams including the mounts and connections therefore using a single motor is the leading option for further studies. Another important thing is the real cost. We would not pay for an extra motor if we choose to use single engine. Keeping all these in mind, it was decided that the design should be made by using single motor.

The most important thing to be considered in the selection of the motor is the power rating of it. Since we are designing the aircraft with only one engine, we must make our choice very carefully. However the most suitable engine can only be selected when the power supply and the propeller size are known.

NiCad batteries are a requirement of the competition, so batteries with high capacity are subjected to use. Another consideration about the cells is their weight. The total weight of the cells to power the motor, as mentioned before, should be less than 5 lb, which constraints the variety of cells that can be used.

Propellers may be of wood or reinforced nylon, but in the conceptual design, the size of the propeller can be selected as the optimum propeller that the selected motor can run. Although the wooden propellers make more noise than the nylon propellers, their efficiencies are higher than nylons. As there is no such limitation for noise, to use the more efficient wooden propellers is reasonable. The size of the propeller is selected as a function of the needed thrust.

A variety of motor, battery and propeller configurations are inspected. During the inspections we have tried to benefit from the software. "MotoCalc" (Ref 3) is one of the softwares that we have used frequently in order to get the best configuration that would be suitable for our design.

3.3.8 Spar Material and Structure

Different types of spar materials were taken into consideration. From comparatively light materials such as balsa, hardwood, fiberglass and polycarbonate to heavier materials such as aluminum and carbon fiber composites were investigated. The purpose was to determine a spar material with the largest specific strength, which would be sufficient for the expected loading on the wing.

Hardwood and balsa wood are widely used and easy to find but their low strength to weight ratio and difficulties in manufacturing complex geometries are disadvantages of these materials. Also the strength characteristics are not uniform within these materials.

Aluminum has a high strength; however, it is a very dense material compared to the other materials considered. Besides, the manufacturability of aluminum is better than carbon fiber composite material. It is also a cheaper material.

A spar from carbon fiber composite is not easy to manufacture and is rather costly when compared to other alternatives. Although it has a higher specific strength than other spar materials described above due to its manufacturing difficulties this alternative is also eliminated.

The spar provides the primary structural strength of the wing. It is designed to withstand a variety of loads, which the wing will experience during flight. The two parameters that affected the initial design of the spar were its weight and its length. The main variables that governed these parameters were the cross-sectional geometry of the spar and its material characteristics.

The geometry considered for the beams' cross section included an I-beam, C-beam, solid rectangular, circular cross-section and box beams. An I-beam has the greatest moment of inertia and therefore the greatest strength about its horizontal axis; however, the strength about the vertical axis is small. An I-beam would also be very difficult to manufacture using lightweight materials.

A circular cross-section has equal moments of inertia about both the horizontal and the vertical axes providing equal strength in all directions. However, this design makes the spar unnecessarily heavy and it would be cumbersome to join it with other parts, for example with the wings, of the aircraft.

A solid rectangular beam is the strongest, but like the circular beam, much of the material towards the center of the beam is relatively unstressed and therefore unnecessary. The solid rectangular beam is simple to manufacture, yet it has a low strength to weight ratio.

The box beam has a larger strength about the horizontal axis but also has a higher weight than an I-beam, which is not desired. The box beam is hollow, decreasing the amount of wasted material, which helps reducing the weight of the spar. However, it would still be hard to produce this type of spar for the wing considered.

A C-beam has a large moment of inertia but less than an I-beam and a large strength about its horizontal axis; on the other hand, the strength about the vertical axis is small as an I-beam. A C-beam would also be easy to manufacture using lightweight materials. Also C-beam provides us a weight advantage.

3.3.9 Control Surfaces

Primary control surfaces are ailerons, flaps, elevator and rudder. The sizing of these surfaces is very important as maneuverability and stability is affected by their sizes. Before considering the areas of the control surfaces, a decision between flaperon and aileron+flap combination must be made. Based on our previous experience, it was decided to use a flaperon.

In order to have enough maneuvering capability, depending on very rough calculations, the flaperons should be about %25 of the wing chord and about %80 of the wingspan, as high amount of lift might be necessary for takeoff since the take off distance is limited to 200-ft.

For rudder and elevators, control surface areas depend on the selected tail configuration. If a more effective configuration is selected to manufacture, the control surface areas can be reduced or more analytic values can be obtained.

3.4 Configuration Selection

The design parameters were examined based on the figures of merit (FOM) and the detailed examination on these parameters are represented in Table 3.1. The design parameters were graded as -1, 0 and 1. A grade of -1 indicates that this parameter can cause a disadvantage, while a grade of 1 indicates that the parameter is an advantageous one. If the parameter does not have neither advantages nor disadvantages, that is neutral, a grade of 0 is assigned. Then the grade of each item is multiplied by the weighting factor of each figure of merit and summed up to get the total score of that design parameter. The ones with the greatest total score were selected for further analysis. The "FA" abbriviation in the decision column stands for further analysis. "E" stands for eliminated in the decision column. A N/A was assigned for the cases that the figure of merit ment to be not applicable for that parameter. The configurations marked bold are the ones that are selected for further analysis.

	Design Parameters	Manufacturability	Power Consumption	Specific Strength	Handling&Perf.	Cost	Stability	Total	Decision
	Weighting Factor	1.5	1.5	2	1	1	1		
Wing Planform	Rectangular	1	-1	NA	-1	1	-1	-1	E
	Elliptical	-1	1	NA	0	-1	0	-1	E
	Tapered	**0**	**0**	**NA**	**0**	**1**	**1**	**2**	**FA**
Wing Configuration	**Monoplane with winglets**	**1**	**0**	**NA**	**1**	**0**	**1**	**3.5**	**FA**
	Monoplane w/o winglets	**1**	**1**	**NA**	**1**	**1**	**1**	**6**	**FA**
	Biplane with winglets	-1	1	NA	0	-1	0	-1	E
	Biplane w/o winglets	0	0	NA	0	-1	-1	-2	E
Wing Structure	Wooden	-1	-1	-1	NA	0	NA	-5	E
	Foam core & Wood Sheeting	**1**	**1**	**0**	**NA**	**1**	**NA**	**4**	**FA**
	Foam core & Composite Sheeting	**0**	**0**	**1**	**NA**	**0**	**NA**	**2**	**FA**
Fuselage Structure	Rectangular	1	-1	NA	-1	1	-1	-1	E
	Symmetry	**0**	**1**	**NA**	**0**	**0**	**1**	**2.5**	**FA**
	Airfoil	**0**	**1**	**NA**	**1**	**-1**	**1**	**2.5**	**FA**
Fuselage Material	Balsa Wood	1	0	-1	NA	1	NA	0.5	E
	Carbon Fiber Composite	**0**	**1**	**1**	**NA**	**-1**	**NA**	**2.5**	**FA**
Tail Structure	**Conventional**	**1**	**1**	**1**	**1**	**1**	**0**	**5**	**FA**
	T-tail	-1	0	-1	1	-1	1	-2.5	E
	V-tail	0	1	0	1	-1	1	2.5	FA
	Canard	**1**	**1**	**1**	**0**	**0**	**0**	**4**	**FA**
Tail Material	**Balsa Wood + Foam**	**1**	**1**	**0**	**NA**	**1**	**NA**	**4**	**FA**
	Carbon Fiber Composite	**0**	**0**	**1**	**NA**	**-1**	**NA**	**1**	**FA**
Landing Gear	**Tricycle**	**0**	**0**	**NA**	**1**	**1**	**NA**	**2**	**FA**
	Tail dragger	0	0	NA	0	1	NA	1	E
	Retractable	-1	-1	NA	0	-1	NA	-4	E
Spar Structure	I-beam	-1	NA	1	NA	1	NA	1.5	E
	C-beam	**1**	**NA**	**1**	**NA**	**1**	**NA**	**4.5**	**FA**
	Solid Rectangular	0	NA	0	NA	-1	NA	-1	E
	Circular Cross Section	0	NA	0	NA	-1	NA	-1	E
	Box beam	0	NA	0	NA	1	NA	1	E
Spar Material	Wood	1	NA	-1	NA	1	NA	0.5	E
	Fiberglass	0	NA	0	NA	0	NA	0	E
	Polycarbonate	0	NA	0	NA	0	NA	0	E
	Aluminum	**1**	**NA**	**0**	**NA**	**1**	**NA**	**2.5**	**FA**

Table 3.1 Alternative design concepts

3.5 Alternative Configuration Concepts

By examining Table 3.1 it can be seen that some of the design parameters were eliminated due to their low score compared to their competitors. This elimination could be made among more alternatives. However, because of the very limited time allocated for conceptual design, the team had to consider the tradeoff between the numbers of alternative design parameters.

Depending on the chosen design parameters to which an "FA" is assigned, the team decided to create alternative configuration concepts. For these design parameters, the team created 32 alternative concepts. These concepts are tabulated in Table 3.2. "Y" indicates that the configuration had that property and "N" indicates it did not have. However, it was decided that a better final design would probably result, if fewer configurations were analyzed in greater detail. To reduce the number of the alternatives, all the concepts were considered one by one and the elimination was made depending on the results obtained as follows.

All the alternative concepts have the following properties in common; tapered mono-wing, tricycle faired landing gears, C-beam-aluminum spars and a fuselage made of carbon fiber composite.

The team decided not to use winglets in the first place because they increased the rated aircraft cost. Also, the effects of side winds might be amplified by the existence of the winglets. However, depending on the performance of the aircraft, winglets may be added later.

When the wing structure was considered, the team wanted to be on the safe side by choosing the foam core+composite sheeting because if the wing structure was composed of foam core+wood sheeting (balsa wood), a bad landing could cause the wing to break. In such a situation, the competition ends for the team because the teams are not allowed to repair the wing or wing substitution. Foam core+composite sheeting would be heavier than the balsa sheeting, however it worth for its structural strength.

The team did not really make a choice between an axisymmetrical and an airfoil fitted fuselage. Instead, if it we can manufacture, a symmetrical airfoil fitted fuselage would be the best. This would have some advantages especially in climb. Also, such a fuselage would decrease the drag produced on it and may produce some amount of extra lift. Since it was decided to manufacture the fuselage from carbon fiber composite, the manufacturing process would not be a big problem if such a mold were produced.

The tail structure became an important parameter that would affect the whole configuration of the aircraft, therefore two alternatives were left; conventional and canard. A canard aircraft would probably be shorter than a conventional aircraft and the rated aircraft cost of it would be less than a conventional one. When the lift gained from a canard is concerned, the wing area can be reduced. When all these advantages of a canard aircraft are taken into account, a canard configuration may be lighter than a conventional one. However, its stability may be a problem, although manufacturing a canard aircraft seems advantageous. The team searched for canard type UAVs that do not have automatic control systems but no such record could be reached. Therefore, a conventional tail was selected as the best alternative and the canard configuration is the second best.

From manufacturability point of view, it was decided that a balsa wood+foam tail would be easier and cheaper than a tail made of carbon fiber composite. Although we have selected both for further analysis, it is more advantageous to use balsa+foam for the tail structure.

Design Parameter		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32
		CONFIGURA TION																															
Wing Planform	Tapered	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y
Wing Configuration	Monoplane with winglets	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	N	N	N	N	N	N	N	N	N	N	N	N	N	N	N	N
	Monoplane w/o winglets	N	N	N	N	N	N	N	N	N	N	N	N	N	N	N	N	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y
Wing Structure	Foam core & Wood Sheeting	Y	Y	Y	Y	Y	Y	Y	Y	N	N	N	N	N	N	N	N	Y	Y	Y	Y	Y	Y	Y	Y	N	N	N	N	N	N	N	N
	Foam core & Composite Sheeting	N	N	N	N	N	N	N	N	Y	Y	Y	Y	Y	Y	Y	Y	N	N	N	N	N	N	N	N	Y	Y	Y	Y	Y	Y	Y	Y
Fuselage Structure	Symmetry	Y	Y	Y	Y	N	N	N	N	Y	Y	Y	Y	N	N	N	N	Y	Y	Y	Y	N	N	N	N	Y	Y	Y	Y	N	N	N	N
	Airfoil	N	N	N	N	Y	Y	Y	Y	N	N	N	N	Y	Y	Y	Y	N	N	N	N	Y	Y	Y	Y	N	N	N	N	Y	Y	Y	Y
Fuselage Material	Carbon Composite	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y
Tail Structure	Conventional	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N
	H-tail	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y	N	N	Y	Y
Tail Material	Balsa Wood + Foam	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N
	Carbon Fiber	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y	N	Y
Landing Gear	Tricycle	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y
Spar Structure	C-beam	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y
Spar Material	Aluminum	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y	Y

Table 3.2 Alternative configuration concepts

The four alternative final configuration concepts are as follows:

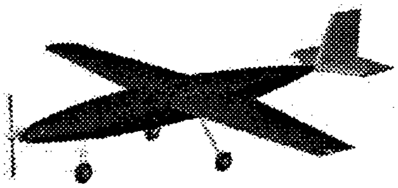
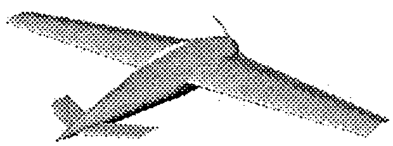
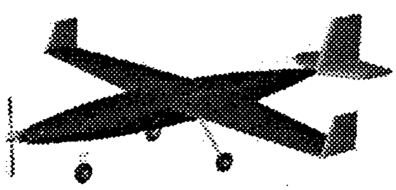
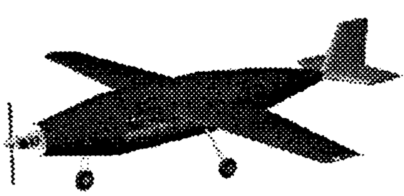
1.		
	Wing Planform	Tapered
	Wing Configuration	Monoplane without winglets
	Wing Structure	Foam core & Composite Sheeting
	Fuselage Structure	Symmetrical airfoil fitted
	Fuselage Material	Carbon fiber composite
	Tail Structure	Conventional
	Tail Material	Foam core+Balsa wood sheeting
	Landing Gear	Tricycle+faired
	Spar Structure	C-beam
	Spar Material	Aluminum
2.		
	Wing Planform	Tapered
	Wing Configuration	Monoplane without winglets
	Wing Structure	Foam core & Composite Sheeting
	Fuselage Structure	Symmetrical airfoil fitted
	Fuselage Material	Carbon fiber composite
	Tail Structure	Canard
	Tail Material	Foam core+Balsa wood sheeting
	Landing Gear	Tricycle+faired
	Spar Structure	C-beam
	Spar Material	Aluminum
3.		
	Wing Planform	Tapered
	Wing Configuration	Monoplane with winglets
	Wing Structure	Foam core & Composite Sheeting
	Fuselage Structure	Symmetrical airfoil fitted
	Fuselage Material	Carbon fiber composite
	Tail Structure	Conventional
	Tail Material	Foam core+Balsa wood sheeting
	Landing Gear	Tricycle+faired
	Spar Structure	C-beam
	Spar Material	Aluminum
4.		
	Wing Planform	Tapered
	Wing Configuration	Monoplane without winglets
	Wing Structure	Foam core & Composite Sheeting
	Fuselage Structure	Axisymmetrical
	Fuselage Material	Carbon fiber composite
	Tail Structure	Conventional
	Tail Material	Foam core+Balsa wood sheeting
	Landing Gear	Tricycle-faired
	Spar Structure	C-beam
	Spar Material	Aluminum

Table 3.3 Properties of alternative configuration concepts

The design team further studied on the final configuration selections given in Table 3.3.

3.6 Initial Guess Sizing

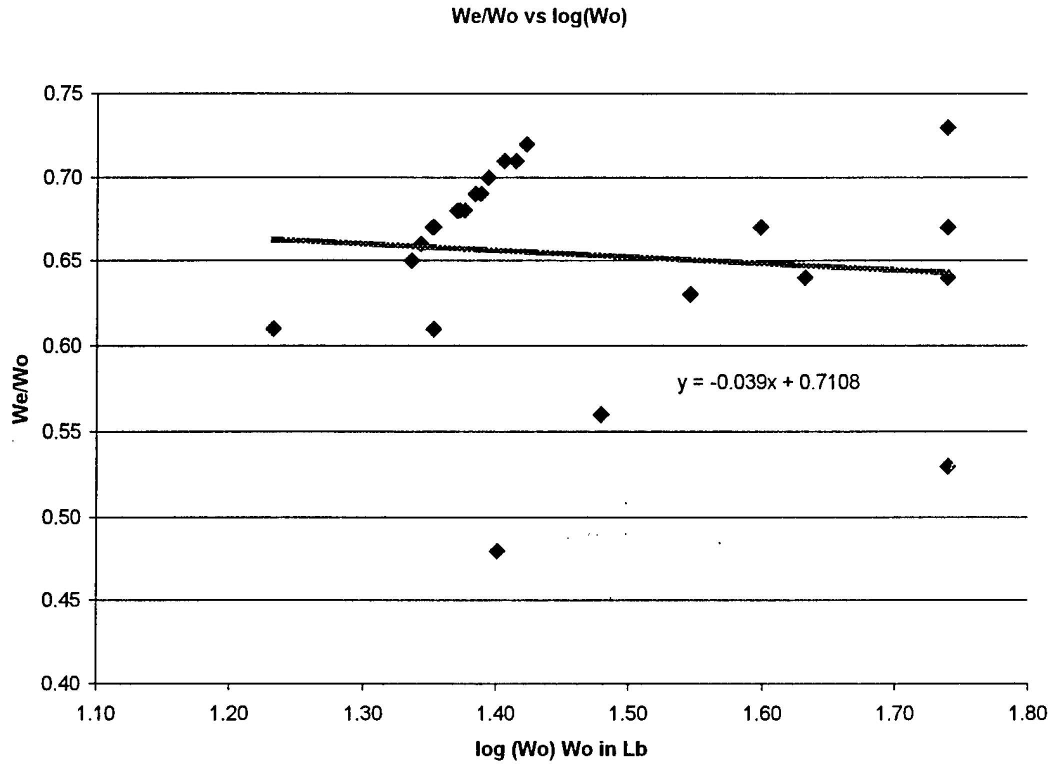


Figure 3.1 W_e/W_o vs $\log(W_o)$ for similar UAV's

In order to make a first guess sizing for this aircraft the formulas in Reference [1] were introduced. Using data of past years' successful competitors of AIAA/DBF Competition, Figure 3.1 was plotted. From the graph:

$A = 0.7108$ $\quad c = -0.039$

K_{ws} = variable sweep constant = 1 (for fixed sweep)

After manipulating the formulas in Reference 1 for W_o (Takeoff Gross Weight) one obtains:

$$W_0 = \frac{9.525}{1 - 0.7108 * W_0^{-0.039}} \tag{3.1}$$

where W_o is in lbs

In order to solve the above equation a trial and error method was used. It is found that W_0 = 25.5 lb. satisfied the equation, and the entire payload, W_p= 9.5 lb., could be carried. Therefore for this to be accomplished the empty weight of the aircraft should not exceed W_e= 16 lb.

The total weight of the aircraft is roughly estimated to be as 25.5 lb. Again referring to the competitor study, and the searches that are done about the engine performance lead us to a statistical data such that; to take-off with 10 ft span if W_o vs. Power Required for take-off statistics are plotted for the past years' competitors, an equation between Power and Total weight is found (Power Required = 26,683*W_o - 30,75) from which it is found out that the power required for take-off is 650 Watts. Therefore the selection of the motor can be done according to this power required.

4. PRELIMINARY DESIGN

4.1 Introduction

The main purpose of the preliminary design phase is to refine the design parameters that were decided in the conceptual phase. Those parameters need to be investigated deeply from many points of view, which are known to be figures of merit. These figures of merit, as explained before, are structural strength, cost, power consumption, and ease of manufacture, stability, performance and ease of transportation. The already narrowed down configurations (Table 3.3) in conceptual design phase were considered in detail during the preliminary design phase, to find out the most proper design within the specified limits.

4.2 Wing Geometry Optimization

The basic design parameters that are relevant to wing geometry are aspect ratio, wing sweep and taper ratio. In the conceptual design phase, the importance of each was explained and the limiting values for each parameter were discussed. However now in the preliminary design phase it is possible to choose the most appropriate value for each one and to improve the pre-assigned values.

Aspect ratio of the wing affects stability and stall characteristics of the airplane, drag and lift distribution over the wing. To have an optimum aspect ratio, the criteria mentioned above were discussed together. It was decided to have an aspect ratio around 5.5 in the conceptual design. However, to have an aspect ratio of 6 was found to be optimal due to the fact that it is high enough to provide less drag, and low enough not to form much stress at the wing and fuselage joints. Also the structural weight is not high, so the wing doesn't need to be strengthened too much, which reduces the weight of the wing as well.

Despite the fact that a high aspect ratio was chosen, the wing tip vortices were still a problem; so having end plates as an addition was found out to be a good solution. Yet the team thought that these end plates might cause some problems in the case of sidewind. The reason for giving importance to sidewind is because of adverse weather conditions of Wichita, which is obtained from historical whether conditions of Reference 13. Also, they affect the rated aircraft cost (RAC) as they are evaluated as vertical surfaces with no active control. Due to these reasons, the final configuration concept number 3 (Table 3.3) was not considered further. So the team decided to manufacture sharp edged wing tips. This will help us to have continuous pressure distribution at the tips.

It is quite known that the best lift distribution and lowest induced drag can be obtained by an elliptical planform. However, as building such a planform requires skilled labor and time, it was decided to have a wing with proper leading and trailing edge sweeps and a taper ratio that would do good enough as an elliptical planform. For the taper ratio, the well-known value of 0.45 was used and the sweeps were chosen accordingly to obtain the best fitting geometry into the desired ellipse. Even though, when rib structures were considered, manufacturing of a tapered wing is much more difficult than building a rectangular wing. The team kept on with that idea, searching for another manufacturing technique.

Finally, it was decided to have a wing with an inner structure formed totally of foam, covered with carbon fiber composite, with embedded spars in it. Such a technique would be easier to apply. This construction technique will be explained in more detail at Section 6.4.1.

Another important concept that was kept in mind during the preliminary design period was the fact that the wing had to be transported overseas, so that it had to be disassembled into smaller components easily. Due to this fact, the wing is designed as two pieces.

4.3 Airfoil Selection

After setting the wing geometry parameters, the airfoil selection has to be completed by another important parameter. In order to select the airfoil, the aerodynamic sub-group searched for an airfoil with the following figures of merit:

- High design lift coefficient to reduce the wing area and weight.
- High maximum lift coefficient to reduce the take off distance (due to the 200-ft limitation).
- Low drag for using the limited thrust in the most efficient way.
- High stall angle of attack
- Ease of manufacturing due to the simple techniques that has to be used for manufacturing.

In addition to the above characteristics the thickness of an airfoil, which directly affects the performance of an airfoil, has to be considered. Stall characteristics of an airfoil are directly related with its thickness. Fat airfoils (t/c >14%) stall from trailing edge. The turbulent boundary layer increases with angle of attack. At around 10 degrees the boundary layer begins to separate, starting at the trailing edge and moving forward as the angle of attack is further increased. The loss of lift is gradual. The pitching moment changes only a small amount. Also the airfoils with moderate thickness (t/c 6-14%) have gradual loss of lift because of the reattachment of the flow after separation near the nose. By using the above reminders two airfoils was chosen for final selection: SG 6042 and MH 45 (Martin Hepperle).

To obtain the characteristics of both airfoils, a database (Reference 11) was used to analyze the airfoil lift characteristics, C_l and C_d. Also, these airfoil data were checked by wind tunnel measurement data of these airfoils and it was found that that there was no difference.

By looking at the aerodynamic data of these airfoils (Figure 4.1 and Figure 4.2) SG 6042 becomes a more reasonable choice. It was seen that the maximum lift coefficient of SG 6042 is more suitable than MH 45. Although at high angle of attacks MH 45 has lower drag coefficient, SG 6042 has better drag characteristics at reasonable angle of attacks. Also, the stall characteristics of the SG 6042 are better than the MH 45. In terms of weight advantage, both airfoils have the same contribution. Due to these facts mentioned above, SG 6042 is selected.

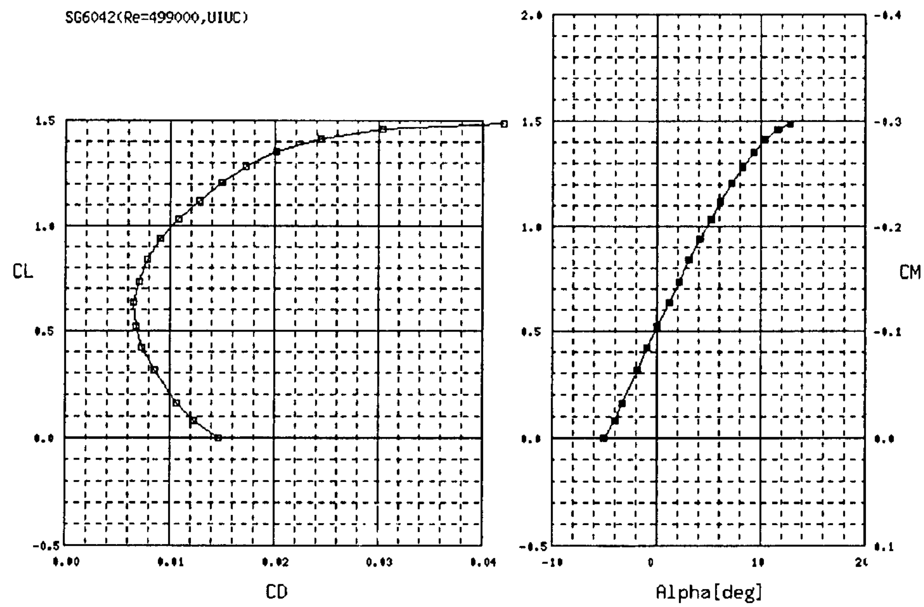


Figure 4.1 Aerodynamic Characteristics of SG 6042

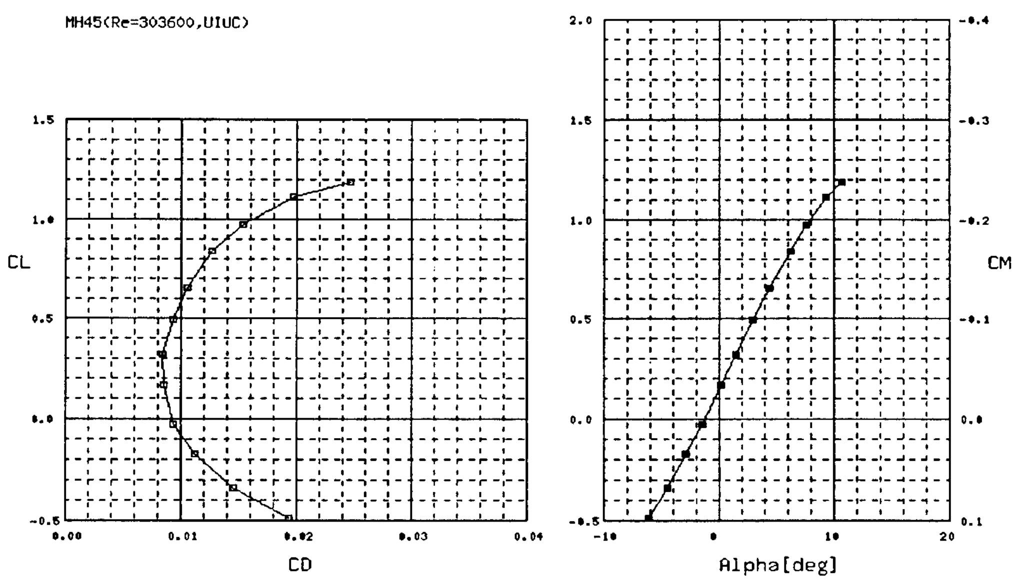


Figure 4.2 Aerodynamic Characteristics of MH 45

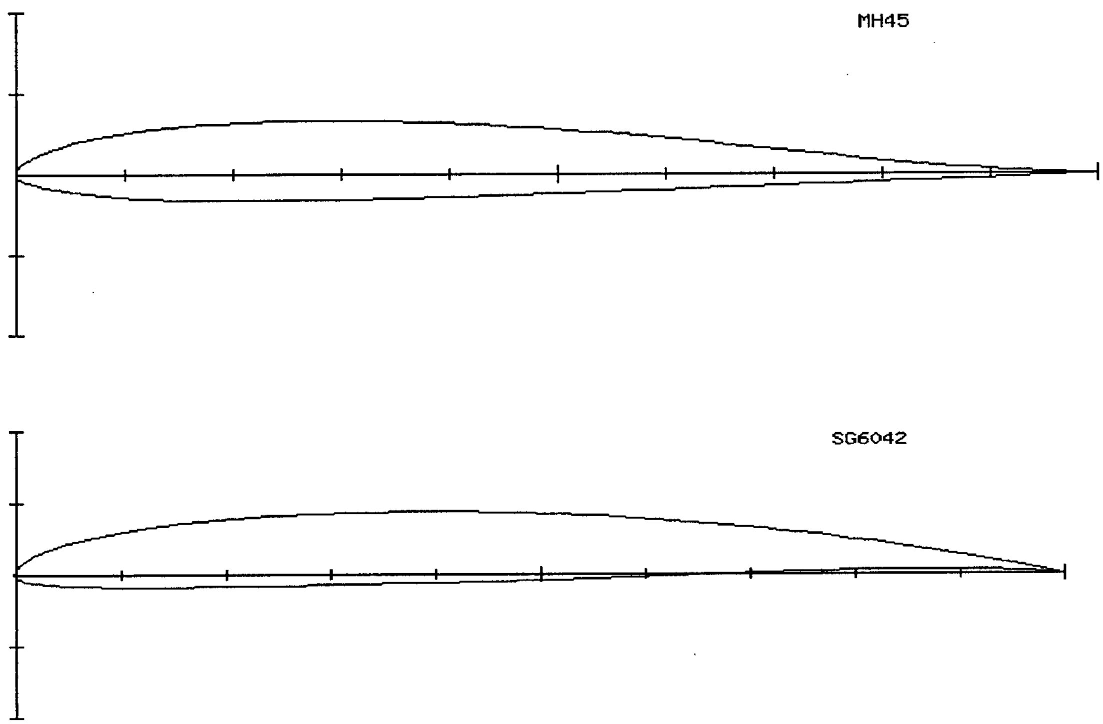


Figure 4.3 Airfoil Profiles of SG 6042 and MH 45

4.4 Tail Geometry Optimization

Tails provide for trim, stability, and control. The horizontal tail balances the moment created by the wing and a vertical tail contributes to yaw stability. During the conceptual design phase, the design team concentrated on two tail configurations, namely a conventional tail and a canard configuration.

A conventional tail consists of vertical and horizontal stabilizers located at the aft of the center of gravity (CG), in which vertical stabilizer is located above the horizontal stabilizer. For most aircraft designs, the conventional tail will usually provide adequate stability and control at the lightest weight. Probably 70% or more of the aircraft have such a tail arrangement. Conventional tail is easy to manufacture and repair.

A canard is a horizontal stabilizer located forward of CG of the aircraft. A canard contributes to the lift while providing pitch stability. However, when one decides to use canard, then he must locate the vertical tail or rudder to satisfy some amount of lift to provide yaw stability, since the moment arm of the vertical tail is reduced when wing is moved backward. Mainly two solutions to this problem can be applied. First is to sweep the wings and increase the moment arm of the vertical tail by this means. Second is to use a larger vertical tail. A larger vertical tail would increase both the weight and the rated aircraft cost. However, sweeping the wings may cause problems during manufacturing. Also, the aircraft

that we aimed to build would not need to be very much fast so there is no need to sweep the wings much. Another thing to mention about a canard is its weak handling. Usually, it is more difficult to control a canard aircraft than a conventional one.

Depending on the analysis made above, canard configuration that was listed in Table 3.3 will not be further analyzed. Therefore, spin recovery characteristics and tail areas will be calculated for conventional tail. For the vertical and horizontal stabilizers, an airfoil selection study was carried out. And the team decided to use NACA 64A0010 airfoil profile for both stabilizers.

4.5 Control Surface Optimization

In the very beginning of the design period, the decision between a flaperon and a flap and an aileron combination was made and an area ratio was determined. Previously determined flaperon size is found to be enough at this stage of the design. Flaperon hinge line is determined to be always at the same percent of the airfoil, so the lift would be distributed evenly when the flap is deflected. Flaperon size is revised as having %25 chord of wing at any section and %82 of span. Finally the areas of the rudder and elevator were calculated according to new assigned vertical and horizontal tail areas.

4.6 Fuselage Optimization

During the conceptual design phase, two types of fuselage geometries are investigated. These were axisymmetric and symmetrical airfoil shaped fuselages. The axisymmetric shaped fuselage has the ease of manufacture and internal volume of it can be used efficiently for payload mission. This would reduce the weight of the airplane. The symmetric airfoil shaped fuselage has better aerodynamic characteristics than axisymmetric one. Besides, the fuselage would provide us an extra lift. However, when the fuselage is manufactured in the shape of an airfoil, there would be some wasted space, which is not desirable. Due to this year's competition rules, the balls are to be placed side by side. Using these criteria above, the design team is decided to manufacture a symmetric airfoil fitted fuselage for further analysis. For this purpose, NACA0012 airfoil is modified to fit the fuselage with a flat bottom surface and an airfoil shaped upper surface. So the final configuration number 1 (in Table 3.3) is preferred and further analyzed due to these advantages above.

Battery pack is placed into the nose part of the fuselage in order to balance the weight of the airplane and to move the CG forward. This would also provide us to access the battery packs easily.

4.7 Landing Gear Optimization

The landing gear must be strong enough to support the weight of the aircraft on the static position and the impulse upon landing being reasonably flexible to allow the aircraft to oscillate a bit in order to absorb the hitting energy. The weight of the landing gear is important. Therefore the material of which the landing gear will be manufactured must be selected by keeping these in mind, that is, the landing gear material must be strong, light and flexible.

During the conceptual design phase we have considered both the retractable and faired gears. Although retractable gears benefit from low drag, they need some empty space in order to take the gears in and the mechanisms mean extra weight. Also a single mission was including 3 takeoffs hence 3 landings. Therefore it is not very convenient to lower or raise the gears all the time the aircraft lifts off or land. Hence, we have decided on faired gears.

In the light of this selection, to determine the properties of the landing gear further, an experimental process was followed. First landing gear produced was from thin steel plate. It was sufficiently strong however it was heavy and did not provide enough flexibility. The second one was from 3-mm thick aluminum. This was lighter than the previous one however failed during impact testing. The third alternative was to produce it again from aluminum but this time the gear cross section was a U-beam. This was strong enough but the thickness and the frontal area was a big problem due to drag consideration. The team also worked on an oleo-shock absorber type landing gear. For this purpose, a circular cross sectional aluminum tube was used and a damper system was placed into this tube. And a spring was mounted around the stroke. This landing gear has better characteristics than the others due to the fact that it produces reasonable drag and better structural strength characteristics for the airplane.

4.8 Engine Size and Battery Optimization

For the optimization of the engine at the very beginning there was a roughly determined trust to weight ratio calculated by using the available statistical data. However, those data were insufficient for an electric powered aircraft therefore the search was concentrated on electric powered flight. By examining all the resources that were available, an approximate value for the power required was found. The electric motor-suppliers catalogues were investigated while keeping the competition rules in mind. Aim was to have minimum number of engines and batteries possible since the number of the motors and the weight of the batteries used directly and very badly affect the rated aircraft cost. In order to increase the endurance that would allow us to complete the whole mission the battery number must be increased as much as possible, however, the 5-lb battery weight limit further complicated the motor selection procedure. Also as stated in the DBF-2002 rules, all motors must be from the Graupner or Astro Flight families of brushed electric motors with a 40 Amp fuse.

Astro Flight Cobalt 60, Astro Flight Cobalt 90, and Graupner Ultra 3450-7 were the alternative motors selected from the motor suppliers catalog for further analysis. In order to analyze the selected motors (Astro Flight -Cobalt 60, Cobalt 90 and Graupner Ultra 3450-7) a program (Reference 3) was used. MotoCalc is a program for predicting the performance of an electric model aircraft power system, based on the characteristics of the motor, battery, gearbox, propeller, speed control and the aerodynamic data of the aircraft. MotoCalc will predict the current, voltage at the motor terminals, input power, output power, power loss, motor efficiency, motor rpm, power-loading, system efficiency, propeller rpm, static thrust, pitch speed, and run time. It can also do an in-flight analysis for a particular combination of components, predicting lift, drag, current, voltage, power, efficiency, rpm, thrust, pitch speed, and run time

at various flight speeds. It can also predict the stall speed, hands-off level flight speed, maximum level flight speed, rate of climb, and power-off rate of sink.

Before using the MotoCalc in order to narrow down the possibilities for the battery pack, it was decided to use either the Sanyo RC 2000 or the SR 2400 max batteries, since they are not heavy, so they can be packed together to provide enough power within the 5-lb limitation.

To find out the most proper condition that can provide 10 minutes of flight with 200 ft take off distance and an adequate speed; analysis were made with MotoCalc, for too many combinations of motors, batteries and propellers. Performance calculations for each alternative motor are presented in Figure 4.4, Figure 4.5 and Figure 4.6. Finally it was decided to use two Graupner Ultra 3450-7 motors with 18"x12" propellers, with 36 SR 2400 max. batteries.

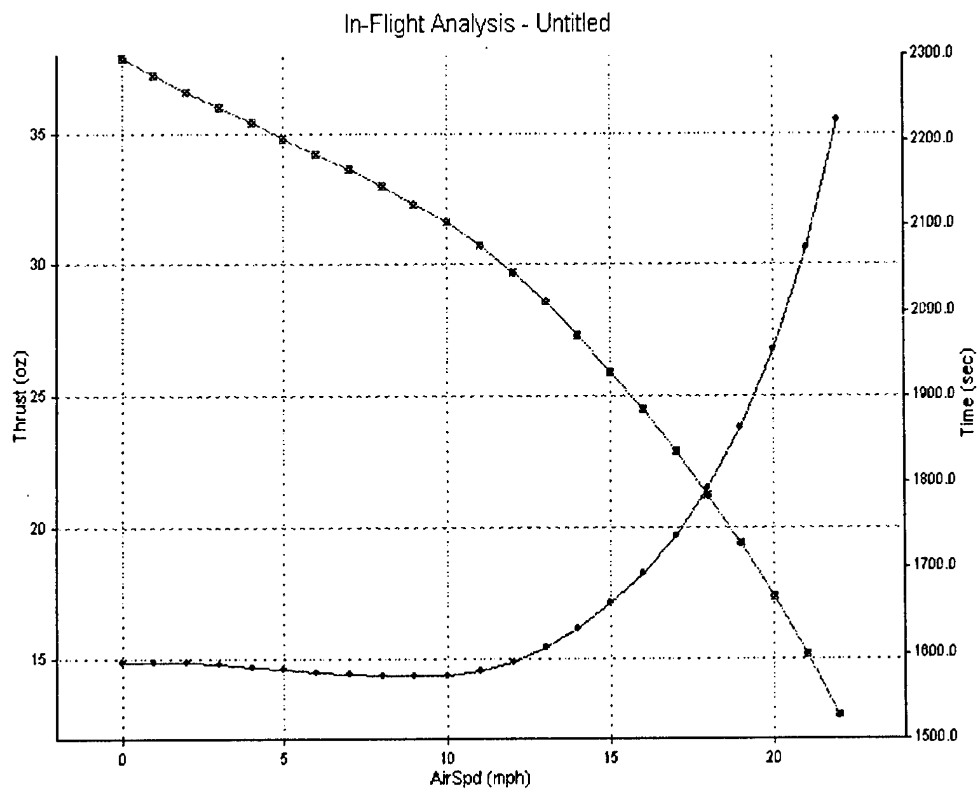


Figure 4.4 Performance Analysis of Astra Cobalt 60

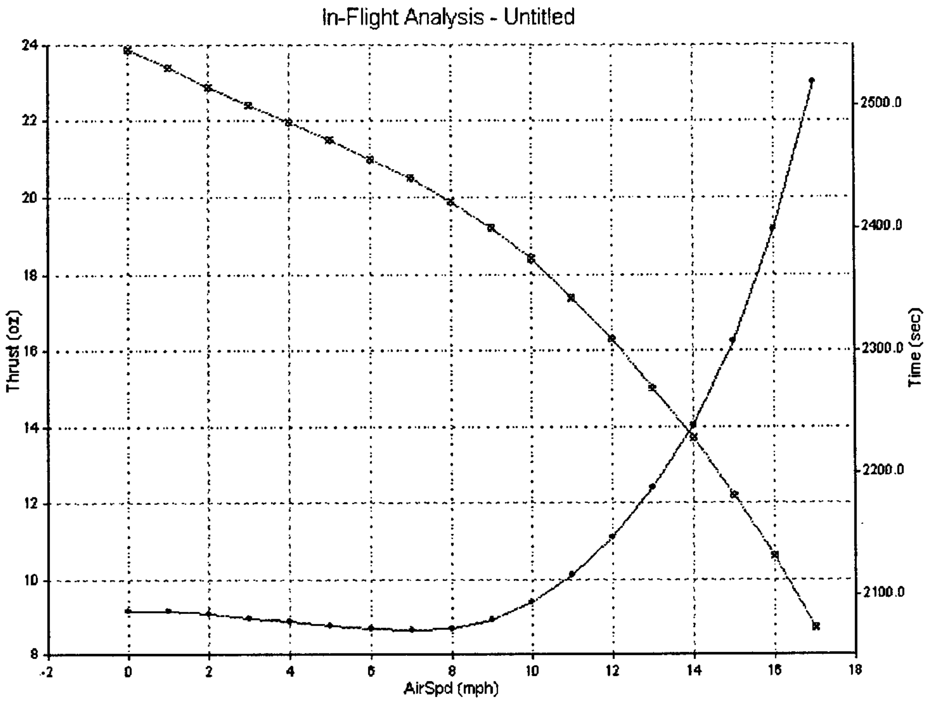


Figure 4.5 Performance Analysis of Astra Cobalt 90

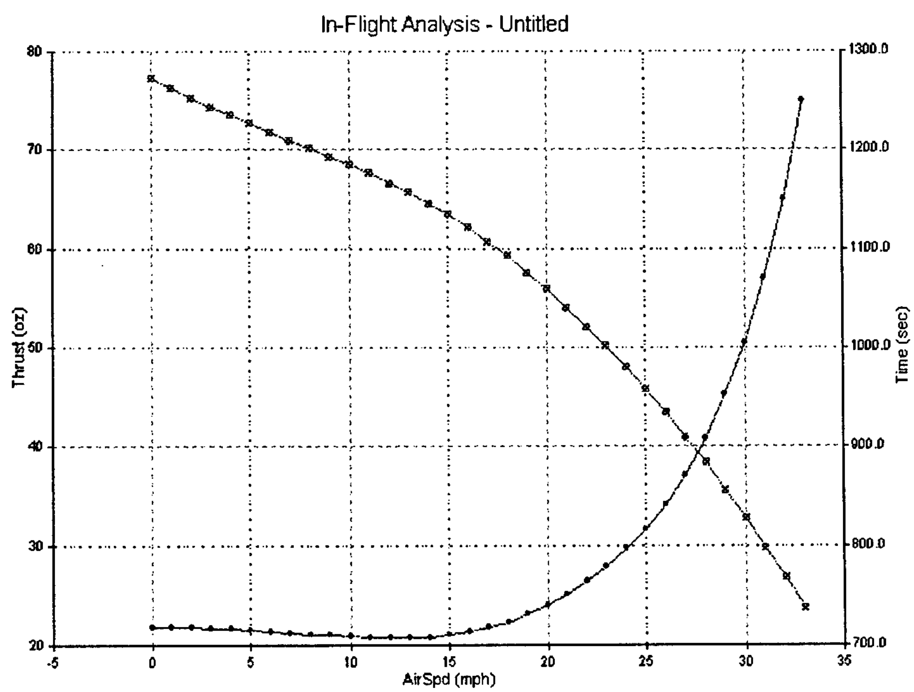


Figure 4.6 Performance Analysis of Graupner Ultra 3450-7

4.9 Preliminary Design Results

During the preliminary design phase, several modifications and assumptions made in the conceptual design phase are validated. This detailed analysis lead us to select the final configuration concept number 1 in Table 3.3. The results of the preliminary design did not affect this conceptual configuration. The detailed analysis was made for the main components such as the wing and the tail areas, the power plant and the landing gear.

Obtained preliminary design parameter are listed in Table 4.1

Wing Span	2.65 m (8.69 ft)
Root Chord	0.57 cm (1.87 ft)
Wing Area	1.1 m^2 (11.829 ft^2)
Aspect Ratio	6
Taper Ratio	0.45
Leading Edge Sweep Angle	3°
Wing Airfoil	SG 6042
Horizontal Tail Airfoil	NACA 64A010
Vertical Tail Airfoil	NACA 64A010
Tail Configuration	Conventional
Fuselage Length	2.706 m (8.862 ft)
Fuselage Maximum Width	0.351 m (1.149 ft)
Max Payload Capacity	4.464 kg (9.8 lb)
Empty Weight	10.5 kg (33 lb)
Horizontal tail	
Span	0.650 m (2.12 ft)
Area	0.1298 m^2 (0.425 ft^2)
Aspect Ratio	3.255
Vertical tail	
Span	0.310 m (1.015 ft)
Area	0.0709 m^2 (0.232 ft^2)
Aspect Ratio	1.355
Power Plant Parameters	
Number of Motor	1
Motor	Graupner Ultra 3450-7
Power	1160 kw (1.532 hp)
Cells	SR 2400 max
Number of Cells	36
Propeller	18x12 two-blade propeller
Landing Gear Specifications	
Configuration	Tricycle
Type	Faired
Material	Aluminum
Features	Oleo-shock absorber

Table 4.1 Preliminary Design Results

5. FINAL DESIGN

5.1. Introduction

In the subsequent design phases, several configurations are examined and "Configuration 1" in Table 3.3 is decided to be analysed further. This final design chapter is concerned with the final analysis of this configuration and to decide whether a modification in the design is required or not.

The final design phase involves detailed analysis on the selected design configuration. Stability, performance and rated aircraft cost analysis is performed to examine whether the design requirements were satisfied or not. Weight and balance analysis was essential to validate the assumptions made during the early design phases.

5.2. Stability and Control Analysis

5.2.1. Trim Plot

Trim analysis is performed for the aircraft using the methods presented in (Reference 1). Trim plot of the designed aircraft for different elevator deflection angles (delta) is shown in Figure 5.1.

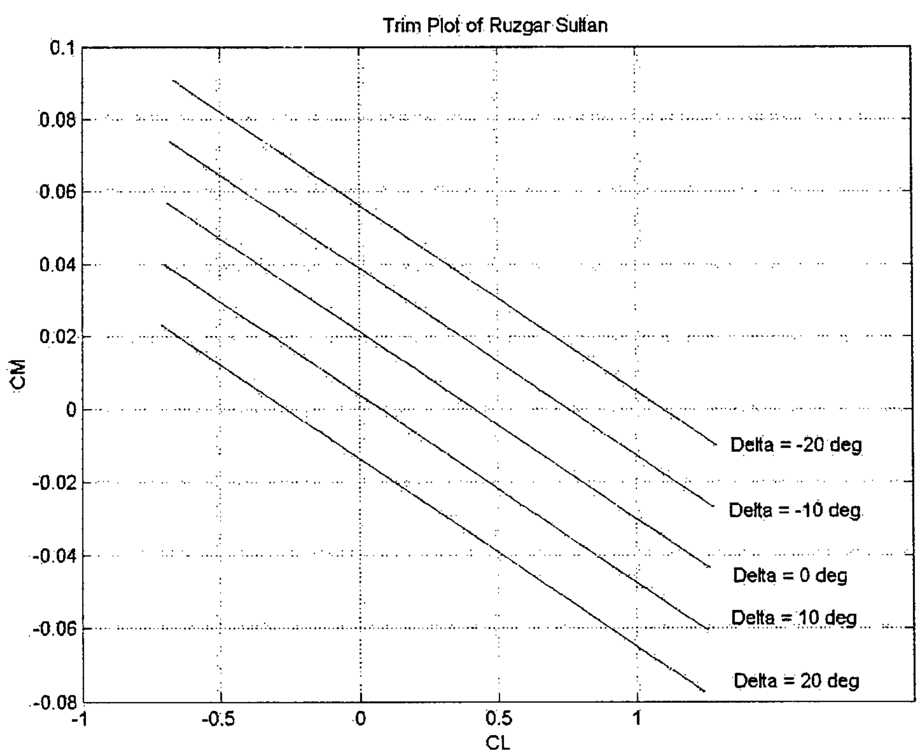


Figure 5.1 Trim Plot

As can be seen from Figure 5.1, for elevator deflection of 0°, the aircraft is trimmed approximately at a C_L value of 0.5, which is very close to the cruise C_L value of the unloaded aircraft. Since for most of the mission, the UAV will be unloaded; it was decided to adjust the elevator for empty aircraft. The results indicate that the goal is achieved.

5.2.2. Longitudinal Static Stability Analysis

Static Stability analysis is done according to the techniques obtained from (Reference 1). The results obtained are represented in Table 5.1.

	Unloaded	Loaded
Static Margin	0.4012	0.5555
Neutral Point	1.1821 (m)	1.1821 (m)

Table 5.1 Static Margin and Neutral Point data

The static margin for conventional airplane ranges in the vicinity of 0.1. Therefore the static margin of the designed aircraft might seem too high. However in the calculations of the static margin, effect of fuselage is calculated from approximate formulas. The designed aircraft has a slender fuselage which will cause vortex generation on the upper side of the fuselage. Therefore it is expected that the lift contribution from the fuselage will be enhanced. So the aerodynamic canter is expected to move to forward and static margin will be reduced. The flight tests will show whether our judgement is wrong or right.

The further stability and performance analyses is performed with keeping in mind that the secondary effects which are neglected in analytical methods will alter the behaviour of the aircraft. However this does not mean that further analyses are unnecessary. In fact they are indeed very important to initiate the flight tests safely.

5.2.3. Longitudinal Dynamic Stability Analysis

Dynamic stability analyses has been performed using methods presented in (Reference 15) and (Reference 16). The stability derivatives are calculated and the state matrix of the aircraft is created. With the help of small disturbance equations, the free response of the designed aircraft under trim conditions is analysed.

While calculating the stability derivatives, some parameters are neglected. These assumptions are made due to different reasons. For example, compressibility effects are neglected since the cruise velocity of the aircraft is far below the limits of compressibility. Whereas, especially in lateral derivatives, the effects of fuselage and some other components are neglected due to the fact that there is no simple analytical method to calculate them. More reliable methods such as wind tunnel testing or numeric calculations were not feasible. Instead some observations are left to flight tests, since the neglected

parameters and their effects are known at least qualitatively and some modifications can be made with this knowledge.

The results of longitudinal dynamic stability analyses is presented in Table 5.2, in terms of damping ratio, natural frequency, period and required time to half the amplitude of free response of the UAV for short period and phugoid flight modes.

	Empty		With Payload	
	Short-Period	Phugoid	Short-Period	Phugoid
ξ	0.50658	0.02728	0.29854	0.02683
ω_n	5.32237	1.04849	6.47591	1.01397
T (period)	1.36921	5.99484	1.0166	6.19885
t_d (time for half amplitude)	0.2559	24.1259	0.3569	25.3677

Table 5.2 Longitudinal stability parameters

The most important parameter to consider is angle of attack. The free response of fully loaded and unloaded UAV for a 5-degree disturbance in angle of attack is shown in Figure 5.2 and Figure 5.3.

The behavior of loaded aircraft is perfect. However for the unloaded case, the damping is not very good. Yet the amplitude of the disturbance reduces to 1° within less than 5 seconds. Therefore only the careful respond of the pilot to the aircraft is enough to kill this behavior of the aircraft.

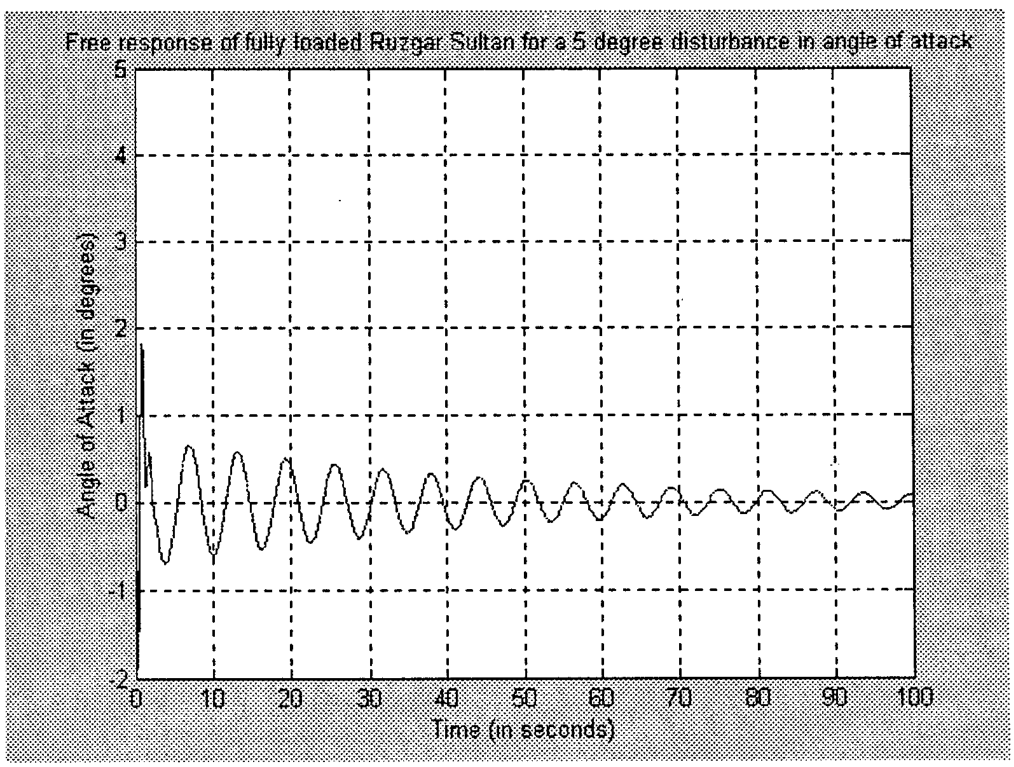


Figure 5.2 Longitudinal Response of loaded Ruzgar Sultan

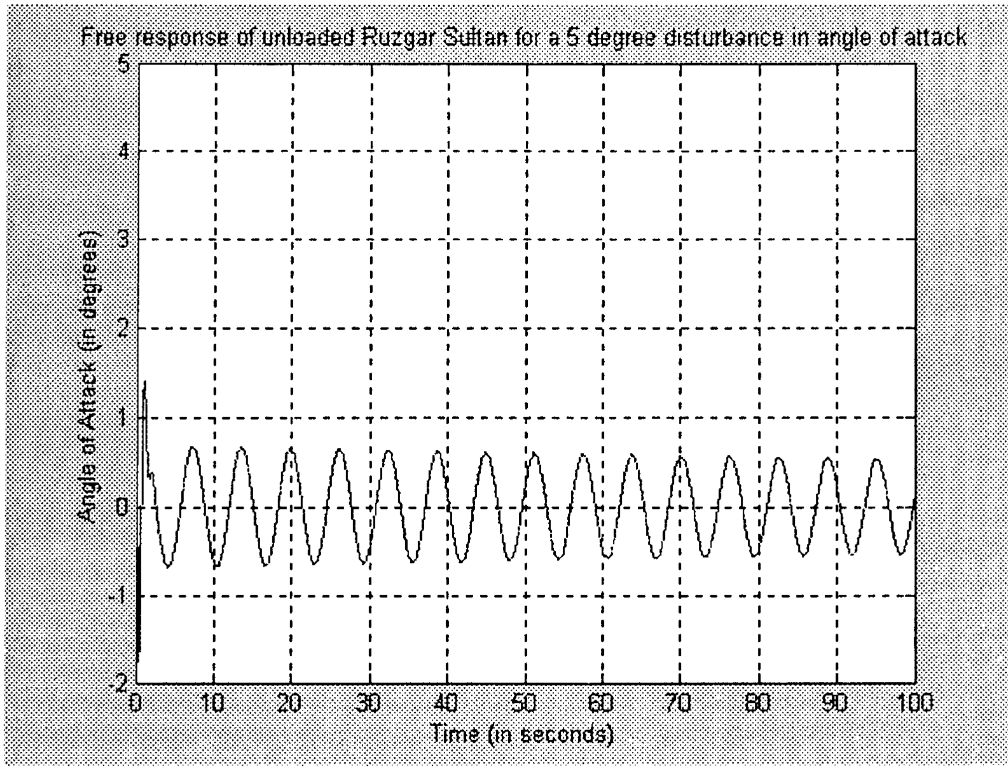


Figure 5.3 Longitudinal Response of unloaded Rüzgar Sultan

5.2.4. Lateral Stability Analysis

Lateral stability analysis is undertaken in a similar fashion. Lateral neutral point is found using the methods presented in (Reference 1) and it is found to be 0.9615 m for loaded case and 1.0956 m for unloaded case. Lateral dynamic analysis is again based on (References 2 and 3). The eigenvalues of the lateral characteristic matrix of the designed aircraft is shown in Table 5.3

	Empty	With Payload
λ_1	0	0
λ_2	-11.5236	-11.6240
λ_3	-0.0948	-0.2058
$\lambda_{4,5}$	-0.3833±1.7928j	-0.5591±1.7596j

Table 5.3 Eigenvalues of Lateral Characteristic Matrix

The results are satisfactory and it is expected that Rüzgar Sultan will have a very good lateral stability. Of course, as indicated in (Reference 17), especially in lateral dynamic analyses, the methods available are not fully satisfactory and the real behaviour could deviate from the calculations significantly.

The free response of Ruzgar Sultan for a 5° disturbance in angle of sideslip and bank angle are shown in Figures 5.4 to 5.7

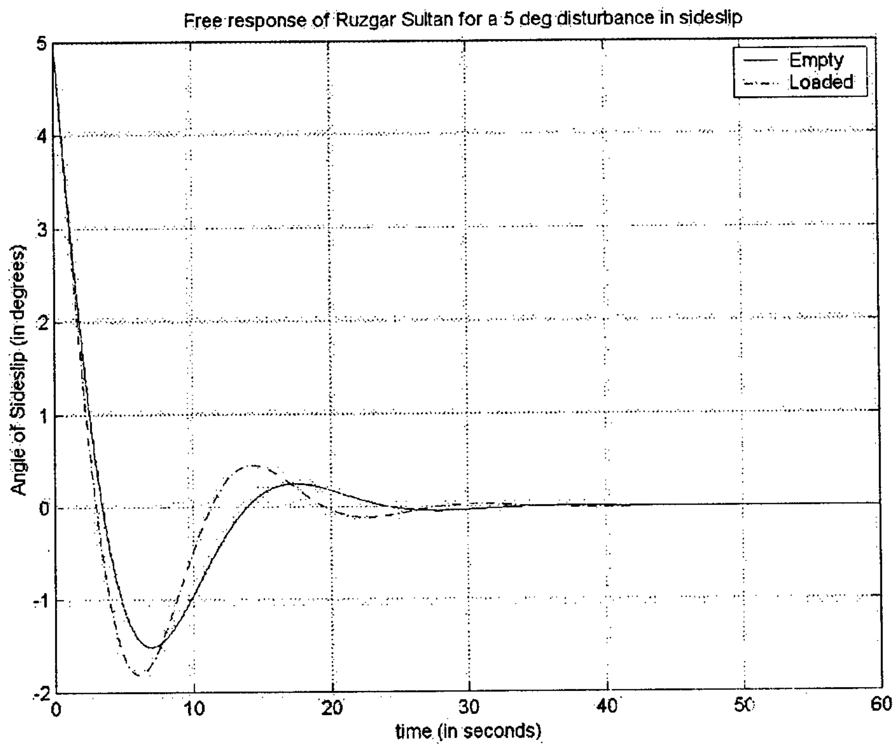


Figure 5.4 Lateral Response of Rüzgar Sultan

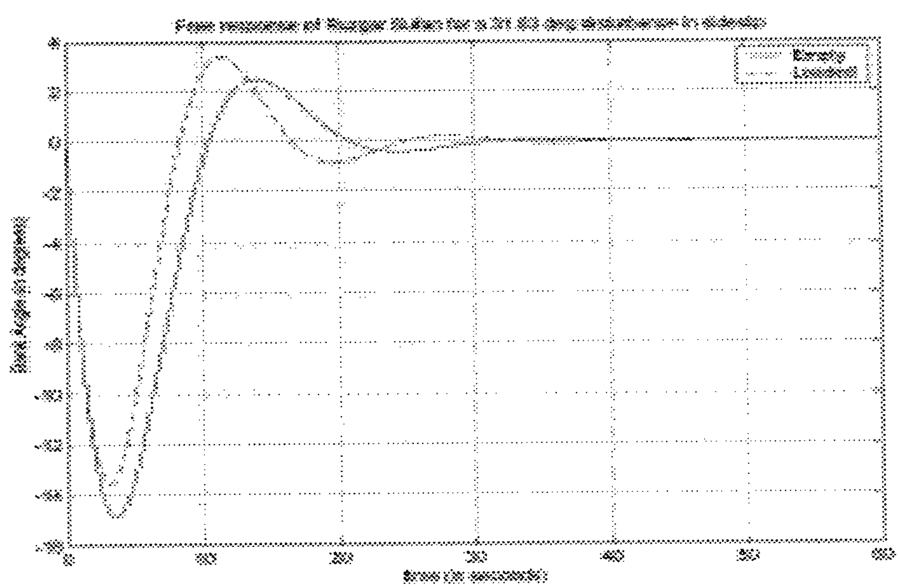


Figure 5.5 Lateral Response of Ruzgar Sultan

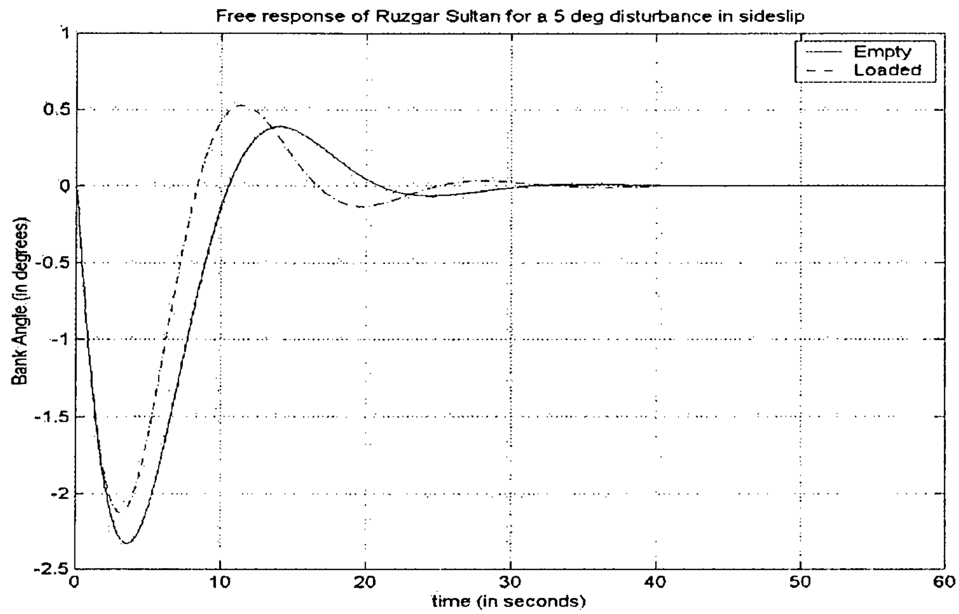


Figure 5.6 Lateral Response of Rüzgar Sultan

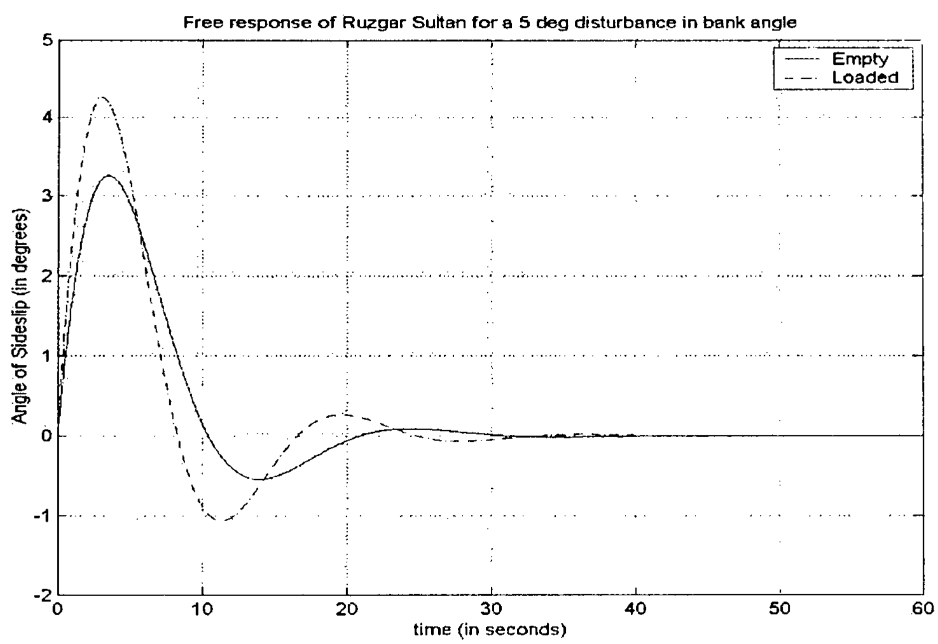


Figure 5.7 Lateral Response of Ruzgar Sultan

In the previous contest at Wichita, the wind speed of maximum 25 mph is observed. For the designed aircraft cruising at 18 m/s, such a crosswind would correspond to 32.83 degree of sideslip. The response of Rüzgar Sultan in terms of sideslip angle, β and bank angle, φ is shown in Figure 5.8.

As shown in Figures 5.3 to 5.8, Rüzgar Sultan is capable of handling very harsh conditions of Wichita.

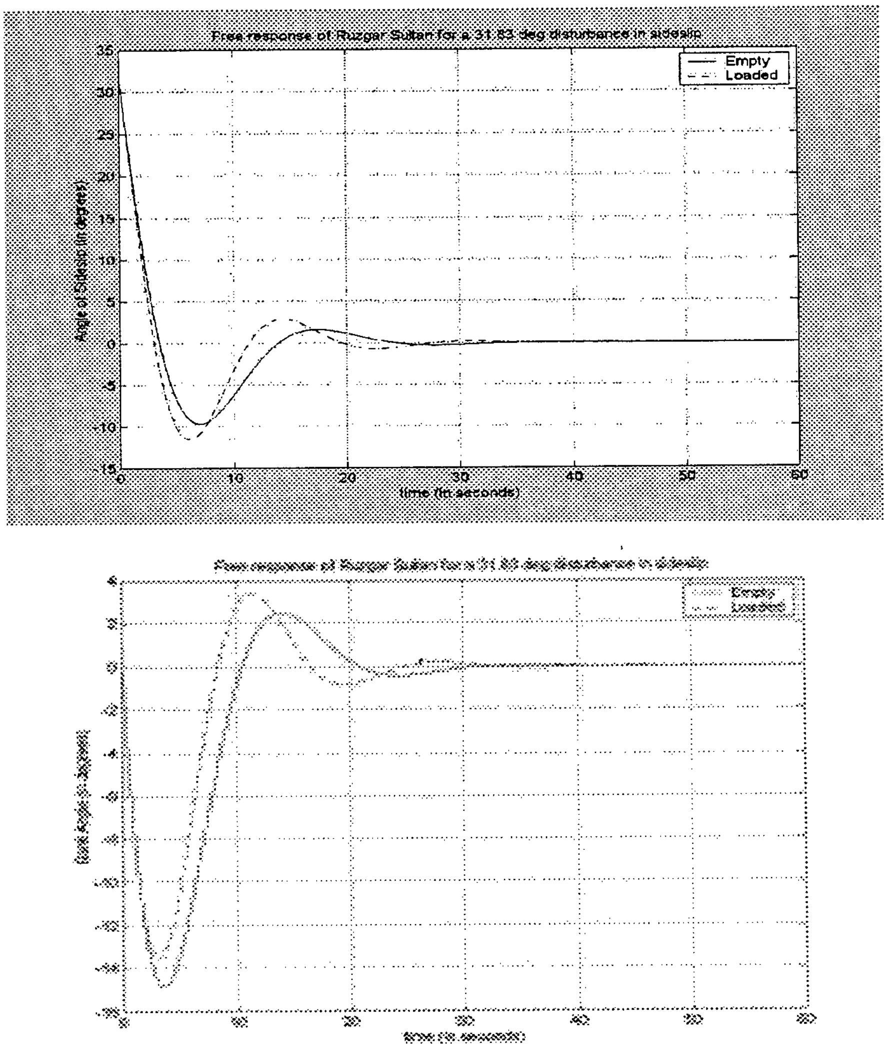


Figure 5.8 Lateral Response of Rüzgar sultan

5.2.5. Dutch-Roll Analysis

The complex eigenvalues, shown in Table 5.3 corresponds to the Dutch-Roll oscillation of the aircraft. The damping ratio, natural frequency, period and time to half the amplitude are tabulated in Table 5.4 .

	Dutch-Roll Oscillation	
	Empty	With Payload
ξ	0.2091	0.3028
ω_n	1.8333	1.8463
T (period)	3.5047 sec.	3.5708 sec.
t_d (time for half the amplitude)	1.8 sec.	1.2342 sec.

Table 5.4 Dutch-Roll properties of Rüzgar Sultan

Dutch-Roll characteristics of the Rüzgar Sultan are satisfactory. Damping ratio and time required for half the amplitude of the oscillation are appropriate to handle and recover from Dutch-Roll condition.

5.3. Performance Data

	Unloaded	**Fully Loaded**
Weights		
Payload	4560 g.	
Payload Fraction	0.305	
Electrical Components	2850.4 g.	
Total Weigh	10.379 kg	14.939 kg.
Climb Performance		
Climb Angle	29.5°	20.45°
Climb Rate	6.86 m/s	3.12 m/s
Turning Performance		
Load Factor	3.37 g	2.37 g
Sustained Turn Rate*	1.57 rad/s	1.28 rad/s
Instantaneous Turn Rate*	1.78 rad/s	1.04 rad/s
Cruise Performance		
Cruise Velocity	18 m/s	18 m/s
Stall Velocity (flaps up)	10.04 m/s	12.06 m/s
Maximum Velocity*	42.6 m/s	41.2 m/s
Range	10282.58 m	5187.64 m
Endurance	31:25 min.	17:20 min.
Landing & Take-off Performance		
Touchdown Velocity	12.048 m/s	14.472 m/s
Landing Ground Roll (for μ=0.5)	15.74 m	28.12 m
Lift-off Velocity	11.04 m/s	13.27 m/s
Take-off Ground Roll (for μ =0.5)	27.7m	38.29m

Table 5.5 Performance Data (*{at 3000 ft altitude}, μ {friction coefficient})

The estimated performance data are listed above which were calculated in the detail design phase. Performance criteria for weight, takeoff, and cruise, landing, turning performance are investigated.

5.4. Component Selection and System Architecture

5.4.1. Propulsion system

Single Graupner Ultra 3450-7 Electric Motor with 3:1 Gear reduction ratio is used to drive an 18x12 inches two-blade wooden propeller. For supplying energy to the system a pack of 36 cells are used which satisfies the 5 lbs battery weight limit. The cells selected are SR batteries 2400 max series 1.2V NiCd batteries, which have a capacity of 2400 mAh. The motor and the gearbox assembly is mounted to the nose of the aircraft using steel sheet mount structure. The motor requires a minimum of 28 Volts to run where it gives 429 rpm/V. It has a resistance of 0.078 Ohms and has an idle current of 1.61 Amps. The assembly has a capability of outputting 1030 Watts of power to the shaft, where it produces 50.83 Newtons of static thrust. The maximum speed that can be reached using this assembly is 24 m/s. The speed control works in a range of six to thirty six cells without any tuning or setup required. The power linkage is done by low resistance wores and zero-loss golden contact connectors.

5.4.2. Control system:

A Futaba T9ZHP transmitter and RI29DP receiver system is used. The system has a synthesizer, which allows changing the frequency channel, which is useful for conflict situations. The system has nine channels, where only five of these were used. The system also includes fail-safe feature, which is a requirement for the competition. The servo cycle curves are adjustable, and include a sub-trim feature, which helps much at the phase of testing. One Futaba S3001 ball-bearing servo is used for elevator, where two S9202 high-torque ball-bearing servos are used for the flaperons. Also an S9204 metal-geared, high speed, high-torque servo is used to control both the nose gear and the rudder. A dual conversion system filters the signals twice for maximum reliability.

5.4.3. Weights and Balance

The UAV is composed of body, wings, tail, an electric engine, motor mount, 24 softballs, two main ribs, 4 stiffeners, 36 batteries, main and nose landing gear.

All the area, volume calculation are achieved by "Rhinoceros nubs modeling for Windows" software. And the calculation errors of the software are the 1/10000 times of the millimeter.

For fuselage and wing, we don't need volumetric density since the weight calculation of the body can be made by materials (graphite/epoxy composite) density, which is in gr/mm^2. We have cut some parts of the fuselage in order to reduce weight.

Empennage is produced from balsa wood filled with polystyrene foam, since it is needed to be light for center of gravity location.

Ribs and stiffeners are made with Aluminum in order to be light and durable for structural needs.

Motor mount is produced from steel sheet of ST 37. There are 36 battery cells where each piece weighs 58.9 grams.

Landing gears are manufactured from aluminum also with the same reason with ribs and stiffeners. Also oleo shock absorbers are used in main landing gear in combination with steel springs, where on the nose landing gear, only a steel spring is used.

5.5. Weight Summary

The obtained weight summary is shown in Table 5.6 below in detail.

Component	W (gr)	Volume (mm^3)	P (gr/cm^3)	X (mm)	$W_{Component}$ (gr)	Moment (gr.mm)
Body	1735.45657	N/A	N/A	1182.6	1735.456567	2052368.291
Wings	2820.66339	N/A	N/A	N/A	2820.663388	3545931.097
Tail	632.74817	N/A	N/A	2507.9	632.7481698	1586862.808
Main Rib	N/A	116771.219	2.7	1247.1	315.2822913	393188.5455
Sec. Rib	N/A	104123.57	2.7	1247.1	281.133639	350601.7612
Main Stiff's	N/A	217,472.370	2.7	1247.1	587.175399	732266.4401
Sec.Stiff's	N/A	113,543.470	2.7	1247.1	306.567369	382320.1659
Elec.Motor	730	N/A	N/A	127.29	730	92921.7
Batteries	2120.4	N/A	N/A	281.92	2120.4	597783.168
Motor Case	N/A	36,899	2.7	153.1	99.6268761	15252.87473
Main LG	600	N/A	N/A	1294.9	600	776964
Nose LG	150	N/A	N/A	516.07	150	77410.5
Balls&Foam	4560	N/A	N/A	918	4560	4184383.68
Total Moment with Payload (softballs)						14788255.03
Total Moment without Payload (softballs)						10603871.35

With Payload (Softballs)	Total Weight	= 14939.0537gr
	CG Location from Nose	= 989.906 mm
Without Payload (Softballs)	Total Weight	= 10379.0537gr
	CG Location from Nose	= 1021.66 mm

Table 5.6 Weight Summaries

5.6. RAC Worksheet

According to Rated Aircraft Cost (RAC) calculation of this year, the team calculated the RAC worksheet for our final design in Table 5.7.

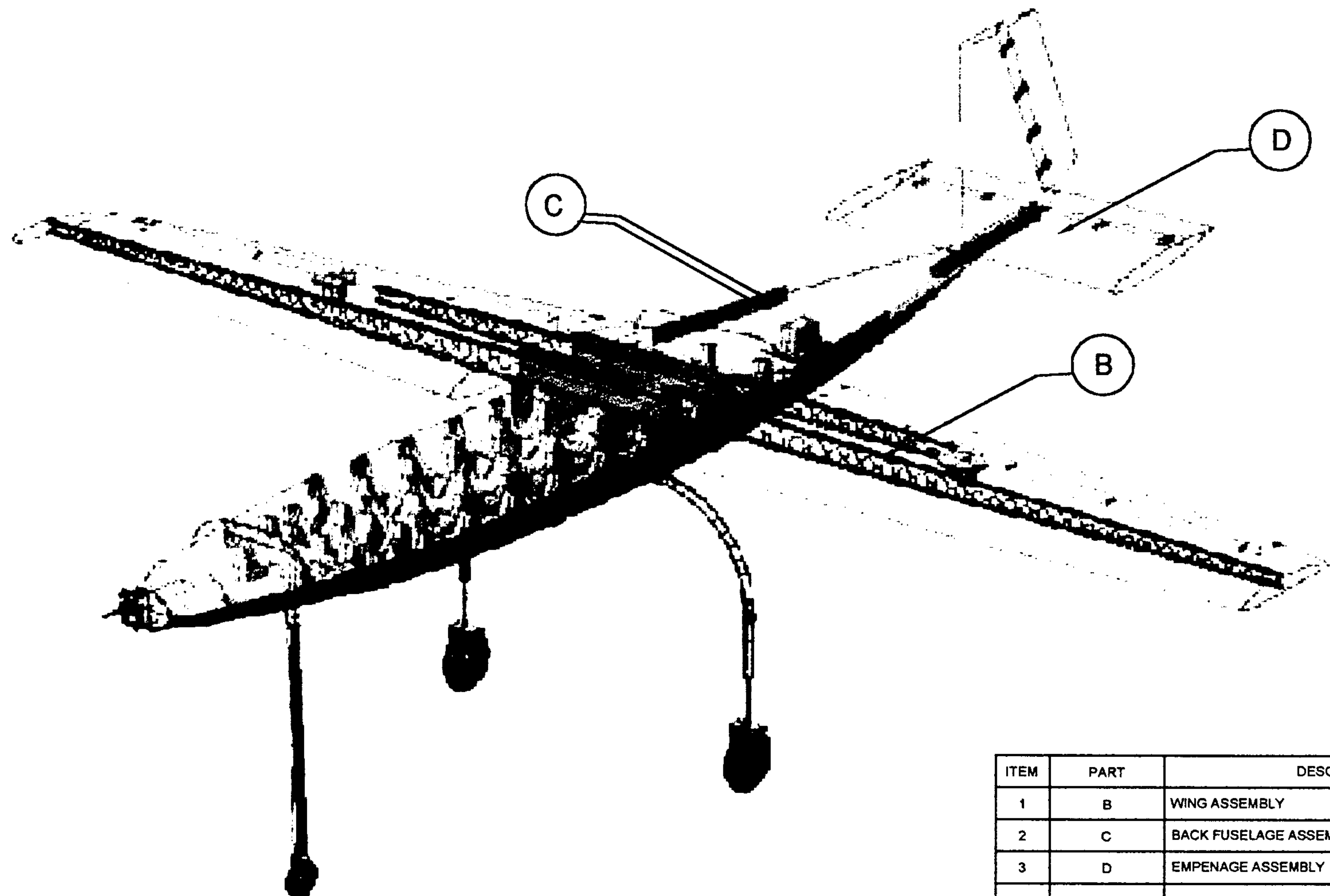
Empty Weight	22.88 lb.
Battery Weight	4.675 lb.
Wing Span	9.02 ft
Maximum Chord Length	1.81 ft
Number of Control Surfaces	4
Maximum Longitudinal Length	7.64 ft
Vertical Surface Without Active Control	0
Vertical Surface With Active Control	1
Horizontal Surface	1
Servo/Speed Controller	5
Number of Engines	1
Number of Propellers	1
Rated Aircraft Cost	**13.9013**

Table 5.7 RAC worksheet

5.7. Total Score

With the parameters tabulated in section 5.3.1., the total time required to accomplish the mission is found to be 4:04 min. During calculations, the time required for loading/unloading is taken as 30 sec. With safety factor 80%, the time required becomes 5:05 min. Since 24 softballs will be loaded, the total flight score is expected to be **0.09836**, if the time is taken in seconds.

If a written report score of **80** is taken, with the rated aircraft cost of **13.9013**, total score becomes **11306.47**. However this value does not mean anything, since the score formula is revised. Therefore we can not compare this value with the previous years scores.



ITEM	PART	DESCRIPTION
1	B	WING ASSEMBLY
2	C	BACK FUSELAGE ASSEMBLY
3	D	EMPENAGE ASSEMBLY

METU DBF2002 — Middle East Technical University, Ankara, TURKEY

UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE MILLIMETER

TOLERANCES
DECIMALS: FRACT ANGLES
X ±.1 ±1/10 ±.5'
XX ±.01

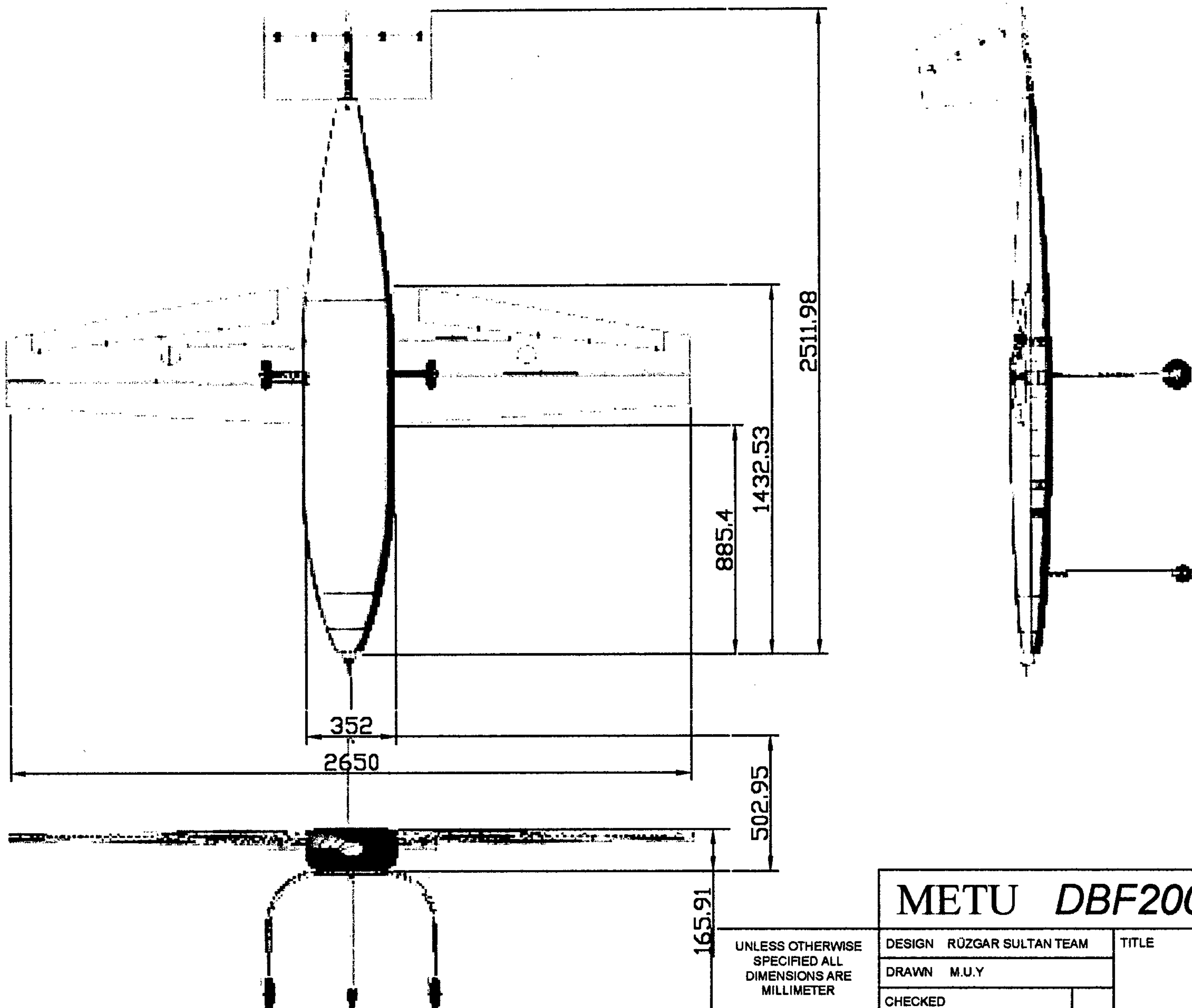
DESIGN	RÜZGAR SULTAN TEAM
DRAWN	M.U.Y
CHECKED	
APPROVED	
DATE	3/01
SCALE	NTS

TITLE: Main Assembly

MAT'L NOTED — SHEET 1 of 4

DRAWING NO. A-001

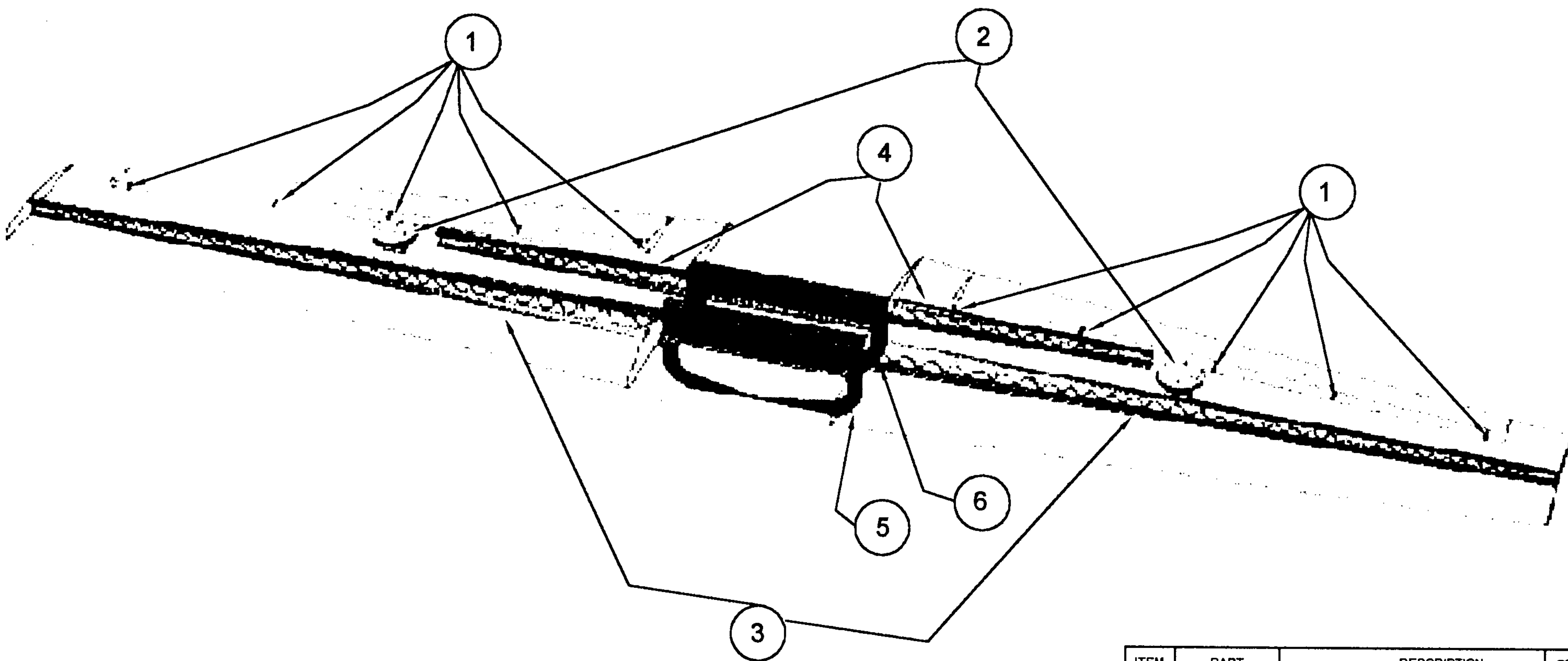
NOTE:
Unless Otherwise Specified All fuselage Covered with Carbon Fiber Composite



UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE MILLIMETER

TOLERANCES
DECIMALS: FRACT ANGLES
X ±.1 ±1/10 ±.5'
XX ±.01

METU DBF2002		Middle East Technical University Ankara, TURKEY	
DESIGN RÜZGAR SULTAN TEAM		TITLE 3-View	
DRAWN M.U.Y			
CHECKED			
APPROVED		MAT'L	SHEET 2 of 4
DATE	3/01	DRAWING NO. A-002	
SCALE NTS			



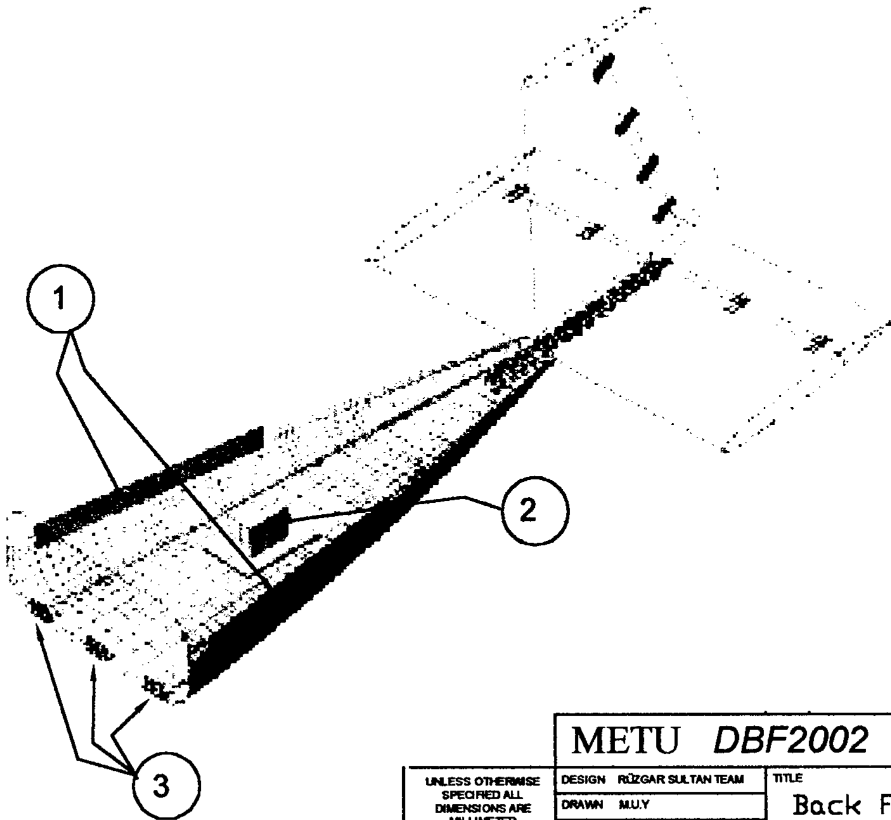
ITEM	PART	DESCRIPTION	QTY
1	B1	AILERON HINGES	10
2	B2	HIGH TECH SERVOS	
3	B3	MAIN STRUTS	2
4	B4	SECONDARY STRUTS	2
5	B5	MAIN RIB	
6	B6	SECONDARY RIB	

METU DBF2002

Middle East Technical University
Ankara, TURKEY

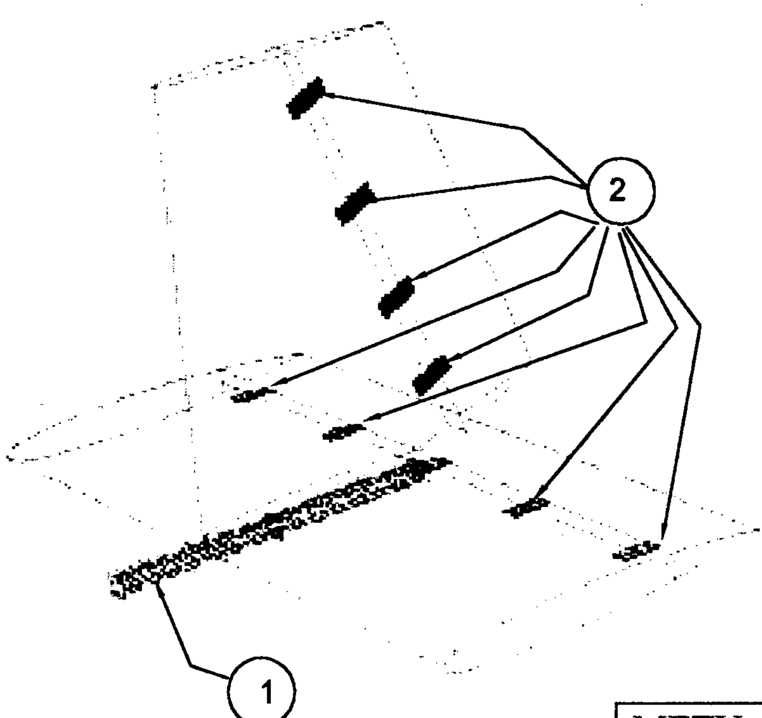
UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE MILLIMETER	DESIGN RÜZGAR SULTAN TEAM	TITLE Wing Assembly
	DRAWN M.U.Y	
	CHECKED	
TOLERANCES DECIMALS: FRACT ANGLES X ±.1 ±1/10 ±.5' XX ±.01	APPROVED	MAT'L
		SHEET 3 of 4
	DATE 3/01	DRAWING NO. B-001
	SCALE NTS	

NOTE:
1 - Struts are manufactured from 1mm Al sheet
2 - Ribs are manufactured from 2mm Al sheet



DESCRIPTION
ASSEMBLY BARRIERS
TWO HIGH TECH. SERVO
MAIN-BACK FUSELAGE ASSEMBLY

	METU DBF2002	Middle East Technical University Ankara, TURKEY
UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE MILLIMETER	DESIGN RÜZGAR SULTAN TEAM	TITLE Back Fuselage Assembly
	DRAWN M.U.Y	
	CHECKED	
TOLERANCES DECIMALS FRACT ANGLES	APPROVED	MAT'L / SHEET 4 of 4
X ±1 ±1/10 ±5' XX ±.01	DATE 3/01	DRAWING NO. C-001
	SCALE NTS	



ITEM	PART	DESCRIPTION
1	D1	FLAT HINGES
2	D2	Al ASSEMBLY PROFILE

	METU DBF2002	Middle East Technical University Ankara, TURKEY
UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE MILLIMETER	DESIGN RÜZGAR SULTAN TEAM	TITLE Tail Assembly
	DRAWN M.U.Y	
	CHECKED	
TOLERANCES DECIMALS FRACT ANGLES	APPROVED	MAT'L NOTED / SHEET 4 of 4
X ±1 ±1/10 ±5' XX ±.01	DATE 3/01	DRAWING NO. D-001
	SCALE NTS	

NOTE:
Both Vertical & Horizontal Empenage
Covered with 1/16" Balsa

6. MANUFACTURING PLAN

6.1. Introduction

After the final design consideration, the next step was to manufacture the designed airplane. Manufacturing a model aircraft is always a difficult task but it is even more difficult for a team of designers with less or no experience at all in manufacturing. Therefore, a short training period was foreseen for some of us at the start of the manufacturing phase. We learned and improved our manufacturing skills as we proceed with the construction of the aircraft. The level of expertise available within the team at the start of the project is indicated in table 6.1.

Manufacturing process for different sub-systems of the airplane was selected carefully. In addition, a manufacturing schedule is prepared to help and follow the complete manufacturing phase of the aircraft. The selection phase for the manufacturing plan was important because the structural strength and the integrity of the finished airplane would largely depend on this selection and provide us the performance as planned through various design phases. Table 6.3 shows the schedule for manufacturing of major components of the airplane and time needed to manufacture them. Table 6.2 shows the figures of merits used for comparisons.

6.2. Figures of Merit

A number of figures of merit were prepared for the manufacturing plan. It is prepared to help organizing the manufacturing process, to eliminate the unnecessary manufacturing techniques, to have effectiveness and precision during the manufacturing phase. The figures of merit chosen during the manufacturing process are availability, time required, required skill level, precision, structural strength and cost.

6.2.1 Availability

The material and the equipment that is required to produce a sub-component of the airplane are considered under the availability. It is used to eliminate of a material or an equipment that is time consuming and not available for the intended manufacturing process. If material availability is low, it is assigned with a score of –1, if the material access is a bit difficult it is assigned a score of 0. If a material readily available it is assigned a score of 1.

6.2.2 Time Required

The time required to complete a process is also rated. It is important to accomplish a process in an allocated time frame. Because the total time required manufacturing the airplane will also effect its testing before the competition. Besides, any failure or malfunctioning of a component during this testing would require its replacement and repair or even its re-manufacturing. It would be time consuming to re-manufacture these parts if they have a long production time. A process which requires longer amount of

time to complete was scored with –1, if it was a slightly time consuming then it is scored with 0, and a score of 1 is given if it requires less time to manufacture.

Available Skill Level	CNC Milling	Lathe Operations	Carbon-Composite Lay-ups	Foam hot wiring	Balsa sheeting	Aluminum working	Wood working	Monokote Applications	R/C airplane construction
Number of Skilled People	1	0	6	4	4	4	3	2	2

Table 6 .1 Skill Level of the team

	Manufacturing Plan	Availability	Time required	Required Skill Level	Precision	Structural Strength	Weight	Cost	Result
	Weighting Factor	2	2	1	2	2	2	1	
Main Wings	Foam Core w/Balsa Sheet & Rib	1	0	0	0	0	0	1	3
	Foam Core&Composite Covering w/spar	**0**	**-1**	**1**	**1**	**1**	**1**	**0**	**5**
Spar	**Aluminum**	**1**	**1**	**0**	**1**	**1**	**0**	**0**	**8**
	Carbon Fiber Composite	0	0	-1	1	0	1	-1	2
Horizontal Vertical Stb.	Balsa & Wood Rib	1	0	0	0	0	0	0	2
	Foam Core & Balsa wood covering	**1**	**1**	**0**	**0**	**1**	**0**	**0**	**6**
Cargo Door	**E-gall Composite fiber**	**1**	**0**	**1**	**1**	**0**	**1**	**0**	**7**
	Carbon Fiber Composite	1	0	1	1	0	0	0	5
Tail Boom	**Aluminum Box-Beam**	**1**	**1**	**1**	**1**	**1**	**0**	**0**	**9**
	Wooden tail boom	1	1	0	1	-1	1	1	7
Fuselage	**Carbon Composite w/ Rib**	**0**	**-1**	**1**	**1**	**1**	**1**	**0**	**4**
	Carbon Composite w/ no Rib	0	-1	0	-1	-1	1	0	-5
	Foam Covering w/ Aluminum Rib&Stiffener	1	0	-1	0	1	-1	1	1
Ball Nests	**Complete foam**	**1**	**0**	**0**	**0**	**0**	**1**	**1**	**5**
	Carbon Composite Fiber	0	-1	-1	0	1	0	-1	-2
Motor Mount	Aluminum Chassis	0	0	0	1	0	0	0	2
	Steel Chassis	**0**	**0**	**0**	**1**	**1**	**0**	**0**	**4**
Landing Gear	Aluminum plate	1	0	0	0	0	1	1	3
	Circular Aluminum w/ oleo shock absorber	**1**	**-1**	**0**	**1**	**1**	**0**	**0**	**4**

Table 6.2 Manufacturing Plan Considered

			December '01				January '02				February '02				Mar '02	
Task Name	*Start*	*Finish*	3	10	17	24	1	8	15	22	1	8	15	22	06	
Wings																
Wing Moulds	10.12.01	27.12.02		■	■	■										
Composite Lay-up	16.01.02	20.01.02							■							
Wing Inner Foam	15.01.02	24.01.02							■							
Wing Spars	22.01.02	25.01.02								■						
Finish Work	01.02.02	17.02.02									■	■	■			
Empenage																
Airfoil Foam Hot-Wiring	22.01.02	26.01.02								■						
Balsa Covering	23.01.02	26.01.02								■						
Finish Work	1.02.02	7.02.02									■					
Fuselage																
Fuselage Mould	10.12.01	27.12.02		■	■	■										
Composite Lay-up	16.01.02	20.01.02							■							
Inner Ball Nest	22.01.02	25.01.02								■						
Ribs	22.01.02	06.02.02								■	■					
MountMotor & L.G.	01.02.02	28.02.02									■	■	■	■		
Finish Work	01.02.02	02.03.02									■	■	■	■		
Balls Nest Preparation																
Foam Hot-wiring	22.01.02	25.01.02								■						
Finish Work	22.01.02	25.01.02								■						
Landing Gear																
Circular Aluminum	22.01.02	27.01.02								■						
Damper	23.01.02	06.02.02								■	■					
Spring	22.01.02	28.01.02								■						
Finish Work	01.02.02	04.02.02									■					
Tail Boom																
Box-Beam Aluminum	01.02.02	12.02.02									■	■				
Control Systems																
Servo&ControllerMount	13.02.02	18.02.02										■				
Final Preparation																
Testing	15.02.02	05.03.02											■	■	■	
Manokote Application	08.02.02	27.02.02										■	■	■		

Planned time

Completed earlier

Completed later

Final Exam.

Not Completed (Only ground test)

Table 6.3 Manufacturing Plan Schedule

6.2.3 Required Skill Level

Some manufacturing processes require extensive skill. So it is desirable and preferable to use easy manufacturing techniques for the team. The team's skill level was summarized in Table 6.1. Processes that increase the workload are decided to be eliminated. This figure of merit scored –1 for the processes that required specialized skill. 1 is given if it did not require any special skill.

6.2.4 Precision

During the manufacturing, not all parts of the aircraft require same degree of precision. Some parts require more strict tolerances than the others. Hence, a score of –1 is assigned to manufacturing processes that required high precision and a score of 1 is assigned to processes that require high precision and a score of 1 is assigned to processes that require relatively low precision.

6.2.5 Structural Strength

Structural strength and integrity of the manufactured airplane is important for the project. All the processes used in the construction should provide the desired strength to the finished product. If the structural strength desired couldn't be met, a score of –1 is assigned for its structural strength. A process providing the desired structural strength is scored as 1.

6.2.6 Weight

Weight of the considered manufacturing process and the material is used an important criterion for the airplane. With increasing weight of the airplane it is expected that its performance will decrease. So a score of 1 is given for any process providing weight advantages, a score of –1 is given for processes increasing the weight.

6.2.7 Cost

The cost is another important criterion for the manufacturing plan. Despite the fact that the cost of the final airplane doesn't affect the final score for this competition, the cost of the airplane is still an effective criterion for completing the construction. A manufacturing process that requires high cost is therefore scored with a –1. Where as any component that can be built with less cost, is assigned of a score of 1.

6.3 Results Obtained

Using the obtained results of the figures of merit in table 6.2 above, the manufacturing processes were chosen for the airplane. The main wing had a characteristic of a foam core and a composite covering over it with spars in it. Spars are chosen to be from aluminum. Horizontal and Vertical Stabilizers are decided to have a foam core with balsa covering in the exterior part of it. And for the payload-carrying mission, the cargo is selected to be e-glass composite fiber. Ball Nests are chosen to

be complete foam. To overcome the torsion effect of the motor at the nose of the airplane motor mount is chosen to be steel chassis. Besides, the landing gear of the airplane for landing and take-off performances was chosen to be Circular Aluminum with oleo shock absorber.

6.4 Manufacturing Plan Summary

In the following sections the manufacturing plan for the major components of the airplane will be described in detail. The result of the manufacturing phase as 3-D drawing is given in a separate part.

6.4.1 Main Wing

Construction of the main wing was the major part of the manufacturing phase. The wing should have the necessary structural strength and precision. Besides, its manufacturing method shouldn't affect the aerodynamic properties of the wing. Using these ideas, during the conceptual design phase, our team investigated two possible method of wing construction; wing with a foam core covered with balsa sheet surface and rib construction in it and a wing with a foam core having a composite covering and spar construction in it. Due to figures of merit considerations listed above in the table, wing with a foam core having a composite surface covering and load carrying spar constructions in it is chosen as the best manufacturing process for the wing.

In the manufacturing process of the wing, mainly three types of materials are used. These are; 1) carbon fiber composite skin covering, 2) foam for the core structure and 3) the aluminum load carrying C-beam spars. For the production of the carbon fiber composite skin, first a decision about its mould is given. The team decided to have two moulds for the wing, one for the lower part of the wing and the other for the upper part. The mould was made from lime tree because after the carbon fiber composite material was laid over the mould they would be cured in a furnace. Also it was taken care of having the necessary airfoil thickness length for the moulds. After the whole surface of the moulds was made smooth, they were sent to CNC operation. The desired wing shapes were machined into four separate moulds. These are two molds for the starboard side and two for the port side wings both for upper and lower parts respectively. The machining operation for one mould lasted about six hours. Designated team members observed the operation. After the operations were completed the team members decided to prepare a steel frame for all of the moulds. Since it was observed that the humidity in the mould tree would be a big problem while curing in the furnace operation and would affect the mould shape. These shape changes in the airfoil and wing shapes are not acceptable hence in order to avoid this danger the moulds should be treated. Any change in the mould shape would alter the desired airfoil shape, which in turn will affect the aerodynamic properties of the wing. Therefore the framing was done for all wing moulds. As a result, the moulds were ready for laying the carbon composite fiber over them. For this process, the preparation of epoxy was investigated. For any laying process, first the weight of carbon composite fiber was measured. Then the epoxy mixture corresponding to one third of this measured weight was prepared. On the surface of the mould before laying the carbon composite fiber, a chemical

material called mould removal was applied. This would facilitate the separation of the carbon fiber composite profile from the mould surface. Then the carbon fiber composite fiber was smeared with the prepared epoxy mixture. This was overlaid on the mould and they were put into a nylon bag in order to vacuum the moulds. The vacuuming process helped carbon fiber composite on the mould to absorb the epoxy mixture. Then the mould is ready for curing. The process of curing takes about 16 hours at 60°C. When curing is complete, the moulds are separated from the carbon composite fiber. Each part separated from the moulds are then cut with a saw to the desired shape and smoothed with sandpaper for final finishing process. Next step in the manufacturing of the wing was to produce the inner parts of the wing.

The other important part of the wing is its foam core. The designed wing of the airplane was tapered, swept with no twist. Having no twist in the wing helped us to manufacture the foam core of the wing. Firstly, foam having low density about 10 kg/m^3 but necessary strength was bonded together with the help of liquid polystyrene foam. Then the whole wing in the form of a rectangular prism with exact wing span length was obtained. With the help of the airfoil shapes at the prism's root and tip, an electrically heated hot-wire device was used to cut the foam to the required core shape. Surface finishing of the foam is realized with sand papers. Also, for spars, control surface devices and electrical connections, there was a need of place to be carved. But it is decided to make this process after the foam core was bonded to the carbon composite fiber.

For load carrying spars, there were two candidates. These were carbon composite fiber and aluminum. It would be time consuming and a bit expensive to make the spars from carbon composite fiber. So the team decided to manufacture them from 2mm thick aluminum sheets. The sheet aluminum was first cut to the exact length of the spars. The remaining sheet is cut from upper and lower parts to have a converging C-Beam structure in the direction of the wing tip. The sheet was then folded to C-beam shape. In order to reduce the weight of the spar, circular parts were cut from it along its main vertical part. This weight reduction is done in such a way that the spars are not weakened at all. Strength analysis of this reduced weight spars are done using the finite element structural analysis package ANSYS and is seen that it was okay.

The last part of the wing manufacturing was to join the carbon composite fiber skin, foam core and spars. The foam core was bonded to the upper carbon composite fiber skin of the wing and the necessary sections for control surface devices, electrical cable connections and spars are carved in the core with the help of a razor blade. So the spars could be bonded onto the smooth surfaces. And the spars were bonded to the upper carbon composite fiber skin placing on the mould. Extra foam in the shape of the C-beam's open part was also bonded. Control devices were placed into a nest in the foam core. And for repairing process it was desired to have an access door to the servos in case of necessity. The next step was to open the access door to the lower part of the skin at the part of the corresponding servo places. Then, a hole was drilled for the control devices. The lower part of the carbon composite skin was bonded onto lower part bonded to the foam with liquid polystyrene foam. Then control surfaces

were cut off from the integral structure of the wing. The five nylon pivot hinges for the control surfaces are placed in equal distances from each other were placed into the sandwiched wing. Besides, control devices were connected to these surfaces. After the sandwiched and shaped wing was manufactured the winglets are placed and bonded to the wing tips. The starboard side and portside wings were manufactured separately for the ease of transportation.

6.4.2 Tail Construction

For the tail construction of the tail plane, two types of manufacturing processes were considered; balsa wood with foam core and carbon fiber composite construction. Due to the fact that the team decided to produce various types of horizontal and vertical stabilizers, it would be expensive to produce them from carbon fiber composite. So it is decided to produce the tail from balsa wood with foam core in it.

For the tail construction, foam core were manufactured the same way as the wing's foam core. The vertical stabilizers were manufactured as a whole. In tail construction, there was no need for a load carrying spar, since sufficient rigidity and strength was provided by foam core and the balsa wood surface skin. Connection of the fuselage and the tail, vertical and horizontal stabilizer linking was another problem. For this purpose, an aluminum box-beam tail boom is manufactured. A box beam part extending from the fuselage was buried into the foam core structure of the horizontal tail.

6.4.3 Fuselage

The fuselage was manufactured from composite material with load carrying rib structures embedded into it. The carbon composite had no ribs. However, it has foam covering with an aluminum rib and stiffener system.

Because, airfoil shaped fuselage have some difficulties in manufacturing from foam. From the figures of merit results and the manufacturability of the composite material in case of destruction and reparation, team is decided to have a composite covered fuselage with to rib system in it. This will also provide a good way of connection of the landing gears to the fuselage.

The fuselage of the airplane consists of three parts; these are 1) carbon composite fiber fuselage covering, 2) rib system and 3) foam ball nests. The most important part of the fuselage was its covering. The air motion around the fuselage would be very effective over the tail, and the wings. So the composite covering would provide us with the necessary rigidity and the aerodynamic efficiency. For the manufacturing of the fuselage covering the same manufacturing technique as used for the wing is employed. The details of this technique are given in the manufacturing of the wings.

The ribs in the fuselage are important to carry the static and dynamic load of the wings, fuselage or the landing gears. The ribs in the fuselage are analyzed by ANSYS to experience the structural strength. It is seen that structural strength of the designed is enough for our airplane.

6.4.4 Power System Integration

During the conceptual and the preliminary design phases, it was decided to use single nose mounted electric motor. In order to mount the motor to the plane, a steel sheet ST37 that has a density of 768 kg/m^3 and thickness of 0.8mm was used. It has cut into proper shape, then shaped by bending and placed to the carbon composite by rivets. The motor is mounted to the mount plate by from its gearbox. Also the batteries were planned to be placed just behind the motor in order to shorten the electric power cable. For this reason a room between the motor and the carriage bay is created for the batteries. Also the speed control unit was placed over there.

6.4.5 Landing Gear

The tri-cycle landing gear is selected at the design phase. For the main landing gear oleo shock absorbers that work with oil were produced and placed in an aluminum pipe that is bent into the desired shape. The aluminum pipe is bent by filling it with dry sand for preserving the circular cross-section, heating the pipe using an oxy-acetylene torch, and bending around a solid cylinder with the desired radius. The oleo shock absorbers were easily fitted into the pipe since the outer diameter of the absorbers was the same as the inner diameter of the pipe. The wheels were mounted using steel springs and placed by screwing to the shock absorbers.

For the nose landing gear, an aluminum circular pipe was fitted into an elliptic pipe, and fixed by riveting. The elliptic pipe is left outside of the fuselage where only a small part of the circular pipe is left out which designated the stroke for the nose landing gear. A steel spring is placed at the connection of the circular and elliptic pipe, which is left out, and the structure was fitted to the aircraft by a steel pipe, which is manufactured by hand using a lathe. The fitting was done, as the outer diameter of the aluminum circular pipe is the same as the inner diameter of the steel pipe. Also the motion of the nose landing gear was limited using a pivot placed at the top of the aluminum circular pipe, which does not allow the nose landing gear to detach from the plane. Also a control horn made of aluminum is riveted to the elliptic pipe in order to steer on ground. The stroke is determined to be around 3 centimeters.

6.4.6 Results

In Table 6.2, the techniques of manufacturing used during the project are presented. The manufacturing technique for the wing is selected to be foam core structure with balsa covering and load carrying C-Beam aluminum spars in it. Tail is selected to be foam core structure with balsa covering. The fuselage was manufactured from carbon composite fiber with aluminum rib structure in it. The cargo door was manufactured from e-glass composite fiber. Aluminum spars extending from two wings are screwed to the fuselage ribs to support the wing. Oleo shock absorber landing gear is selected because of its impact resistance and strength. Finally a smooth aerodynamic surface is provided to the fuselage by covering it with monokote skin.

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2001/2002 AIAA Foundation

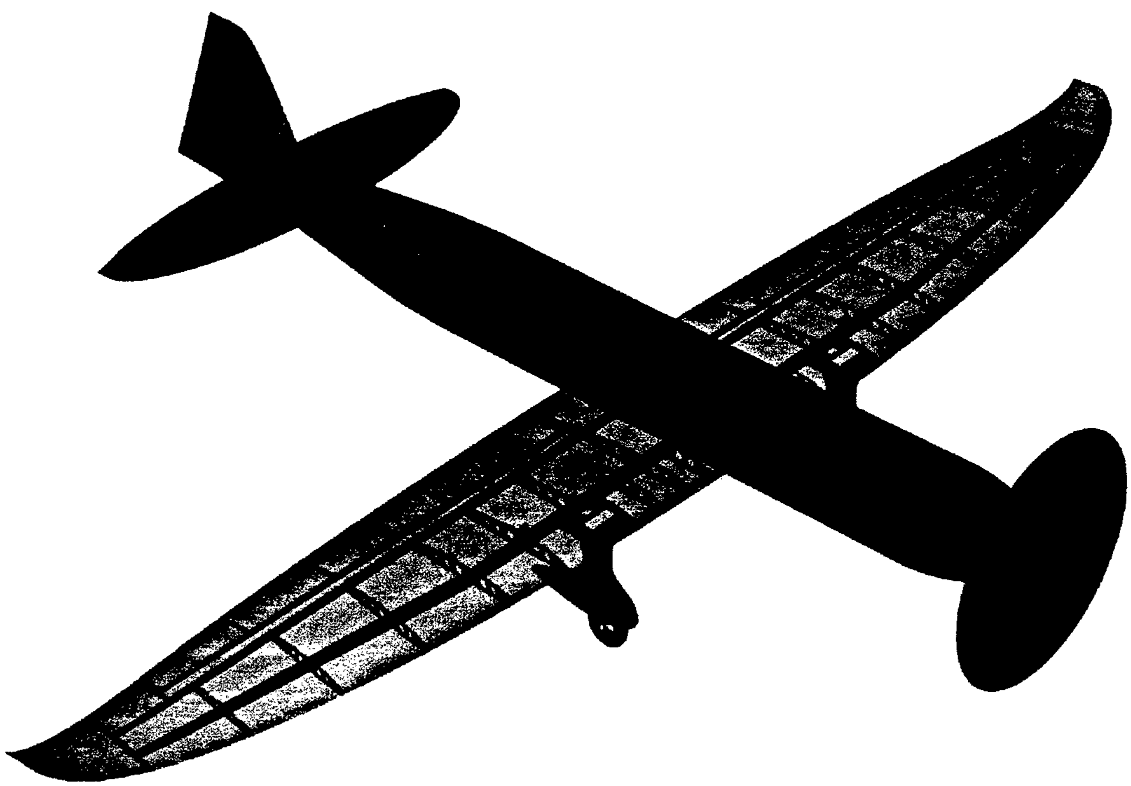
Cessna/ONR Student Design/Build/Fly Competition

Design Report: "Milk Run"

March 12, 2002

Department of Aerospace Engineering

Mississippi State University



1. Executive Summary

After a team of Mississippi State University students participated in the 2001 Design/Build/Fly Competition, several of the students were eager to enter the 2002 competition and employ the hard lessons learned from the previous year. The team *Milk Run* philosophy was to move as quickly as possible through the conceptual, preliminary, and detail design phases. Throughout the design process, flight performance and handling qualities would be heavily emphasized. An extreme design feature would not be undertaken if the effect on flying qualities was questionable. Most importantly, only enough analysis to point the design development in the right direction would be performed. Thus, the design would quickly move through the conceptual, preliminary, and detail design process. The purpose for this philosophy was to start building as soon as possible, so that plenty of flight testing could be done before the competition. The key to winning any competition is to be fully prepared!

1.1. Conceptual Design

The first stage of conceptual design was to develop a list of design parameters such as wing loading and aspect ratio. The Figures of Merit (FOM) were selected to be the various stages of the mission profile, i.e., take-off, climb out, turning, etc. The relative influence of the design parameters on each FOM was then judged. The result of this initial FOM analysis was to discover how large a role in the mission performance each design parameter played, and which design parameters have highest priority when making design decisions. The resulting ranking of aircraft qualities in order of importance was positive static stability, tractor propulsion, low drag, high aspect ratio, and low weight. This ranking was found to be in keeping with the team philosophy established at the onset of the project. A critical assumption was made to assume that the payload would be the maximum of 24 softballs, and that the battery payload would be the allowable maximum of 5-lbf. With the insights gained from this initial FOM analysis, the team then qualitatively judged the relative ranking of three configuration concepts. The conventional configuration heavily outscored all other configurations. The other concepts were ranked as follows: delta wing, canard.

1.2. Preliminary Design

The first task in the preliminary design phase was to perform a more rigorous sizing estimate. The goal was to determine the optimal payload capacity for maximum total score. Some simple hand calculations determined that an aircraft carrying 10 softballs would need to have twice the speed of a 24 softball capacity aircraft. However, there is no decisive optimum in the upper range of the potential payload capacities. The design payload of the aircraft was set at 24 softballs.

Next, the method of aircraft construction was determined in order to make more judicious design decisions concerning wing planform and fuselage shaping. Several construction choices were presented: conventional wood built-up construction, foam and fiberglass construction, and molded fiberglass

construction. Molded fiberglass construction was favored because it offered potential weight savings as well as highly accurate and smooth surfaces, which would result in low drag.

At this point, another FOM analysis was performed to determine a more precise configuration of the aircraft. These design parameters were more specific than those in the conceptual design FOM analysis. The design parameters are classified into wing planforms, tail configurations, power configuration, and landing gear configurations. The FOMs of the preliminary design parameters were broad generalizations of flight performance, handling qualities, and manufacturability. This FOM analysis resulted in a more definite aircraft geometry.

The final stage of the preliminary design process was performance and stability analysis of the aircraft. The first task was to size the propulsion system for the aircraft. A commercially available electric model airplane performance program called Motocalc was used in this task. This program contains a wide database of performance data on commercially available motors, gearboxes, propellers, batteries, and speed controllers. Motocalc can give accurate estimates for the performance of any combination of these items. Only the Astroflight Cobalt 90 FAI with 1.64:1 gearbox provided the required static thrust, flight speed, current draw, and endurance that to complete the mission.

Matlab was then used to calculate the takeoff distance of the loaded aircraft to insure that the 200 ft takeoff distance requirement could be satisfied. Matlab was also used to size the horizontal tail surface to provide a 20% static margin. The vertical tail surface was sized by historical comparison with other model aircraft.

1.2. Detail Design

The primary task of the detail design stage was to design the aircraft structure. After choosing the various lay-ups for the primary aircraft components, accurate three-dimensional CAD drawings were needed in order to directly CNC machine the majority of the molds. Accurate CAD files were also necessary for the laser cutting service. The great majority of design time was spent creating these CAD files.

The secondary task was the selection of system components such as motor batteries, servos, and landing gear. Related to this task was the completion of a weights and balance worksheet, which indicated that 11-ounces of tail ballast would be necessary to balance the aircraft. Finally, a RAC worksheet was developed for the final design.

2. Management Summary

2.1 Architecture of the Design Team

The 2001-2001 DBF design team is organized under the leadership of Michael Cancienne, the Chief Project Engineer. John Freudenthal, Tony Fabiszak, and Robbi Jouben help share the burden of team leadership. The overall team hierarchy is shown in the Figure 2.1 below.

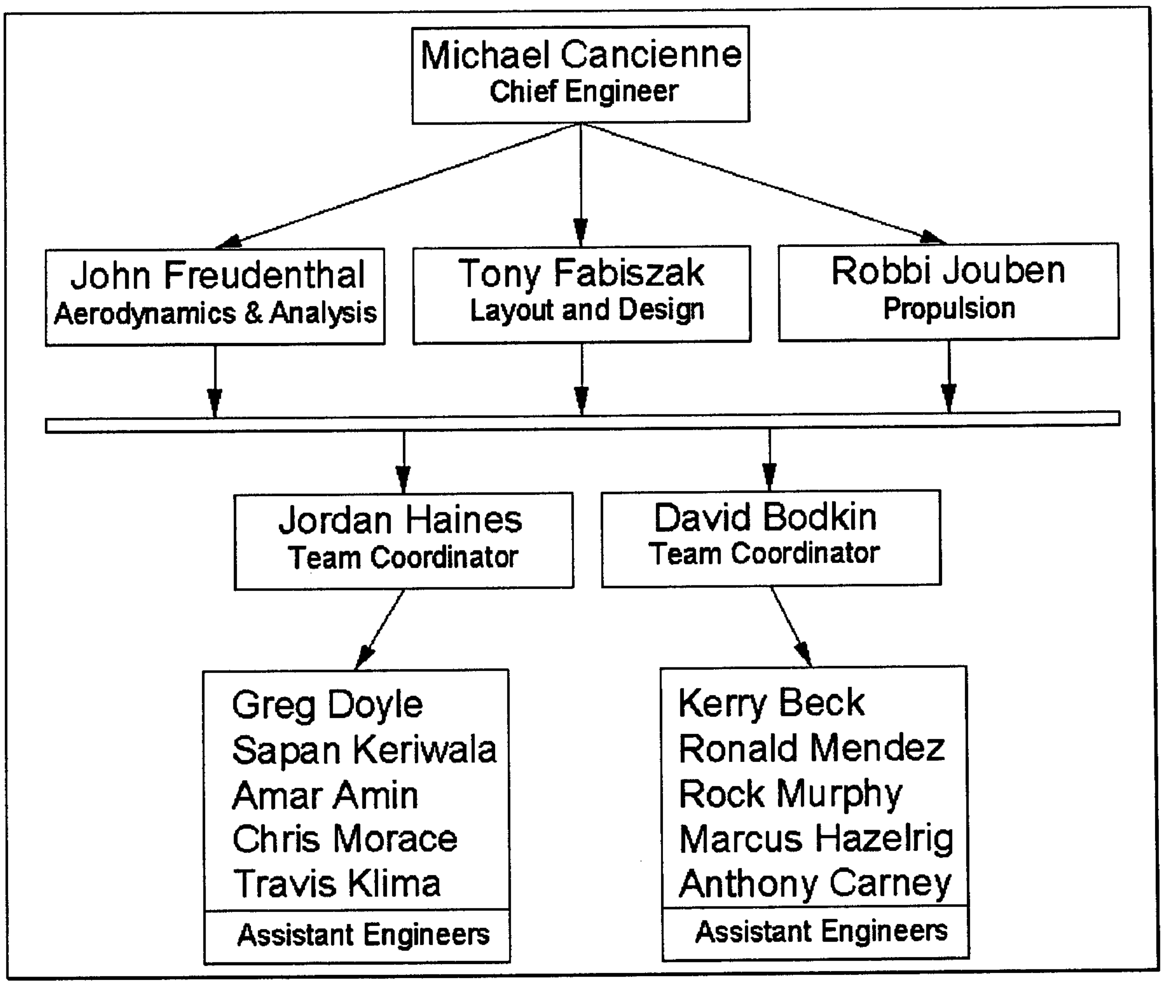


Figure 2.1. Team Architecture

2.2 Design Personnel and Assignment Areas

The design team personnel and their assignment areas are detailed in the list below.

Michael Cancienne
Chief Engineer, Pilot

John Freudenthal
Lead Aerodynamics & Analysis Engineer
Lead Flight Testing Engineer
Backup Pilot

Tony Fabiszak
Lead Layout & Design Engineer
Lead Manufacturing Engineer
Backup Pilot

Robbi Jouben
Team Spirit Coordinator
Chief Documentation and Organization Engineer
Conceptual Design Head

Jordan Haines
Team Coordinator
Lead Assistant Manufacturing Engineer

David Bodkin
Team Coordinator
Lead Assistant CAD Engineer

Assistant Engineers, assigned as needed
Kerry Beck
Greg Doyle
Sapan Keriwala
Amar Amin
Chris Morace
Travis Klima
Ronald Mendez
Rock Murphy
Marcus Hazelrig
Anthony Carney

2.3. Project Milestone Chart

The project milestone chart is shown in the table below.

Table 2.1. Project Milestone Chart

	August	September	October	November	December	January	February	March	April
Conceptual Design									
Preliminary Design									
Detailed Design									
Manufacturing									
Flight Testing									

3. Conceptual Design

The goal for the conceptual design phase was to determine the general arrangement of *Milk Run*. This process involved two Figure of Merit analyses. The first FOM analysis was performed on the mission flight profile itself to determine what design parameters were important to each mission segment. The second FOM analysis determined which of three alternative design concepts possessed the best combination of desirable qualities for mission completion.

3.1. Alternative Configuration Concepts.

The team initially identified three potential aircraft configurations: canard, conventional, and tailless delta. The team divided into three groups to develop the chosen configurations into conceptual designs.

3.1.1. Design Assumptions. Common rules for sizing of the three design concepts were established to insure that the concepts would be judged on their merits rather than on the more detailed design decisions of an individual team. It was assumed that the aircraft, no matter what the configuration, would weigh approximately 30-lbf at maximum gross weight. This estimation was based on experience and observation of previous DBF competitors. The wing loading was set to 44-oz/ft^2, which was found to be a typical wing loading for a 30-lbf sport-scale radio-controlled model airplane. It was felt that this class of model airplane was within the piloting skills of the team's three experienced modelers. The wing area of all three concepts was thus set to 1600-in^2.

For the purposes of conceptual design, the payload capacity was assumed to be 24 softballs. The flight score is directly proportional to the number of softballs and inversely proportional to the amount of time to complete a mission. If less than the maximum allowable payload were to be carried, then the plane would have to fly faster to achieve an equivalent score. It was reasoned that the lift requirement could be satisfied much easier than the high-speed requirement. Generation of lift is relatively easy to quantify and produce, as opposed to drag reduction, which is very difficult to accurately quantify or implement. Furthermore, undeterminable factors such as weather, piloting, course path, and the timing of pylon flags can reduce a flight score that is heavily dependent on a low mission time. Thus, placing all stakes on high speed, i.e. low drag and high power, was reasoned to be very much a gamble.

The maximum battery weight of 5-lbf was specified for the propulsion system. All of the conceptual configurations were assumed to be capable of completing the mission with the available battery capacity and power.

3.1.2. Canard. It seems that SOMEBODY always has to have a canard. The canard design group produced a configuration as shown in Figure 3.1. On a canard configuration, both the forward and aft wings produce useful lift. The main wing was sized to have 70% of the wing area, or 1231-in^2. The main wing was given a 100-in span with a 67% taper ratio, resulting in a root chord of 16.4-in. The canard was sized to 30% of the total wing area, or 370-in^2. A canard of this size is necessary to counter the pitching

moment of flaps on the main wing. The canard span was drawn to 50-in, with a root chord of 7.75-in. These physical dimensions are based on typical canard proportions gathered from other designs.

The length of the fuselage of the aircraft was estimated to be about 90-in. This estimate is based on a length of 48-in for 24 softballs seated 2-abreast, with additional length for propulsion system and streamlining. The 2-abreast seating was chosen to minimize frontal area. The aircraft has tricycle landing gear, as is necessary for a canard configuration. Also, the single engine is necessarily mounted in the pusher configuration for balance considerations. Tip sails provide directional stability for the aircraft.

The RAC score for the canard concept based on the estimated physical dimensions is 14.96. This is a point higher than the estimated conventional aircraft RAC. This is because the canard suffers from an extra wing in the manufacturing man-hours cost.

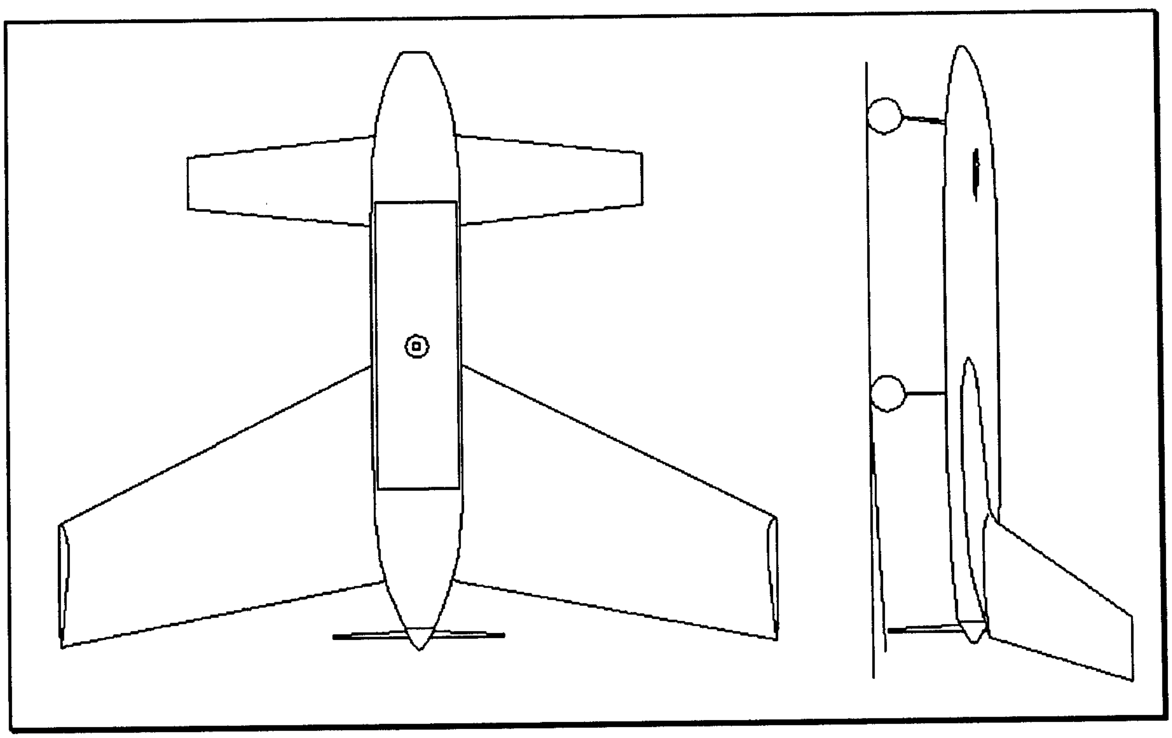


Figure 3.1. Canard conceptual design layout sketch. The rectangle in the top view represents the payload area. The small circle in the middle of the payload area represents the aircraft center of gravity location.

3.1.3. Conventional. The advantages of the conventional configuration are that it is familiar and it is a good all-around performer. The proposed conventional concept is illustrated in Figure 3.2. The fuselage was estimated to have a length of 90-in using the same rationale that was applied to the canard. The wing was given a 100-in span, as this gave the airplane a properly proportioned look. Assuming a taper ratio of 67%, the root chord was calculated to be 19.2-in. This produced an aspect ratio of 6.25. The single engine was mounted in a tractor configuration at the nose of the fuselage, which had tricycle landing gear arrangement. The RAC score of this conventional design concept was estimated to be 13.98.

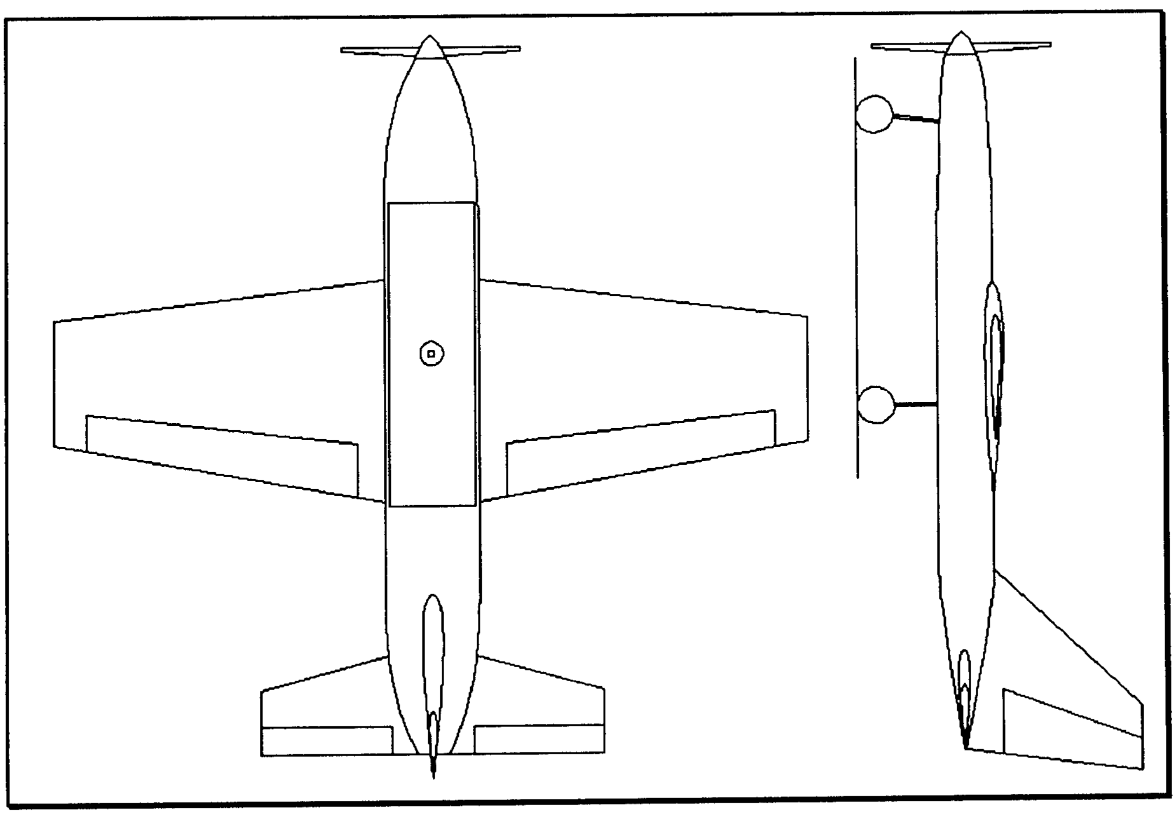


Figure 3.2. Conventional conceptual design layout sketch. The rectangle in the top view represents the payload area. The small circle in the middle of the payload area represents the aircraft center of gravity location.

3.1.4. Delta Wing. A tailless aircraft design always catches the fancy of a DBF designer, due to its potential for a low RAC score. This potential is most evident in the absence of horizontal tail structure and its associated pitch controls. Also, there is no need for "extra" fuselage length to create a sufficient moment arm for the horizontal tail; this reduces the fuselage cost. A less obvious RAC cost reduction is found in the low aspect ratio of the delta wing, which reduces the RAC cost for the wing, which is based on the sum of the maximum wing chord and the wingspan.

The delta wing concept is illustrated in Figure 3.3. The fuselage was sized at 70-in to make room for 12 softballs seated 2-abreast, plus additional fuselage length for the tractor motor and additional fuselage streamlining. A tricycle landing gear is necessary. A single high-aspect ratio fin is mounted on the rear of the fuselage for directional stability. The wing root chord runs almost the full length of the fuselage, with a chord of 60-in. The wingspan is 68-in. The RAC score for the delta wing concept is 13.49 -- about a half-point lower than the conventional concept.

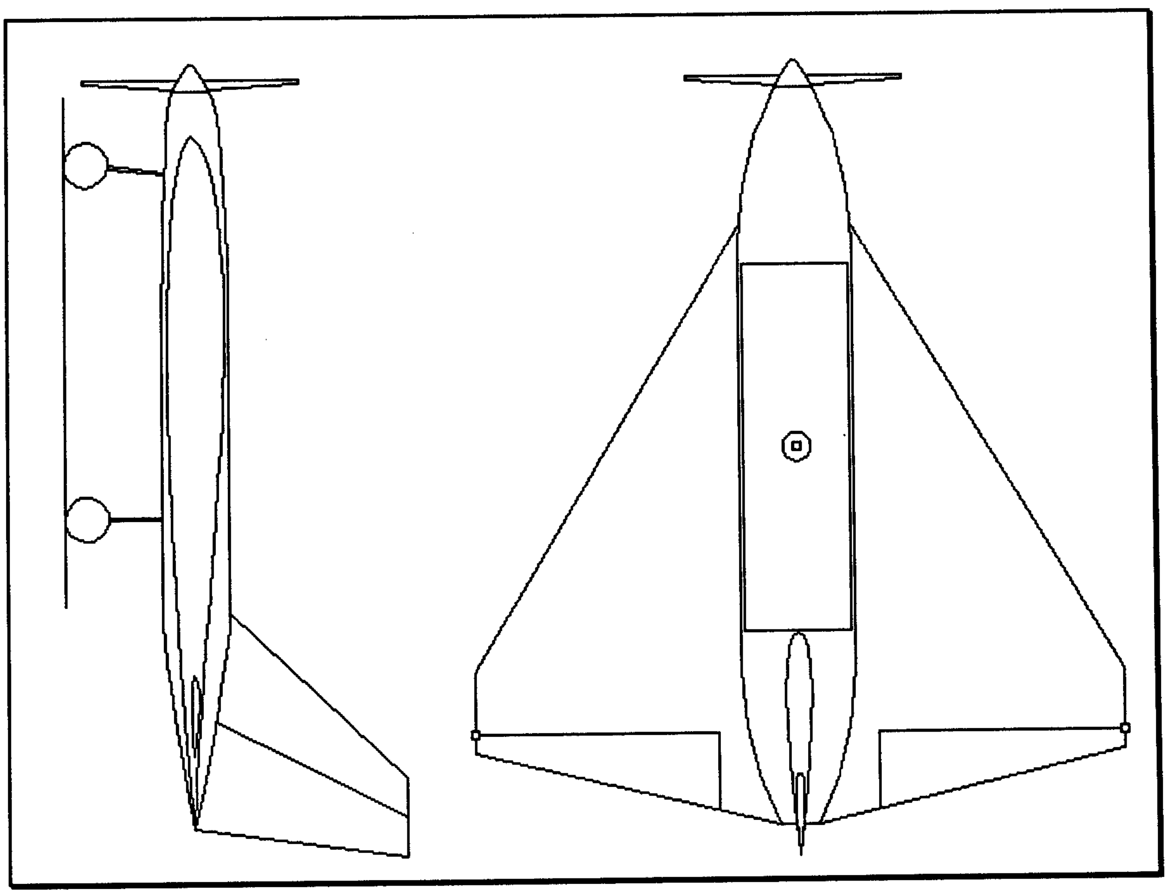


Figure 3.3. Delta wing conceptual design layout sketch. The rectangle in the top view represents the payload area. The small circle in the middle of the payload area represents the aircraft center of gravity location.

3.2. Mission Analysis: Figures of Merit

A single flight score is determined by 3 sorties. Each of the sorties consists of a combination of six maneuvers: takeoff, climb, turning, cruise, descent, and landing. These six maneuvers establish the performance FOMs for the mission analysis.

Takeoff and climb were determined to be the two most critical performance criteria, and were given a weighting of 3. The maximum allowable takeoff distance of 200-ft is the only specific performance requirement stipulated in the rules. Climb performance was deemed equally important because it is indicative of overall flight performance, and is a must for good recovery out of in-flight emergencies.

Cruise and turning performance were given a weighting of 2, since the bulk of the flight time is spent performing these maneuvers. Also, cruise performance must not impinge on turn or climb performance.

Descent and landing were not given any weighting. Although handling qualities must be adequate for easy landing, the landing performance is not a mission driver.

3.3. Mission Analysis: Generalized Design Parameters.

Fundamental aircraft characteristics were measured against the mission segment FOMs to determine what characteristics are most dominant in optimizing mission performance. The generalized design parameters considered in this FOM analysis are: propulsion configuration, wing area, aspect ratio, stability, drag reduction, and weight reduction. The results of the FOM analysis are tabulated in Table 3.1.

Table 3.2. Tabulated rankings of design parameter influence on mission performance FOMs.

		Figures of Merit						Totals
		Take Off	Climb	Turning	Cruise	Descent	Landing	
FOM Importance Factors:		3	2	2	2	1	1	
Design Parameters								
Propulsion Configuration	Pusher	-1	1	1	1	0	-1	2
	Tractor	2	2	2	2	0	2	20
Wing Area	High W/S	1	1	1	2	2	1	14
	Low W/S	2	2	1	1	1	2	17
Aspect Ratio	High AR (~10)	2	1	2	1	0	0	14
	Low AR (~4)	1	1	-1	1	0	0	5
Stability	Positive Static	2	2	2	2	2	2	22
	Neutral Static	-2	-2	1	-2	-2	-2	-16
Low Drag	(No Flaps)	2	2	2	2	1	1	20
Low Weight		2	2	0	0	1	2	13

3.3.1. Propulsion Configuration. Two propulsion configurations were considered: tractor and pusher propeller configurations. A tractor configuration was determined to be vastly superior to the pusher configuration for several reasons. Most importantly, the pusher propeller must operate in the wake of the aircraft. If the aircraft were going to be a high-performance laminar flow craft, the pusher propeller would be viable; unfortunately, the aircraft will likely have turbulent flow over most of the surface. Most commercially available model airplane propellers are optimized to operate in the flowfield of a tractor configuration. Furthermore, adequate cooling is concern in the pusher configuration. Also, proper location of the center of gravity is more difficult, since the motor and batteries are located in the rear of the aircraft. Another disadvantage of the pusher propeller is that it is prone to ground strikes during takeoff rotation and landing.

3.3.2. Wing Area (Wing Loading). The wing area of an aircraft is generally sized to a desired wing loading. Wing loading is a key driver in performance and handling qualities. A high wing area (corresponding to lower wing loading) is beneficial for takeoff, turning, and landing performance. A low wing area (or high wing loading) is beneficial for cruise because of the decreased skin friction drag. However, higher wing loadings raise the stall speed and make the aircraft more difficult to land. Because the low wing area may be supplemented with flaps for high lift maneuvers, there is no large advantage for either extreme. The FOM analysis provides inconclusive results in this area.

3.3.3. Aspect Ratio. High aspect ratio wings (aspect ratio of approximately 10) have lower induced drag than low aspect ratio wings (aspect ratio of approximately 4). For those mission features that require higher lift coefficients, such as takeoff, climb, and turn, the higher aspect ratio has the advantage. However, high aspect ratio wings are more difficult to construct due to the large moments and thinner cross sectional area. The FOM analysis suggests that the aspect ratio of the aircraft should be relatively high.

3.3.5. Static Stability. The FOM analysis determined that the aircraft should possess positive static stability. Neutral static stability would result in less trim drag; however, it can make the aircraft difficult to fly in windy conditions. This can overload the pilot, increasing the risk of a crash. Dynamic stability was not considered at this stage of the design process.

3.3.6. Low Drag. The relative importance of an exceptionally clean airframe is scored for each mission maneuver. The purpose of including this design parameter is for the relative weighting of clean airframe design to other design factors.

3.3.7. Low Weight. The relative importance of low aircraft weight is scored for each mission feature. Low weight is important in any aircraft design, but it is not necessarily the predominate factor. The purpose for including this factor as a design parameter is to gain insight into how much emphasis should be placed on weight saving when considering design options.

3.3.8 Mission Analysis Conclusions. The FOM analysis not only provided concrete decisions about certain design parameters, but it also gave insight into what design characteristics should be given priority when considering the overall configuration choice. The final ranking of significant design attributes using the FOM analysis is:

1. Positive static stability—22 points
2. Tractor propulsion configuration—20 points
3. Low drag—20 points
4. High aspect ratio—14 points
5. Low weight—13 points

3.4. Aircraft Configuration Selection: Figures of Merit

Since the team is well experienced with the conventional configuration, it is the benchmark by which all other configurations are judged. The five significant design attributes from the mission analysis are now used as figures of merit in a configuration selection process. The weighting of these five FOMs is roughly proportional to their ranking: Positive static stability, tractor propulsion, and low drag are given a weighting of 3; Low weight and high aspect ratio are given a weighting of 1. To Ground handling, cargo loading, and technology/manufacturability were also added as figures of merit. Good ground handling capability and efficient cargo loading can drastically reduce the turn-around time between sorties. The Technology/Manufacturability FOM is a measure of how difficult or risky it was thought to be to

incorporate the given configuration into the overall design. Since building and flying the aircraft well before the contest is essential to overall success, the design should not exceed the technical limits of the team. Thus the technology/manufacturability is weighted with 2. The results of the FOM analysis are tabulated in Table 3.2.

Table 3.3. Figure of Merit rankings for aircraft configuration selection.

	Figures of Merit								Totals
	Positive Stability	Tractor Propulsion	Low Drag	Low Weight	High Aspect Ratio	Cargo Loading	Technology / Manufacturability	Rated Aircraft Cost	
FOM Importance Factors:	3	3	3	1	1	1	2	2	
Alternative Configurations									
Canard	-1	-2	0	0	1	-1	-2	0	-13
Conventional	2	2	2	0	1	1	2	1	26
Delta Wing	1	2	1	2	-2	-1	1	2	17

3.4.1. Positive Static Stability. The conventional configuration was judged superior in this category, because it is well understood. Delta ranked one point less because it not as familiar a design concept. The canard was given the lowest possible ranking of –1 because stories abound about the short and violent lives of unstable original-design canards.

3.4.2. Tractor Propulsion. The canard must be a pusher configuration, so it ranks –2 in this category. The delta and the conventional configurations each rank +2.

3.4.3. Low Drag. The conventional aircraft scored highest in this category, at +2. The delta wing was ranked at +1 because its low aspect ratio wing produces a lot of induced drag for climb-out and turns. The canard configuration was ranked last, as it was felt the wake of the canard would adversely affect the wing drag.

3.4.4. Low Weight. The delta wing ranked highest in this category, due to its shorter fuselage and lack of horizontal tail. The low aspect ratio wing can also be built lighter than a longer aspect ratio wing. The canard and the conventional airplane are equivalently scored in this category.

3.4.5. High Aspect Ratio. The canard and conventional configurations can easily adjust their aspect ratios to a similar range of values; they are thus scored evenly. The delta wing is restricted to very small aspect ratios, resulting in a minimum scoring of –2 in this category.

3.4.6. Cargo Loading. The canard and the delta were scored –1 in this category. The two large wings of the canard and the wide root chord of the delta inhibit the movement of cargo handlers around the airplane. The conventional configuration was scored +1.

3.4.7. Manufacturability/Technology. Unsurprisingly, the conventional configuration scored +2 in this category. Most all textbooks and teaching is directed towards the design of the conventional configuration. The delta scored only slightly less at +1, because it is a simple and more common form of tailless design. The canard was ranked –2 because the team has heard too many bad stories about the canard's sensitivity to center of gravity location.

3.4.8. Rated Aircraft Cost. The three concepts were ranked from lowest to highest RAC score: delta (+2), conventional (+1), canard (0).

3.4.9. Configuration Selection. The FOM analysis overwhelmingly favored the conventional aircraft configuration, with a total FOM ranking of +26. The delta wing trailed nine points behind at +17. The canard never had a chance at –13. For the purposes of transitioning smoothly into the preliminary design phase, the conceptual design dimensions are summarized in Table 3.3.

Table 3.4. Conceptual Design Results

Conventional Configuration	
Empty Weight (lbf)	20.16
Maximum Weight (lbf)	30.00
Weight of Batteries (lbf)	5.00
Number of Softballs	24.00
Weight of Softballs (lbf)	9.84
Wing Loading (oz/ft^2)	44.00
Wing Area (in^2)	1600.00
Wing Span (in)	100.00
Aspect Ratio	6.25
Fuselage Length (in)	90.00
Rated Aircraft Cost ($1000)	13.98

4. Preliminary Design

The team's goal during the preliminary design phase was to determine the dimensional size of *Milk Run.* First, a simple numerical analysis was performed to determine whether 24 softballs was indeed the optimum cargo capacity. Next, an FOM analysis was performed to determine the specifics of the aircraft layout. Finally, the physical dimensions of the aircraft were calculated and a propulsion system was chosen.

4.1. Aircraft Payload Sizing

Rather than investigate numerous cargo capacities and power configurations to determine what arrangement would yield a maximum score, it was decided to ignore the difficulties of predicting the performance of numerous propulsion options and focus on cargo capacity alone. With some "back of the envelope" calculations, the physical sizes of aircraft that could carry 10, 18, and 24 softballs were estimated. The RAC was computed for each aircraft, and the necessary flight score for competitiveness was estimated.

Rules of thumb based on practical experience were followed in the sizing estimates. First, a weight estimate was made for each aircraft based on experience. Next, a wing loading was assigned to each airplane. A wing loading of 44-oz/ft^2 was assigned to the 24-ball aircraft, because this is a typical wing loading found on giant scale radio control model airplanes of the same size and weight. Specifically, the number is obtained for the advertised values of a Top Flite Giant Scale Corsair. This same principle was used to assign a wing loading of 29-oz/ft^2 to the 10-ball aircraft. The 18-ball aircraft was assigned a wing loading of 36-oz/ft^2, which is an interpolation of the wing loadings of the 10-ball and 24-ball aircraft.

The wing area of each aircraft was derived from the assigned wing loading and weight estimate. In order to establish some scaling equivalence, the 10- and 18-ball aircraft were assigned the same aspect ratio as the conceptual design 24-ball aircraft. The wingspan of each aircraft could then be calculated. Next, a taper ratio of 67% was assumed for each aircraft, which allowed an estimation of the wing root chord. The fuselage lengths of each aircraft were set at 90% of the wingspan. This length/span ratio is based from the conceptual design result.

These estimates allowed the estimation of a RAC score for each aircraft. It was assumed that all three aircraft possessed a single engine, a conventional tail, 5-lbf batteries, 6 controllers (2 ailerons, elevator, rudder, throttle, retracts), and a horizontal tail span less than 25% of the wingspan. Assuming that each plane was capable of finishing the six-lap course, the competitive time ratio was calculated for each plane. This ratio indicates the fraction of time a smaller capacity aircraft would have to finish the mission in order to tie with the 24-ball aircraft. Thus, the 24-ball aircraft has a competitive time ratio of 1, and the smaller planes have a fraction of that. Competitive flight speeds were also estimated by assuming a flight speed for the 24-ball aircraft, and calculating the flight speed required of the smaller aircraft to have an equivalent total score. The results of these estimations are tabulated in Table 4.1.

Table 4.5. Sizing estimates for determining optimum payload capacity. Yellow fields indicate common factors used in the sizing process.

Number of Softballs	24	18	10
Weight of Softballs (lbf)	9.84	7.38	4.1
Total Weight (lbf)	30	24.38	18.1
Empty Weight (lbf)	20.16	17	14
Wing Loading (oz/ft^2)	44	36	29
Wing Area (in^2)	1600	1560	1438
Aspect Ratio	6.25	6.25	6.25
Wing Span (in)	100	98.75	94.8
Wing Root Chord (in)	19.2	18.96	18.2
Fuselage Length (in)	90	88.875	85.32
Length/Span Ratio	0.9	0.9	0.9
RAC ($1000)	13.93	13.57	13.15
Flight Score (t is mission time)	30/t	24/t	16/t
Competitive Time Ratio	1	0.825	0.572
Competitive Flight Speed (mph)	60	73	105
Competitive Flight Speed (mph)	70	85	122

Surprisingly, the physical dimensions of the 10- and 18-ball aircraft are very close to the 24-ball aircraft. This is largely due to the lighter wing loadings imposed on these airplanes. This kills any RAC score advantages the smaller airplanes might have had over the 24-ball aircraft. The 100+ mph speed requirement for the 10-ball aircraft was deemed unfeasible with the rules-specified brushed motor technology. The results for the 18-ball aircraft indicate that it may be competitive with the 24-ball aircraft. However, continuing with the reasoning that it is much easier to generate lift than to reduce drag, the team set the design capacity of the final aircraft at 24 softballs.

4.2. FOM Used, Mission Features Supported

Before discussing preliminary design parameters and FOM analysis, it is important to note that at this stage molded composite construction was chosen for the aircraft. The final determination of construction method enabled judicious choices to be made in the preliminary design. The rationale behind this decision is explained in Section 5, Manufacturing Plan and Processes.

4.2.1. Preliminary Design Figures of Merit. The goal of the preliminary design phase FOM analysis was to determine what specific design features would be found on the aircraft. The design parameters are wing planform, wing position, aileron span, tail configuration, tail control surfaces, and landing gear. The FOMs for this analysis were chosen to be Mission Performance, Manufacturability/Technology, and Rated Aircraft Cost. Because the team's technical limit was already being pushed by the molded construction, Manufacturability/Technology category was double-weighted to inhibit risky design features. The ranking of design parameters in these FOM categories is shown in Table 4.2.

Table 4.6. Preliminary design FOM analysis of specific design features.

		Figures of Merit			Totals
		Performance	Manufacturability/Technology	Rated Aircraft Cost	
FOM Importance Factors:		1	2	1	
Design Parameters					
Wing Planform	Elliptical	2	0	0	2
	Rectangular	-2	2	0	2
	Straight Taper	0	1	0	2
Wing Position	Shoulder	0	2	0	4
	Mid	2	-2	0	-2
	Low	1	2	0	5
Ailerons	Full Span	-1	1	1	2
	Semi Span	1	-1	-1	-2
Tail Configuration	Conventional	1	2	0	5
	T-tail	2	-1	0	0
	V-tail	2	1	0	4
Tail Control Surfaces	Hinged Elevator	-1	1	1	2
	Flying Elevator	1	-1	-1	-2
Landing Gear	Conventional	1	1	1	4
	Tricycle	-1	-1	-1	-4
	Fixed	-1	1	1	2
	Retractable	1	-1	-1	-2

4.2.2. Wing Planform. Three different wing planforms were considered: elliptical, rectangular, and straight taper. Elliptical wings are efficient and aesthetically pleasing, yet are difficult to build. Straight tapered wings are more practical to build, and may be built to the same level of efficiency as the elliptical wing. Both elliptical and highly tapered wings have a bad tendency to tip stall during low speed maneuvers. Rectangular wings are easy to build, have forgiving flight qualities (no tip stall), but are relatively inefficient when compared to tapered or elliptical planforms. The RAC score would not be affected by any choice. The planform FOM analysis was inconclusive, since all three planforms had a total score of +2. However, the team had already decided on molded composite construction and CNC

machined molds. Elliptical wing molds could be shaped as easily as any other planform. The team decided on an elliptical planform with modified tips to discourage tip stalls.

4.2.3. Wing Position. Three general locations of the wing were considered: shoulder mounted wing, mid-fuselage mounted wing, and low mounted wing. A mid-fuselage mounted wing is the textbook ideal for low drag aircraft, but is more difficult to build. Shoulder-mounted and bottom-mounted wings are easier to design and build. The low wing mounting was chosen to move the wing wake away from the fuselage and tail, and to provide a lower stance for wing mounted main landing gear.

4.2.4. Aileron Span. Full span or semi-span aileron control surfaces were considered. If full span ailerons were used, they would also be used as flaps (flaperons). If semi-span ailerons were used, separate flap surfaces would be required. Separate flaps would increase RAC score, but would allow for more control configurations such as "crow" mixing. Crow mixing, which reflexes the outboard ailerons and droops the inboard flaps, allows for greater glide scope control during landing. The experienced modelers on the team decided that that the value of crow mixing capability was not substantial enough to justify the increased cost. For this reason, full span aileron control surfaces were chosen.

4.2.5. Tail Configuration. Three different tail arrangements were discussed: conventional tail, T-tail, and V-tail. The conventional tail is easy to configure, align, and build. The T-tail suffers from complex structure required to support it, and also more complex control actuation. The V-tail is simple to build, but research into V-tail sizing produced conflicting methods and results. The V-tail was deemed to be an unnecessary technological risk. The conventional tail arrangement was chosen.

4.2.6. Horizontal Tail Control. The advantages of full "flying" stabilator or hinged elevator were compared. A stabilator was initially favored for its low trim drag potential, but the difficulties of sealing the gap between the fuselage and the stabilator were deemed too tedious to make the effort worthwhile. Also, the stabilator stalls at lower angles of attack than the hinged elevator, effecting flight performance and handling qualities during extreme maneuvers. The hinged elevator was chosen because it is simple to produce and has a higher range of effective angle of attack.

4.2.7. Landing Gear. Conventional (tail-dragger) gear was chosen because it has less weight and less drag than the tricycle gear. Also, the short-field performance of conventional gear is better than that of tricycle gear. The ground handling quality of conventional gear is more difficult than that of tricycle gear. Because there are several pilots on the team with RC experience with conventional gear, it was decided that the benefits of the conventional gear were substantial to overall performance.

Although retracts might have added to the top speed of the aircraft, they scored low in the FOM analysis due to the increased rated aircraft cost and technical complexity. No team members had any experience with retractable landing gear, which are notorious for causing problems if not properly installed and maintained.

4.3. Aircraft Sizing

The initial sizing estimation was done only to determine the target payload capacity. After the FOM was completed, a more thorough work was performed in determining the necessary power, wing, and tail arrangement to produce a flyable design.

4.3.1. Power Loading. Since the motor selection is limited to the brushed motors from Astroflight and Graupner, it is easier to size an airplane for a selected motor than to size a motor for a power requirement. Graupner only makes very small electric motors, so their product line was quickly removed from consideration. The performance of the Astroflight product line was estimated with Motocalc. Motocalc is a commercially available program that includes an extensive database of motors, propellers, batteries and speed controllers. The dimensions of the 24-ball airplane were entered into the program, along with physical descriptions such as surface finish, etc, to provide a rough performance estimate of the airframe/propulsion combination. A number of motor/propeller combinations were analyzed, including twin motor configurations. The Astroflight FAI Cobalt 90 with a 1.64:1 gearbox was found to be an adequate single-motor propulsion system for a 30-lbf aircraft. Relevant motor performance data for this configuration is included in Table 4.3.

Table 4.7. Motocalc data for Astroflight 90 FAI Cobalt Motor, 1.64:1 Gearbox, 36 SR2400 mah cells.

	Static Performance							Performance at 60 mph			
Propeller	Current (Amps)	Input Power (W)	Electrical Efficiency (%)	Input Power Loading (W/lbf)	Output Power Loading (W/lbf)	Prop RPM	Thrust (lbf)	Current (Amps)	Electrical Efficiency (%)	Thrust (lbf)	Time (m:s)
19x14	34.7	1364.8	70.3	45.5	35.1	5202	11.87	26.8	74.1	5.35	5:22
20x13	36.8	1440.8	68.4	48	36.2	5034	13.18	20.4	75	4	7:04

Note that the static and in-flight current draw is less than the 40-amp fuse limit specified by the DBF regulations. Another quantity of interest is the input power loading, which is about 46-W/lbf. A common rule of thumb among electric modelers is that an input power loading of 55-W/lbf is necessary for an aerobatic airplane, and a 35-W/lbf input power loading is necessary for mild aerobatic performance. Because the input power loading for *Milk Run* falls in between 35- and 55-W/lbf, satisfactory flight performance is expected. The last quantity of interest is the endurance at 60-mph. A rough estimate of the flight distance of one lap of the course is one mile. Since six laps are required in one mission, roughly six minutes are required to finish the mission at an average flight speed of 60-mph. The endurance of the 19x14 propeller at 60-mph estimated to be 5:22 minutes, and the endurance of the 20x13 propeller is estimated to be 7:04 minutes. None of these numbers are definite; actually, other modelers have reported that Motocalc results tend to be conservative. Nevertheless, it was decided that a number of

propellers near the 20x13 dimension would be purchased and tested to discover the best overall performance. Endurance will also be slightly improved by using 2500-mah batteries.

4.3.2. Wing Loading and Aspect Ratio. The maximum-weight wing loading was retained at 44-oz/ft^2 and 1600-in^2 of wing area. It was estimated that a 30-lbf aircraft with 1600-in^2 of wing area could take off in 112-ft, assuming a constant C_L of 0.8 and 13-lbf of static thrust. However, the mission FOM analysis (Section 3.3) indicated that high aspect ratio was beneficial to mission performance. Increasing the span from 100-in to 120-in increased the aspect ratio of the aircraft from 6.25 to 9.

4.3.3. Horizontal and Vertical Tail Sizing. The static stability of an aircraft is a product of the tail moment arm and the area of the horizontal tail. Using the conceptual design drawing as a basis, the tail moment arm of the horizontal stabilizer was estimated to be 45-in. Assuming that the tail moment arm is fixed, the stabilizer area may be calculated for a desired static margin. In order to make these calculations, the tail span was sized "by eye" to fit the basic proportions of most conventional airplanes. Figure 4.1 illustrates static margin as a linear function of the horizontal tail area. A horizontal tail area of 300-in^2 was chosen for a static margin of 19%. This roughly represents the static margin for a typical sport-class model aircraft.

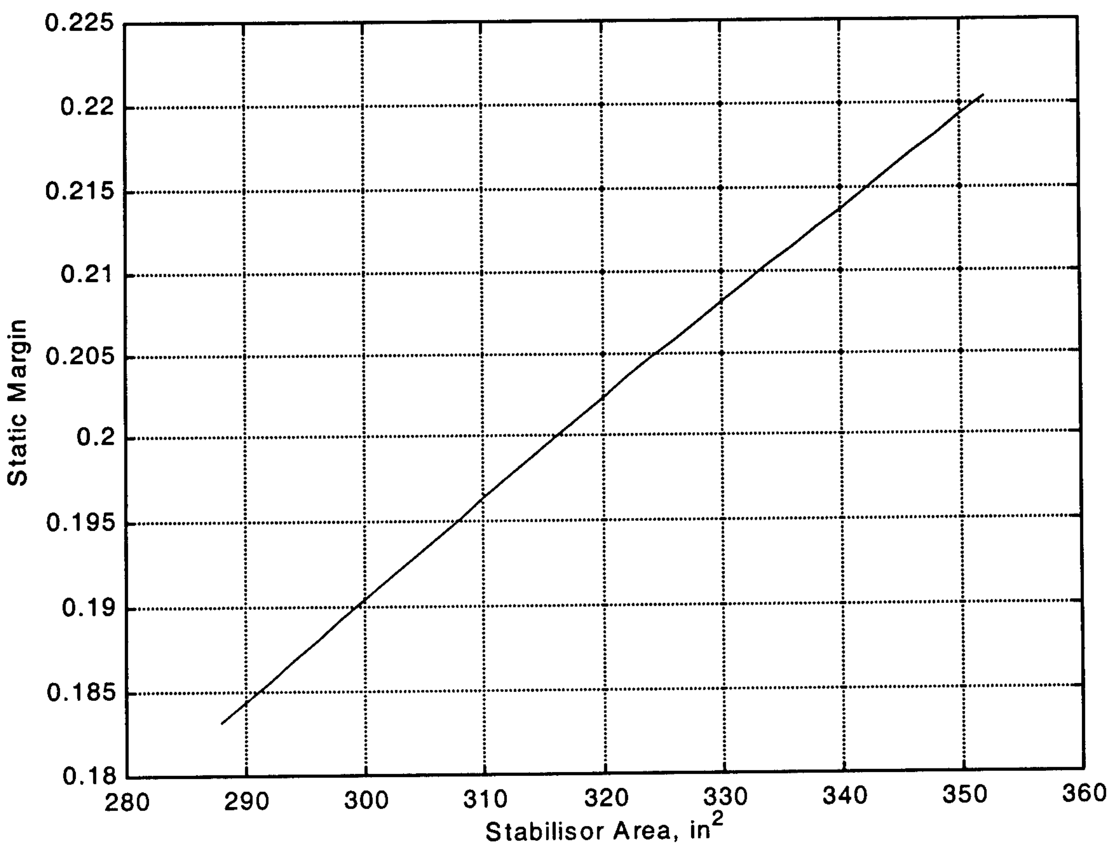


Figure 4.2. Static margin vs. stabilizer area for 45-in tail moment arm and 39-in tail span.

No computational methods were found to size the vertical tail. Historical data from various large scale model airplanes and last year's DBF aircraft were used to size a vertical tail area of 191.1-in^2. This corresponds to 12% of the wing area; a typical fin area for model airplanes is in the range of 8-11%.

4.4. Comparison of Assumptions with Conceptual Phase

In the conceptual design phase, it was assumed that 24 softballs would be carried since it is easier to double lift than to halve drag or double the power available. This assumption was further verified in preliminary design, once scaling of wing loadings came into influence and the RAC scores did not change substantially between the different payload capacities.

Another substantial change from conceptual design was the RAC scoring of the horizontal tail. The 39-in span of the horizontal tail qualifies it as a wing in the RAC work breakdown structure, which stipulates that any horizontal surface with a span greater than 25% of the largest wingspan must be scored as a wing. During conceptual design, this had been overlooked. It had been assumed that the horizontal tail would be scored under the empennage work breakdown structure. This design change increased the manufacturing man-hours for the horizontal tail from 10 hours to 38.7 hours.

5. Detail Design

The primary goal of the detail design phase was to design a strong but lightweight structure for the aircraft. This was accomplished by using molded composite construction. The secondary task was the final selection of system components.

5.1. Structural Design

In choosing the limit load, the goal was to produce a structure that would be very difficult to overload in flight, even with violent maneuvering. Because simple analysis techniques were used in structural sizing, a high factor of safety was chosen. The overall gain of having an aircraft survive flight testing and successfully fly in the competition outweighs the small weight penalty incurred.

The flight limit loads were chosen to be:

- Limit Load of 7.0 g's
- Ultimate Load of 10.5 g's

5.1.1. Wing Design. The wing design stems from an effort to maximize efficiency. A modified elliptical planform was chosen to obtain an efficient lift distribution and to minimize induced drag. The planform is not a true ellipse, as the tip chord is wider than that of a pure elliptical planform. The lengthened tip chord reduces the occurrence of tip stalls at low flight speeds. The airfoil is a SD-7080, which was chosen for its low drag coefficient. The hinge lines of the control surfaces fall along the major axes of the ellipses that define the leading and trailing edges of the wing. This allows the full span flaperon hinge lines to remain at a constant chord position.

The up-swept wing tip design is copied from high performance sailplanes. As the span-wise flow on the lower surface moves out toward the wing tip, it ramps upward and accelerates, thereby lowering the pressure. The span-wise flow on the top surface of the wing is decelerated and the pressure rises. This theoretically reduces the strength of the tip vortex. In reality, the performance gain is probably very small. The tip was used because it has a beautiful appearance and improves the overall look of the aircraft without any additional building effort (due to the CNC milling of the molds).

5.1.2. Overview of Wing Structure. *Milk Run's* wing structure consists of a sandwich skin structure with internal ribs and shear webs as shown in the Figure 5.1. The top and bottom skins are comprised of a ±45° bias fiberglass layer and a span-wise uni-directional layer of carbon fiber on both sides of a thin Rohacell foam core. The outer ribs are made of medium density balsa. The ribs in the center section bear high loads and are made from 1/8-inch plywood. The ribs are laser cut by computer, which is very accurate and allows for judicious use of lightening holes without increased labor time.

The shear webs consist of a balsa core wrapped in a ±45° bias carbon cloth. The shear web running along the 25% chord line is the load-carrying shear web. The shear web in the control surface area provides a closed section for the flaperons, increasing their torsional stiffness.

Common to all of *Milk Run's* control surfaces (except the rudder) is the use of an integral skin hinge. The hinge is a single layer of 2.0 oz Kevlar tape in the skin running along the length of the hinge line. A gap is placed in the core along the hinge line so that the skin is free to flex when the on the

opposite surface skin is cut, also along the hinge line. The excellent fatigue and toughness properties of Kevlar prevents the skin from cracking and separating along the hinge. This method has been widely used in model sailplanes and has proven to produce a strong and reliable hinge.

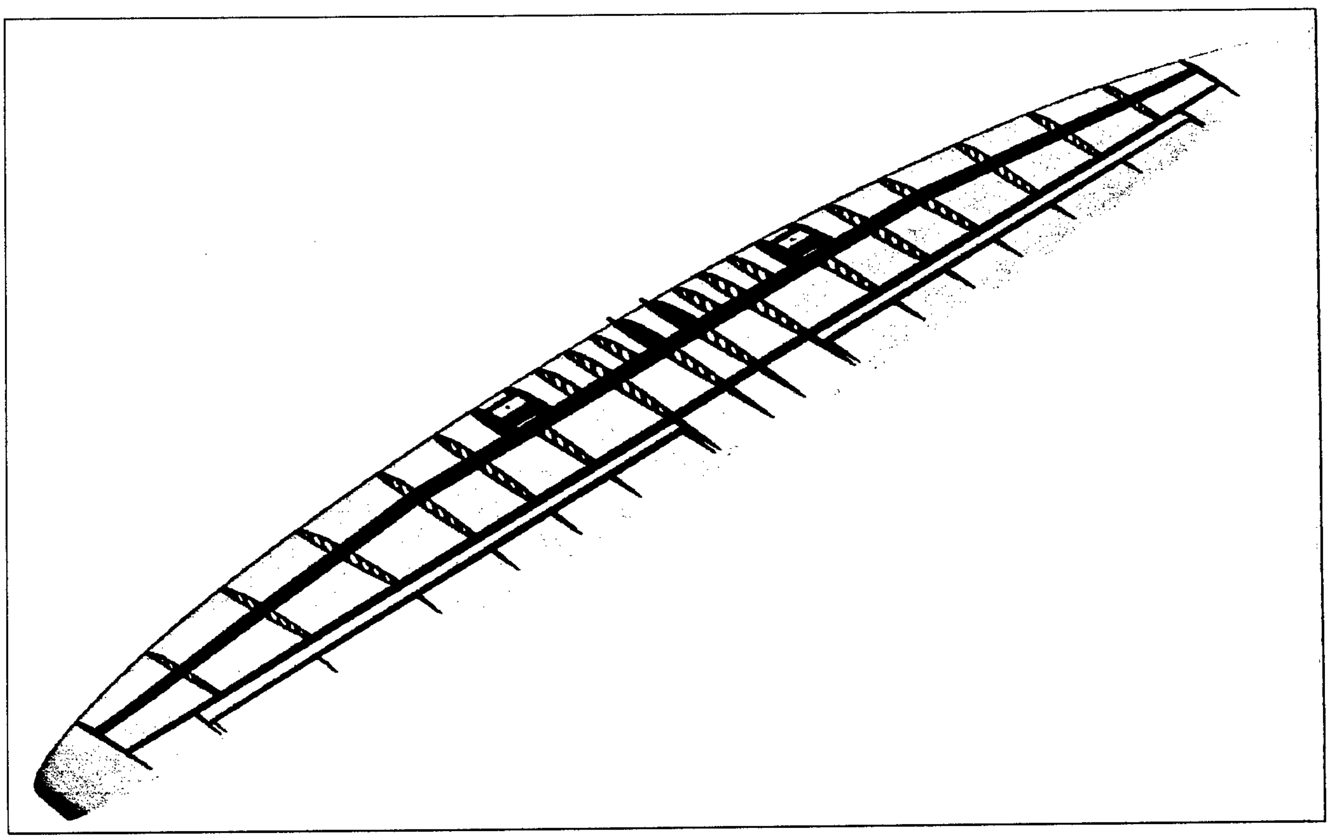


Figure 5.1. ***Milk Run*** **internal wing structure. Tan ribs are balsa, red ribs are plywood.**

In the Figure 5.2, the detail of the landing gear structure is shown. The gear strut is bonded into an aluminum block that is supported by the wing ribs. These wing ribs are reinforced with carbon cloth and transmit the loads from the landing gear into the wing skin and main shear web. The landing gear mount was designed to be extremely robust in order to survive possible off-runway ventures.

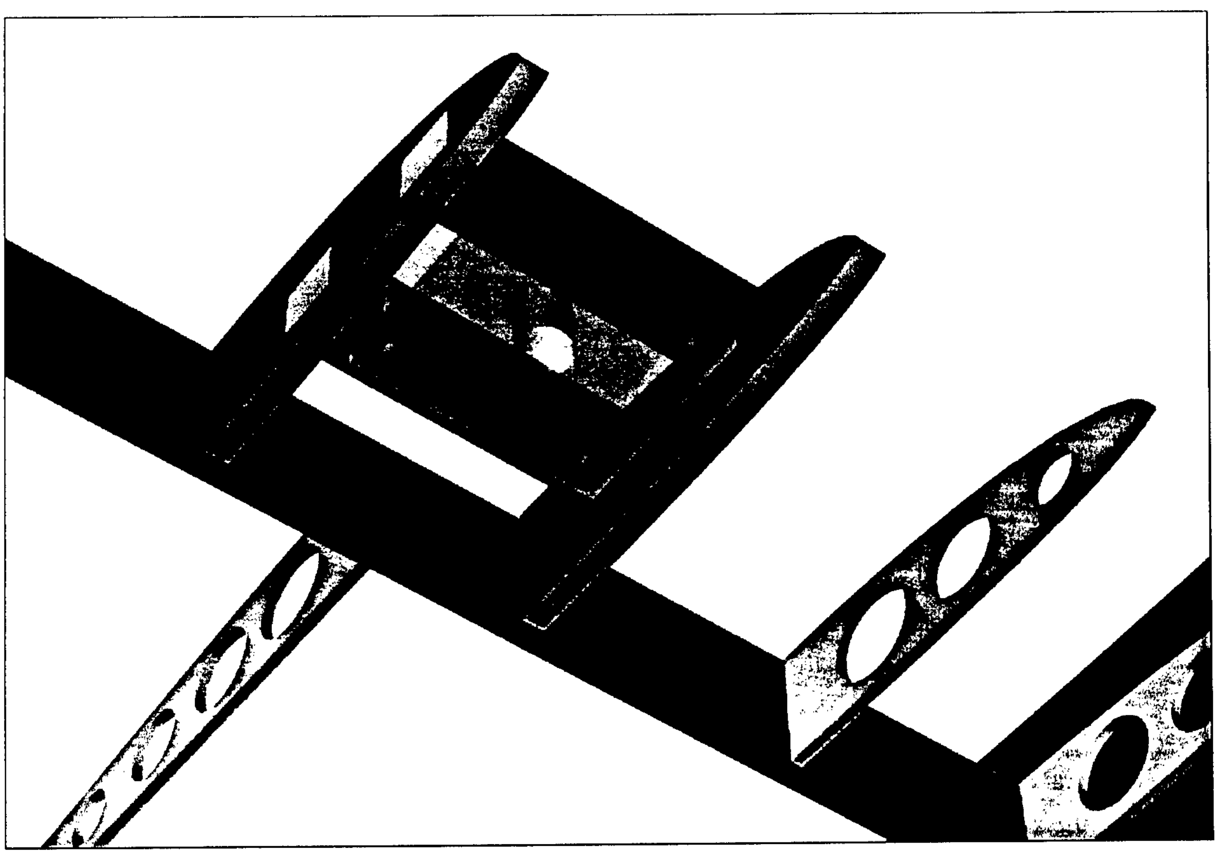


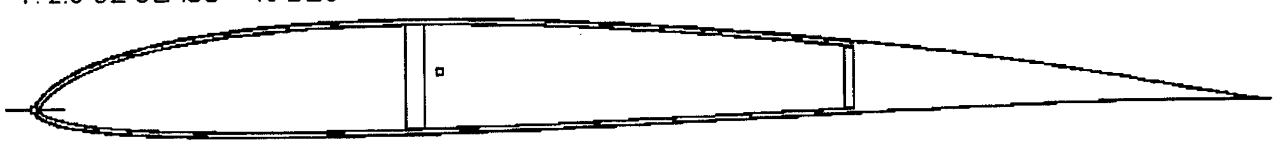
Figure 5.2. The landing gear mount and associated structure. Note that the ribs supporting the gear mount rails are reinforced with carbon fabric.

5.1.3. Wing Structural Analysis. To simplify the structural analysis, the skin was assumed to carry all bending and torsion loads, and the shear webs were assumed to carry all shear loads. Within the wing skins, the uni-directional layers of carbon fiber were assumed to carry all of the bending stresses generated, and the +-45° fiberglass layers were assumed to carry any torsion loads generated. For the shear webs, the bias carbon is assumed to carry the entire shear load.

The wing skin is modeled as a simple cantilever beam with an elliptically distributed load equal to the maximum gross weight multiplied by the ultimate load factor. Using the maximum compressive strength of the carbon composite as the stress limit for the upper surface and the maximum tensile strength for the stress limit for the lower surface, the required thickness of the carbon in the wing skin was estimated.

The bias fiberglass for the skin was sized at first for torsional stiffness, but it was found that this yielded a delicate structure. To make the structure capable of withstanding normal handling and abuse, the skin was made from 2.0-oz/yd^2 fiberglass. A layer of glass tissue is applied to the outer surface to provide a smooth finish. Figure 5.3 details the final sizing of the wing skins and the applied and allowable stresses. The wing structure is documented in more detail in the drawing package.

UPPER SURFACE LAYUP:
1. GLASS TISSUE
2. 2.0 OZ GLASS +-45 DEG
3. SMOOTH WEB 0.004" UNI-DIRECTIONAL CARBON
4. GAP WEB 0.005" UNI-DIRECTIONAL CARBON
5. 0.059" ROHACELL CORE
6. GAP WEB 0.005" UNI-DIRECTIONAL CARBON
7. 2.0 OZ GLASS +-45 DEG



LOWER SURFACE LAYUP:
1. GLASS TISSUE
2. 2.0 OZ GLASS +-45 DEG
3. SMOOTH WEB 0.004" UNI-DIRECTIONAL CARBON
4. 0.059" ROHACELL CORE
6. GAP WEB 0.005" UNI-DIRECTIONAL CARBON
5. 2.0 OZ GLASS +-45 DEG

MOMENT OF INERTIA OF CARBON ALONE: 0.1026 IN^4

	psi
Maximum Compressive Strength	33000
Applied Compressive Stress	32099
Maximum Tensile Strength	55000
Applied Tensile Stress	45847

Figure 5.3. Final sizing details of the wing skin structure.

5.1.4 Horizontal Tail Design and Structure Overview. The planform of the horizontal tail was chosen to match that of the wing. The major axis of the leading and trailing edge ellipses fall at the 60% chord line so that the 40% elevator is hinged along a constant chord position. A fully symmetrical NACA 0009 airfoil was chosen for the tail airfoil. This airfoil is commonly used for tail surfaces.

The horizontal tail structure is very similar to the wing. The skin consists of a Rohacell core with bias fiberglass on each side; but unlike the wing, uni-directional carbon is only placed on the inside of the skin. The shear webs and ribs in the horizontal tail are laser cut from 1/8-inch balsa. The shear webs are reinforced with carbon sock material. Like the wing control surfaces, a Kevlar skin hinge is used for the elevator. An aluminum carry-through structure is employed since the fuselage splits the horizontal tail. The split symmetry of the horizontal tail required a mold for only one half of the horizontal tail skin. These details are illustrated in Figure 5.4.

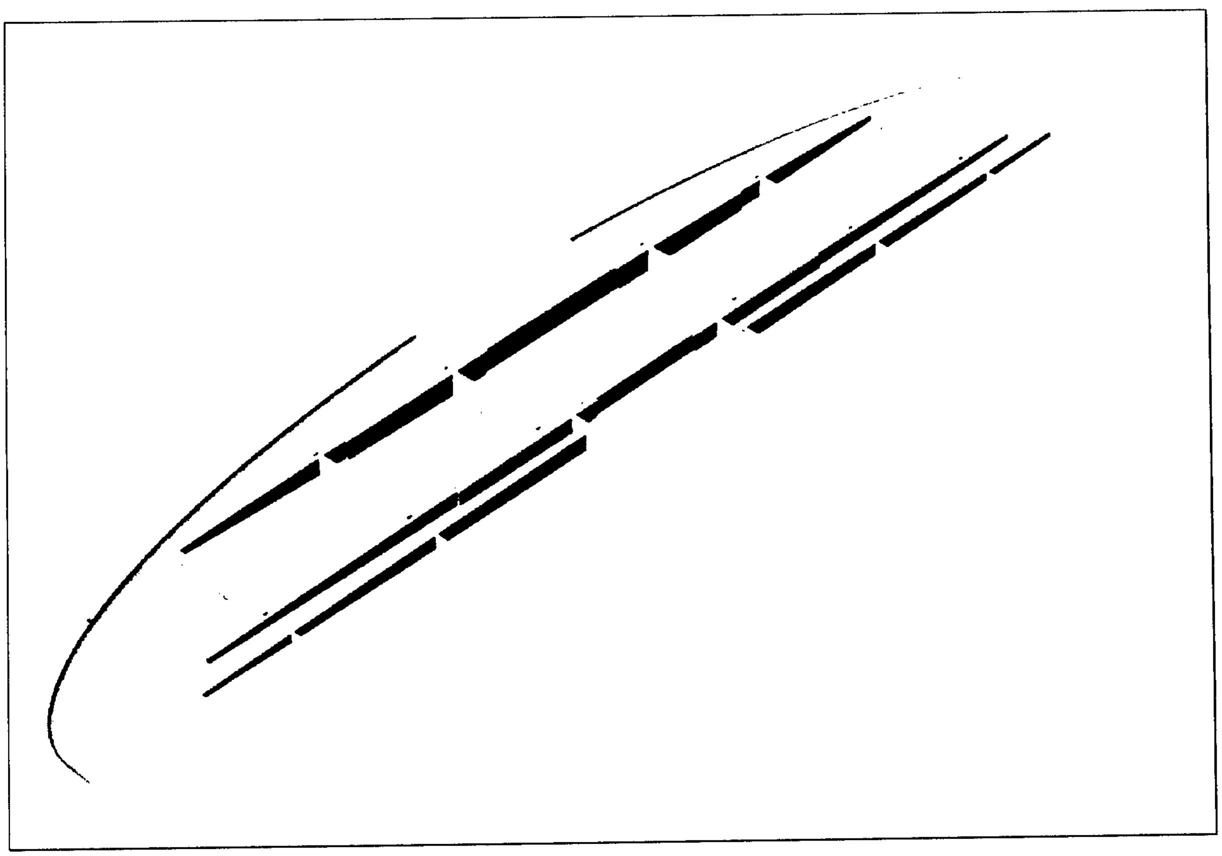


Figure 5.4. Horizontal Tail Structure. The horizontal tail structure is seen in the figure above. Note the aluminum carry-through structure in the center of the tail.

5.1.5. Horizontal Tail Structural Analysis. The horizontal tail is sized for a hard pull-up or push-over at maneuver speed, since the elevator is just as likely to be deflected down as up. The primary load generated on the horizontal tail is the aerodynamic load created by elevator deflection. To estimate the maximum load on the horizontal tail, it was assumed that the tail surface was capable of generating a maximum lift coefficient of 1.6 with full elevator deflection. The resulting aerodynamic load at a maneuver speed of 100-mph was found to be approximately 87-lbf.

As in the wing structure, the unidirectional layers of carbon were assumed to carry all bending stress generated, and the bias fiberglass layers were assumed to carry all of the torsion loads generated. The thickness of the carbon fiber layers were sized by using the maximum stress generated in bending and the maximum compressive strength of the carbon composite. As in the wing, the glass bias layers were not sized by the flight loads, but by flight crew handling requirements.

5.1.6. Vertical Tail Design and Structure. The shape of the vertical tail was chosen to match that of the wing and horizontal tail. The vertical tail and rudder share the same structure as the wing and horizontal tail. The skins consist of a Rohacell core with bias fiberglass and unidirectional carbon fiber on either side. The vertical tail post is made from 1/8" plywood and the ribs are made from 1/8" balsa. The vertical fin structure is illustrated in Figure 5.5.

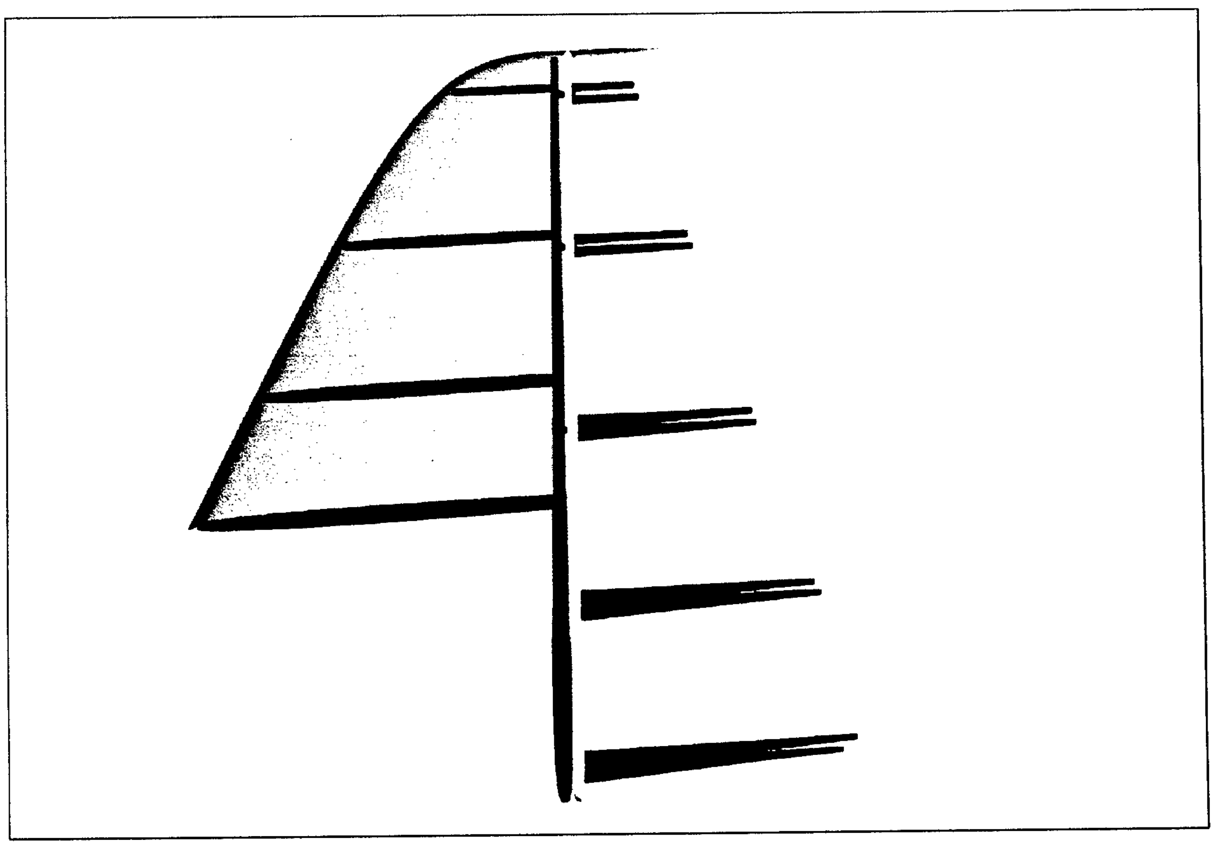


Figure 5.5. Vertical tail structure.

Unlike the other control surfaces, the rudder utilizes a piano hinge. Although a Kevlar skin hinge would allow for relatively free movement, the design team decided that it would not be sufficient for mounting the rudder. To ensure that the vertical tail would not fail in the event of a flip-over, it was made slightly more robust than would be necessary to handle the imposed flight loads. This was done at the expense of a small weight penalty. A frail aircraft is not beneficial to the overall success of the program.

5.1.7. Fuselage Structure. The design of the fuselage was largely influenced by the payload requirements. The final design for the cargo loading system consists of a removable hatch on the top forward fuselage. Placing the hatch in the forward fuselage rather than farther aft prevents ground crew members from having to walk between the wing and the tail. This hastens the cargo transfer process and reduces the likelihood of the aircraft being damaged during the cargo transfer operation. An illustration of the fuselage hatch is shown in Figure 5.6. The cargo is restrained in all directions by balsa longitudinal stringers. Solid bulkheads at the front and rear of the payload bay keep the balls from rolling fore and aft. These bulkheads are documented in more detail in the drawing package.

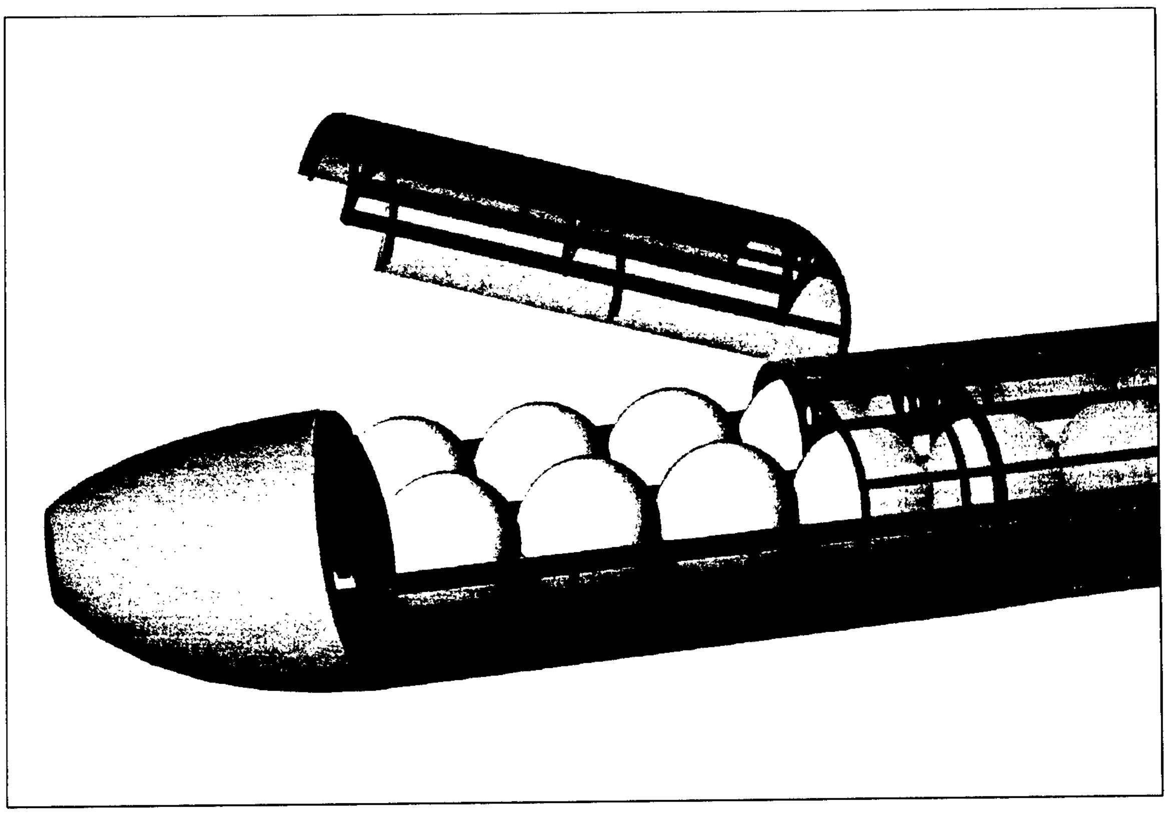


Figure 5.6. Fuselage structure detail. The hatch is shown in a partially open position. Note the longitudinal stringers (colored blue) that support the balls both vertically and horizontally.

The fuselage portion aft of the payload bay was designed for ease of manufacturing and final assembly. The horizontal tail centerline is located along the fuselage split line so that the fuselage mold has no trapped fairings. The rearmost bulkheads have laser-cut slots for the horizontal tail carry-through beams. These slots allow for rapid and accurate positioning of the horizontal tail. Figure 5.7 illustrates the keyed structure.

No structural analysis was performed on the fuselage. Experience has shown that a skin consisting of a core with one 0-90° layer and one ±45° layer of 2.0-ounce fiberglass on each side is sufficient for fuselage strength and rigidity.

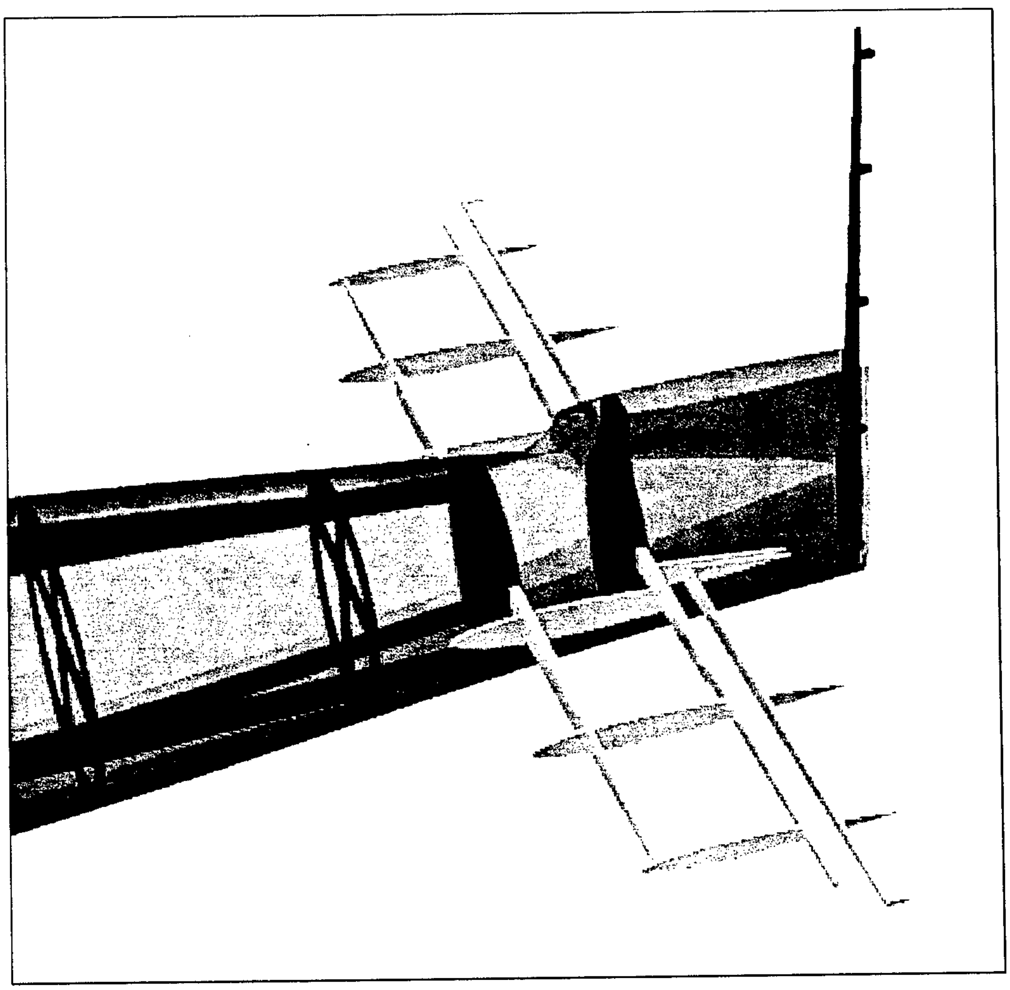


Figure 5.7. Tail structure of the fuselage. The completed horizontal tail (skin not shown for clarity) keys into the aft fuselage structure.

5.2. Weights and Balance

A CAD modeling package was used to determine the locations of the centers of mass of all the aircraft's internal and external parts, including propulsion and flight control systems. Using the position and the estimated weight of each airframe part, the centers of mass for the primary airframe components (wing, fuselage, horizontal tail, vertical tail) were determined. The weight and balance worksheet is tabulated in Table 5.1.

The tail ballast is necessary to balance *Milk Run* at the 30% mean aerodynamic chord location. The addition of tail ballast was a compromise made to facilitate easy battery storage in the nose. Without the use of tail weight, the battery location would have to be split fore and aft of the wing. The team decided that the 11-ounces of tail ballast was justified to offset the increase in structural weight and, more importantly, the long lengths of battery wiring necessary if the batteries were located farther aft. The resistance of the longer length of battery wire would have made a significant reduction in motor performance.

Table 5.8. Weights and balance worksheet for *Milk Run*. The weight of each component is listed along with its distance from the wing root leading edge. The positive direction is in the direction of the nose of the aircraft. The desired CG location and the calculated CG location appear at the bottom of the figure.

Component	Weight (oz)	Distance from Wing L.E.
Wing	115.500	-6.975
Fuselage	197.729	2.230
Horizontal Tail	14.202	-48.590
Vertical Tail Assembly	6.800	-43.322
Ballast	11.600	-55.000
Total Weight no Cargo	345.831	
	Calculated CG from Wing L.E.	-5.75
	Desired CG	-5.75

5.3. Component Selection and Systems Architecture

The radio and propulsion system components were carefully selected to conform with the DBF regulations. A detailed layout of the radio and propulsion systems is included in the drawing package.

5.3.1. Radio System. The JR XP8103 PCM radio system was used primarily because MSU already owns a system from the previous year's competition. The PCM radio has a variety of programmable mixes for flight modes such as camber, reflex, and flaperon. Also, elevator can be mixed in to offset flap induced pitching moments. This PCM radio has electronic failsafe as stipulated by DBF regulations.

5.3.2. Servos. The JR DS3421 Coreless Digital Servo was chosen to actuate all control surfaces. The digital servos have extraordinary torque, speed, and resolution compared to standard analog servos. The digital servos were sized to move the ailerons 90 degrees at 100-mph. The ailerons and elevator are driven by two servos; the rudder is actuated by only one servo.

5.3.3. Propulsion System. During preliminary design, the Astroflight Cobalt 90 FAI motor was chosen as the only viable power plant for this size aircraft (single engine). The Motocalc analysis showed that the Astroflight 1.64:1 Super Gearbox was a good match for this motor, providing the necessary thrust to reach flight speeds of about 60-mph. Unfortunately, Astroflight no longer makes this gearbox. The gearbox they now manufacture has a gear reduction that is too high for this airplane. Due to the lack of commercially available drive reductions, it was decided to purchase the necessary gears, bearings, and other materials to make a custom gearbox. The gearbox will drive APC Pattern propellers. APC propellers are widely known for their accurate airfoils and consistent performance.

Milk Run's battery packs were custom built by SR Batteries, a company widely recognized as the best in model airplane battery needs. The aircraft carries two flat 12-cell packs, and one triangular 12-cell pack, for a total of 36 cells. The batteries are matched Sanyo 2500-mah NiCad cells for optimum performance. A 40-amp blade-style fuse is located in the circuit between the batteries and the speed controller. This fuse is mounted in the top of the forward fuselage for quick disarming of the motor system when the aircraft is being handled on the ground. The speed controller chosen is the standard Astroflight model 204D. This controller was selected because it satisfies the voltage and current requirements.

5.3.4. Landing Gear. The Robart oleo-strut landing gear was chosen to cushion *Milk Run's* landings. These gear have proven reliable for many years in large scale model aircraft. The oleo-type struts provide superior shock absorption and reliability when compared to music wire or aluminum landing gear. Robart 4-inch treaded wheels complete the landing gear system.

5.4. Rated Aircraft Cost Worksheet

The Rated Aircraft Cost (RAC) score is broken down into three components: mean empty weight (MEW), rated engine power (REP), and manufacturing man-hours (MFGHR).

The design MEW of the aircraft is 21-lbf This is multiplied against a multiplier of $100/lbf for a total MEW cost of $2100.

The REP is based on the weight of the battery packs and the number of motors. For a single engine aircraft with 4.9-lbf of batteries, the REP cost is $7350.

The MFGHR cost is computed from a prescribed work breakdown structure. This work breakdown structure is tabulated in Table 5.2. The 265.15 manufacturing man-hours are multiplied by a \$20/hour multiplier to determine the MFGHR cost of \$5303.

The actual RAC score is the sum of the MEW, REP, and MFGHR costs normalized by \$1000. The RAC for *Milk Run* is 14.753.

Table 5.9. Work breakdown structure for manufacturing man-hours.

Manufacturing Hours Summary				
Work Breakdown Structure (WBS)	WBS Component	Number of Component Units	Hours / Unit	Total Hours
Wing	Wingspan (ft)	10	8	80
	Maximum Exposed Chord (ft)	1.417	8	11.336
	Number of Control Surfaces	2	3	6
	Total Wing MFGHR			97.336
Horizontal Stabiliser	Wingspan (ft)	3.25	8	26
	Maximum Exposed Chord (ft)	0.833	8	6.664
	Number of Control Surfaces	2	3	6
	Total Horizontal Stabilizer MFGHR			38.664
Fuselage	Fuselage Length (ft)	7.415	10	74.15
	Total Fuselage MFGHR			74.15
Empenage	Number of Vertical Surface w/ Control	1	10	10
	Number of Horizontal Surface w/ Control	0	10	0
	Number of Vertical Surface w/o Control	0	5	0
	Number of Horizontal Surface w/o Control	0	5	0
	Total Empennage MFGHR			10
Flight Systems	Number of Servos	6	5	30
	Number of Speed Controls	1	5	5
	Total Flight Systems MFGHR			35
Propulsion Systems	Number of Motors	1	5	5
	Number of Propellers	1	5	5
	Total Propulsion Systems MFGHR			10
Total Manufacturing Man Hours				265.15

5.5. Final Specifications and Performance Data

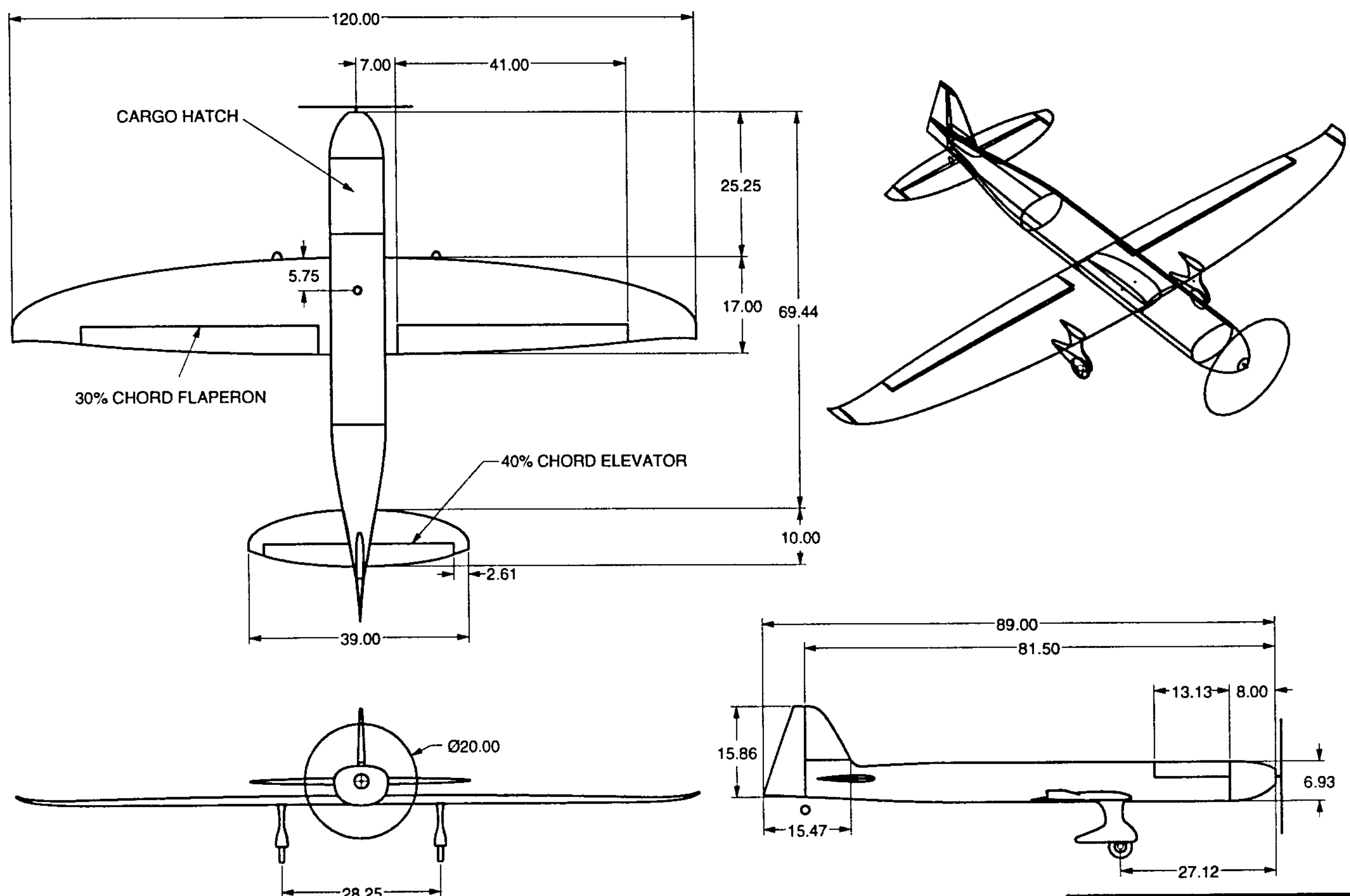
Important dimensions of the final aircraft design are tabulated in Table 5.3. Some general performance measurements are tabulated in Table 5.4.

Table 5.10. Important Dimensions of Final Design

Wing	Wingspan	120	in
	Wing Root Chord	17	in
	Wing Area	1614.4	in^2
	Mean Aerodynamic Chord	15.36	in
	Aspect Ratio	8.92	-----
	Airfoil	SD 7080	-----
	Center of Gravity Location	30%	MAC
	Static Margin	19.5%	MAC
Horizontal Tail	Horizontal Tail Span	39	in
	Horizontal Root Chord	10	in
	Horizontal Tail Area	307.3	in^2
	Mean Aerodynamic Chord	9.05	in
	Aspect Ratio	4.95	-----
	Moment Arm	43.55	in
	Airfoil	NACA 0009	-----
Vertical Tail	Vertical Fin Span	15.86	in
	Vertical Fin Area	191.1	in^2
	Moment Arm	49	in
	Airfoil	NACA 0009	-----
Fuselage	Fuselage Length	88.98	in
	Fuselage Width	9.57	in
	Fuselage Height	6.93	in
	Fuselage Frontal Area	55.3	in^2
Landing Gear	Landing Gear Stance	28.25	in
	Angle	16.5	degrees

Table 5.11. Performance specifications for final design. Performance data is based on results for 20x13 propeller.

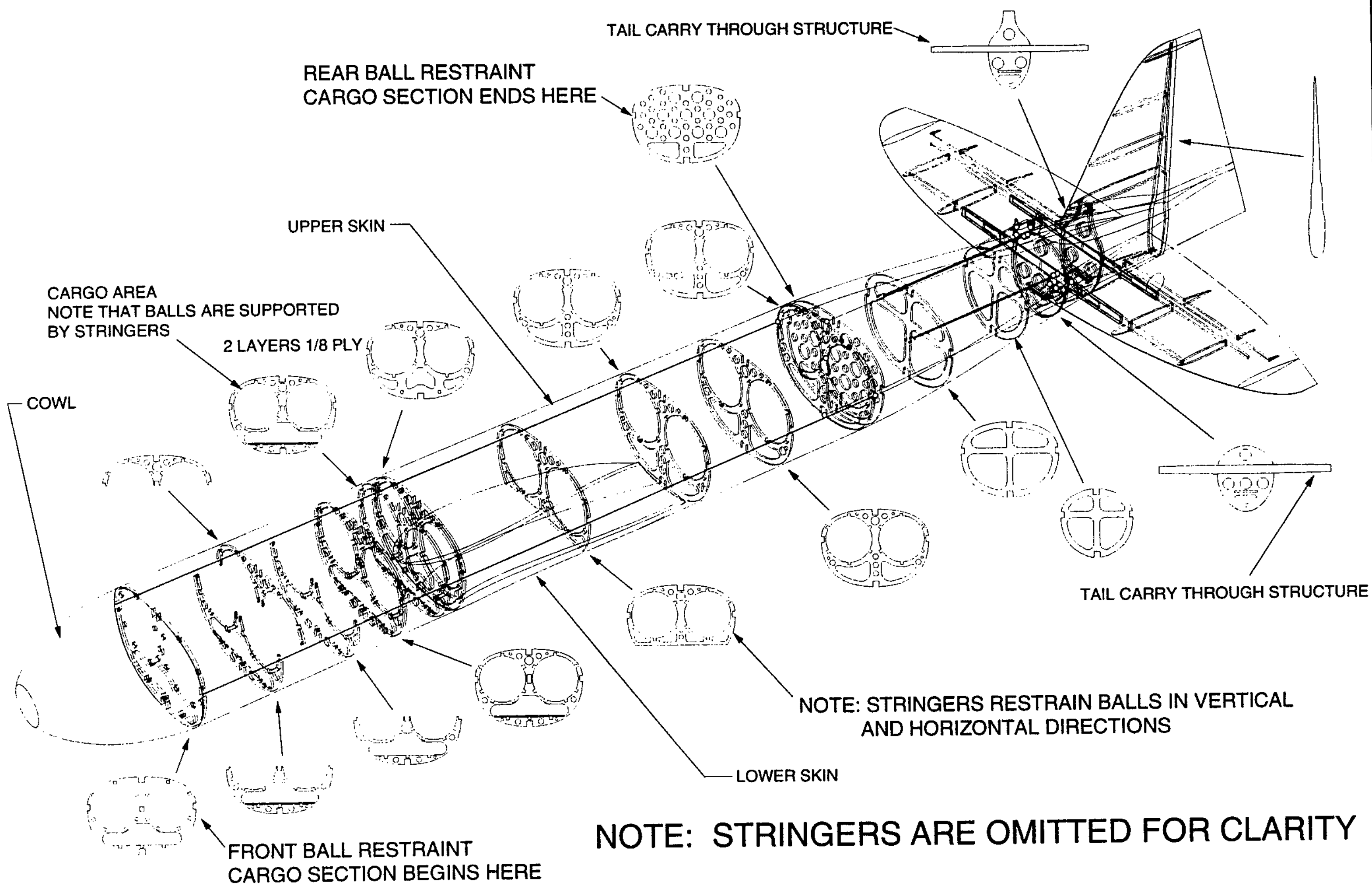
	Empty	Full	units
Weight	20.16	30	lbf
Wing Loading	28.8	42.8	oz/ft^2
Cruise Speed	60	60	mph
Stall Speed	26.7	32.5	mph
Cruise C_L	0.2	0.3	-----
C_{lmax}, no flaps	1.03	1.03	-----
2-G Turn Radius	139	139	ft
2-G Turn Rate	36.3	36.3	deg/sec
Take Off C_L	0.89	0.89	-----
Take Off Distance	80	112	ft
Climb Angle	-----	10	degrees
Climb Rate	-----	665	ft/min
Endurance	-----	7	min.



MILK RUN

THREE VIEW

MISSISSIPPI STATE UNIVERSITY
AEROSPACE ENGINEERING



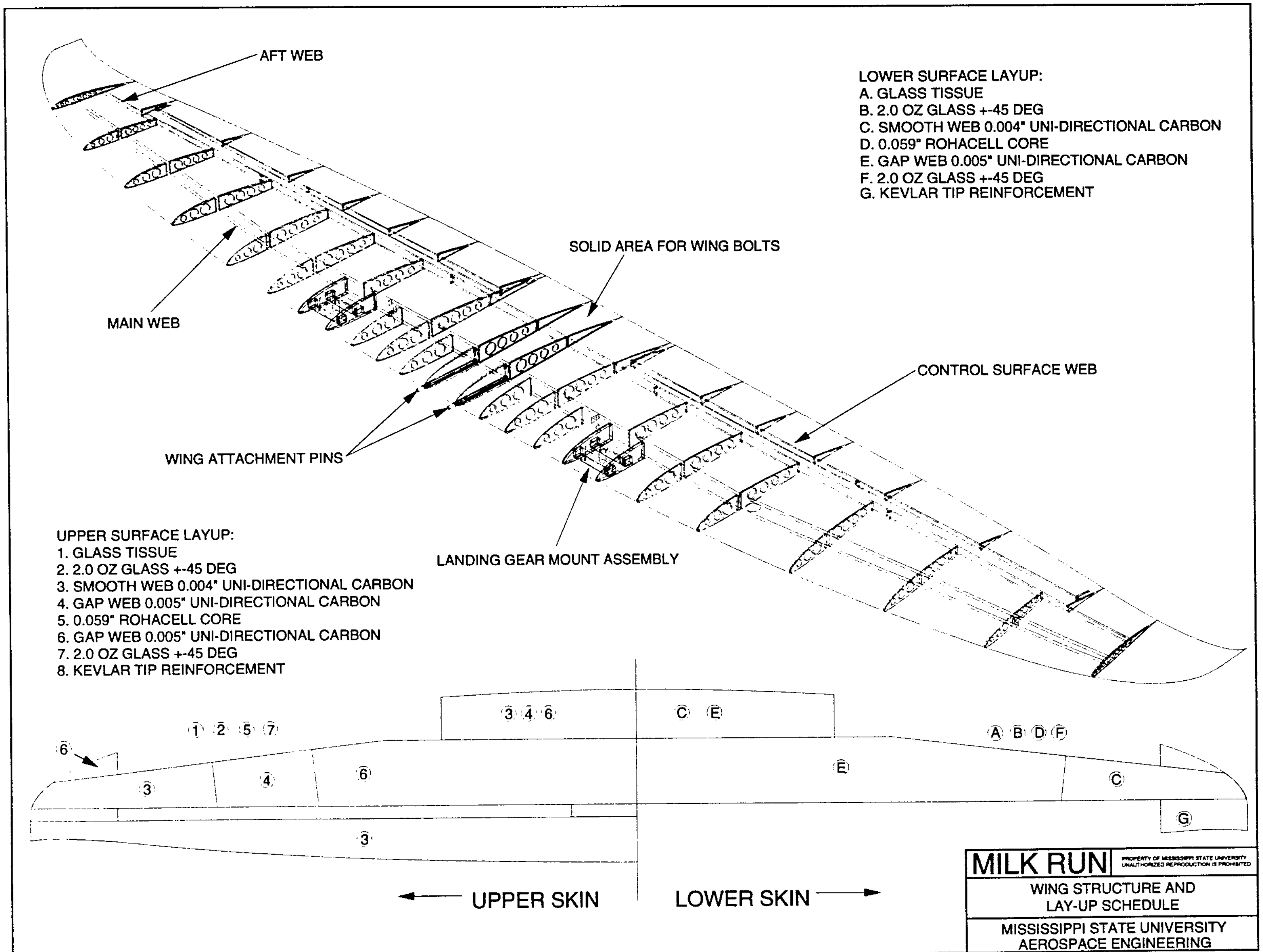
NOTE: STRINGERS ARE OMITTED FOR CLARITY

UPPER FUSELAGE LAY-UP:
1. GLASS TISSUE
2. 2.0 OSY GLASS 0-90
3. 2.0 OSY GLASS +-45
4. 0.0059" ROHACELL CORE
4. 2.0 OSY GLASS +-45
5. 2.0 OSY GLASS 0-90

LOWER FUSELAGE LAY-UP:
1. GLASS TISSUE
2. 2.0 OSY GLASS 0-90
3. 2.0 OSY GLASS +-45
4. 0.0059" ROHACELL CORE
4. 2.0 OSY GLASS +-45
5. 2.0 OSY GLASS 0-90

ALL BULKHEADS ARE FROM 1/8 PLY

MILK RUN	PROPERTY OF MISSISSIPPI STATE UNIVERSITY UNAUTHORIZED REPRODUCTION IS PROHIBITED
FUSELAGE ASEMBLY	
MISSISSIPPI STATE UNIVERSITY AEROSPACE ENGINEERING	


AFT WEB
MAIN WEB
SOLID AREA FOR WING BOLTS
WING ATTACHMENT PINS
LANDING GEAR MOUNT ASSEMBLY
CONTROL SURFACE WEB
LOWER SURFACE LAYUP:
A. GLASS TISSUE
B. 2.0 OZ GLASS +-45 DEG
C. SMOOTH WEB 0.004" UNI-DIRECTIONAL CARBON
D. 0.059" ROHACELL CORE
E. GAP WEB 0.005" UNI-DIRECTIONAL CARBON
F. 2.0 OZ GLASS +-45 DEG
G. KEVLAR TIP REINFORCEMENT
UPPER SURFACE LAYUP:
1. GLASS TISSUE
2. 2.0 OZ GLASS +-45 DEG
3. SMOOTH WEB 0.004" UNI-DIRECTIONAL CARBON
4. GAP WEB 0.005" UNI-DIRECTIONAL CARBON
5. 0.059" ROHACELL CORE
6. GAP WEB 0.005" UNI-DIRECTIONAL CARBON
7. 2.0 OZ GLASS +-45 DEG
8. KEVLAR TIP REINFORCEMENT
6
1 2 5 7
3
4
6
3 4 6
3
C E
A B D F
E
C
G
UPPER SKIN
LOWER SKIN
MILK RUN
WING STRUCTURE AND
LAY-UP SCHEDULE
MISSISSIPPI STATE UNIVERSITY
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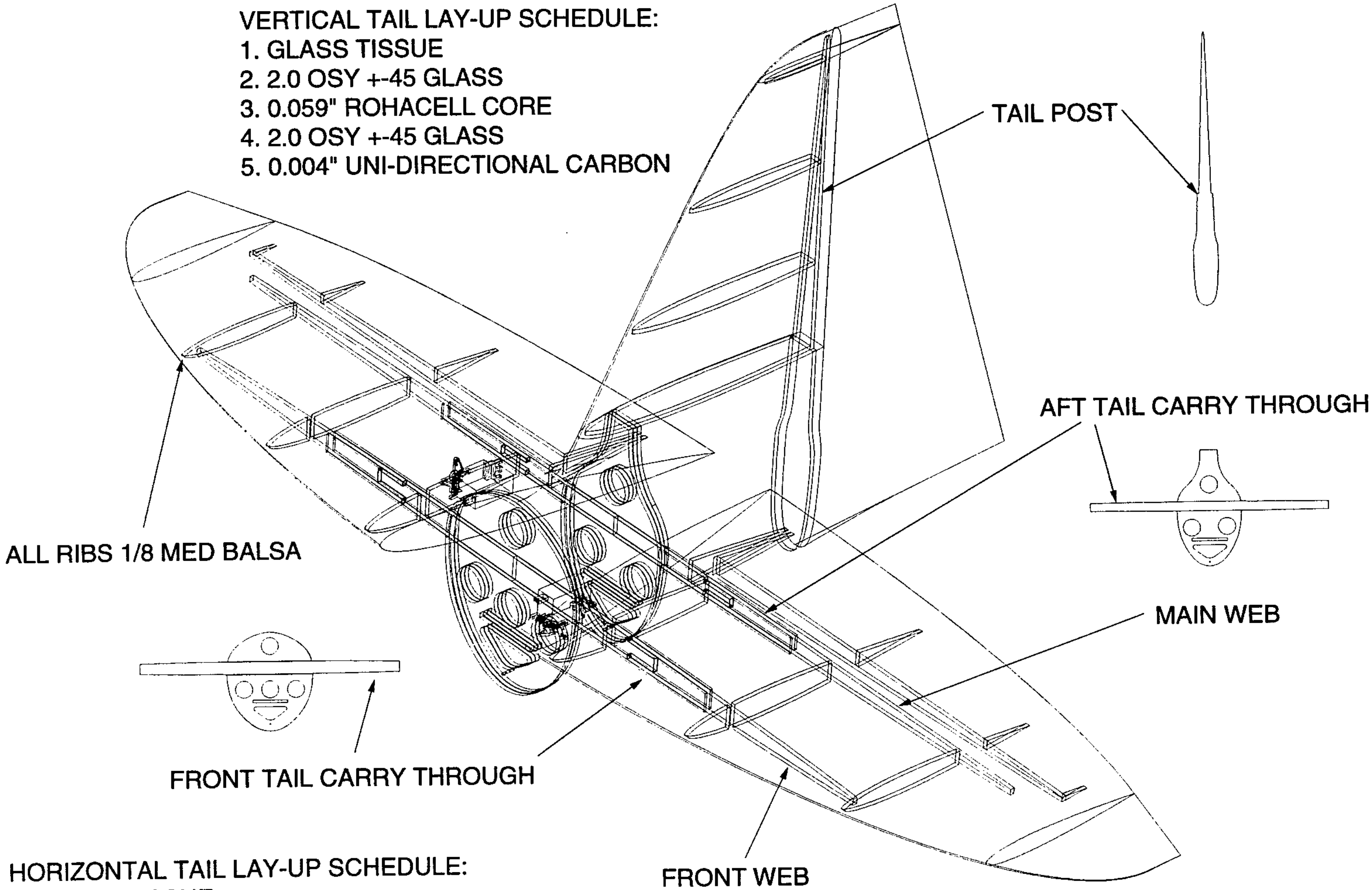
VERTICAL TAIL LAY-UP SCHEDULE:
1. GLASS TISSUE
2. 2.0 OSY +-45 GLASS
3. 0.059" ROHACELL CORE
4. 2.0 OSY +-45 GLASS
5. 0.004" UNI-DIRECTIONAL CARBON



HORIZONTAL TAIL LAY-UP SCHEDULE:
1. GLASS TISSUE
2. 2.0 OSY +-45 GLASS
3. 0.059" ROHACELL CORE
4. 2.0 OSY +-45 GLASS
5. 0.004" UNI-DIRECTIONAL CARBON

CARRY THROUGH MATERIAL: 0.125" THICK 6061-T6

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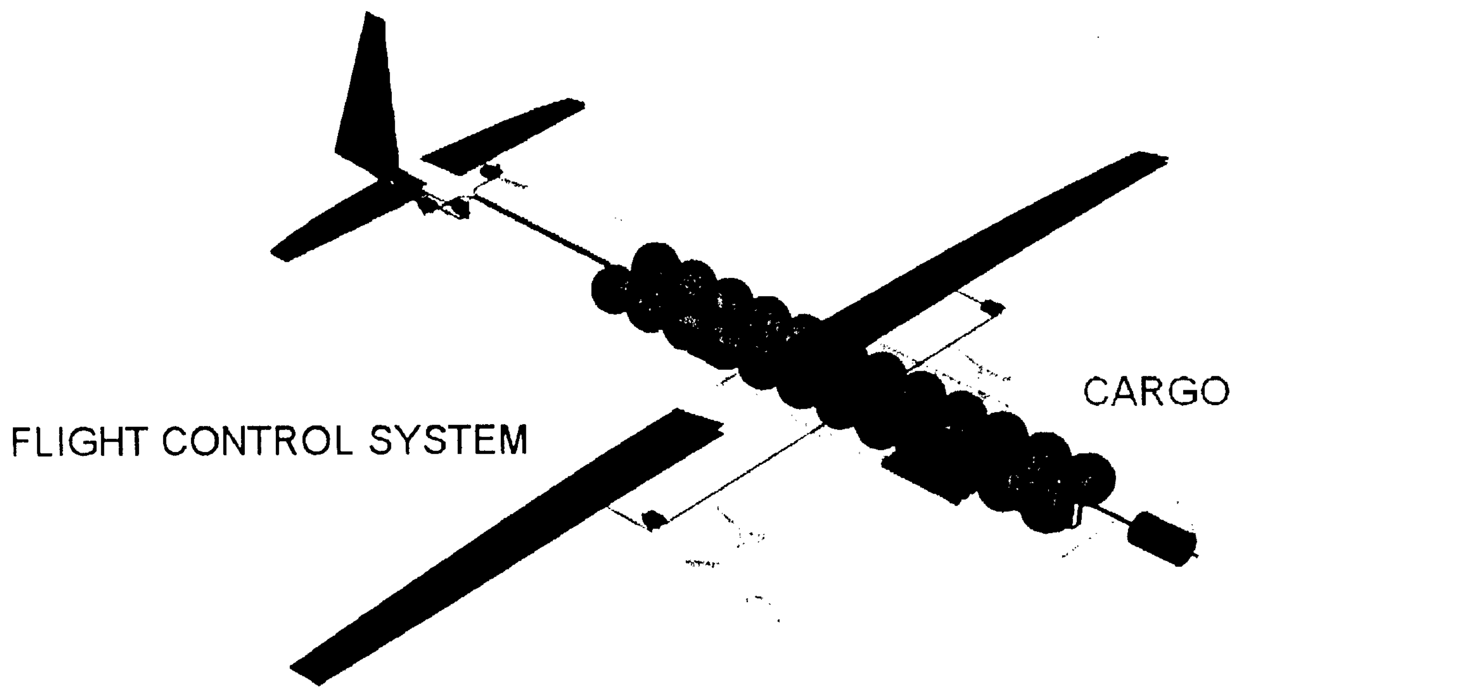
EMPENNAGE ASSEMBLY DEATIAL	
MISSISSIPPI STATE UNIVERSITY AEROSPACE ENGINEERING	



SYSTEMS LAYOUT

MISSISSIPPI STATE UNIVERSITY
AEROSPACE ENGINEERING

6. Manufacturing Plan

6.1. Manufacturing Process Selected

The design of the *Milk Run* was strongly linked with the manufacturing methods to be used. During the conceptual design phase, many different construction methods were considered and investigated, ranging from traditional built-up balsa and ply structures to CNC molded stressed-skin structures. As the team moved further along in the conceptual design phase, a more concrete decision toward using the CNC molded skin method was made. This allowed the team many possibilities for design attributes that would be difficult to manufacture using other methods.

In the preliminary design phase, the team made a final decision that an all CNC molded, stressed skin structure would be implemented for *Milk Run*. This decision shaped the aircraft from then onward, and allowed the team to realistically consider design options such as elliptical platforms for the wing and tail.

To further aid in the construction of the aircraft, Team *Milk Run* chose to use a CNC laser cutting process to manufacture all of the aircraft's internal components such as ribs and bulkheads. Eliminating hand cutting of the aircraft's components greatly increased our design options for these components (for example, the judicious use of lightening holes in the bulkheads and wing ribs). Reducing hand labor also greatly reduces the amount of time required to complete an airframe.

6.2. Description of Molded Construction Process

The molded construction method can be broken down into its basic steps:

1. Lay up sandwich skins
2. Manufacture internal components
3. Installation of internal components
4. Closing (joining together) skins
5. Finish work on closed components (cutting control surfaces, etc.)
6. Integration of finished components

6.2.1. Sandwich Skin Construction of Wing, Tail, and Fuselage Components. After completion of the necessary tooling, each of the major component's skins are constructed in a similar method. For each part, the required number of exterior plies of the appropriate weight of carbon, glass, or Kevlar are placed into the female mold and wet out with epoxy resin. While the material plies are being wet out in the mold, other team members apply resin to the interior skin layers on a plastic sheet. When all of the plies are ready, the Rohacell core is placed on top of the exterior layers, and then the interior layers are placed onto the Rohacell core according to the predetermined lay-up schedule.

A layer of peel ply, porous release film, and then breather cloth are then applied on top of the skin laminate to facilitate the vacuum bagging process. The nylon vacuum bag is then sealed around the

mold, and a 29 in Hg vacuum is applied. By using a strong vacuum to press the plies and core together, good bonds are assured and void content is decreased in each part.

6.2.2. Manufacturing of Internal Components. The spar webs for the wing, horizontal, and vertical tail assembly are constructed using a vacuum bagging process. The core of the spar consists of a medium density balsa wood that is laser cut to the required shape. The core is then laminated with a carbon fiber sock material to provide shear strength.

All of the ribs, bulkheads, and other wooden parts of the *Milk Run* aircraft are laser cut. Upon completion of the cutting, the parts are ready for installation.

6.2.3. Installation of Internal Components. The internal components such as ribs, bulkheads, and webs are bonded to one skin (the top wing skin for example). All of the components are bonded with structural epoxy into their appropriate locations with the use of laser cut jigs. These jigs allowed for rapid and accurate installation of internal components. After the structural components are installed, the aircraft control systems (servos, wires, etc.) are installed in their appropriate locations.

6.2.4. Closing of Structures. After the internal components were bonded to one half of the assembly and cured in the step above, closure of the components could progress. As depicted in the image below, closing the structure is simply bonding the two halves together; however, this is one of the most crucial steps-there is no turning back. If something is forgotten, too bad!

6.2.5. Finish Work on Closed Components. When the structural resin bonding the components has cured, the parts are ready for their final finish work. For the wing, this process includes cutting the control surfaces free. For the fuselage, the finish work consists mainly of cutting the cargo hatch and access panels.

6.2.6. Integration of Finished Components. After completion of major components, the items are integrated into the airframe. The major components are aligned and mounted in the order listed below:

1. Wing mounted to fuselage
2. Horizontal tail mounted to fuselage
3. Vertical tail mounted to fuselage
4. Rudder attached to fin
5. Installation of fairings
6. Landing gear installed

Assembling the components in the above order allows for relative easy alignment of components. Installing the landing gear last allows the assembled airframe to be easily handled without fear of damage to the gear or gear mounts.

6.3. Manufacturing Processes Screened

In the preliminary design phase, several different manufacturing methods were investigated. FOMs were used to screen the competing construction techniques. Only methods that were accessible to the team were investigated.

6.3.1. Figures of Merit

6.3.1.1. Weighting of FOMs. Each figure of merit was given a weighting from 1 to 5 in an attempt to quantify that FOMs impact on the overall mission and construction process.

6.3.1.2. Scoring of Competing Concepts for Each FOM. Each concept was investigated and given a scoring for each FOM. The score was given on a scale from -2 being the worst to 2 being the best.

6.3.1.3. Accuracy. This FOM is an attempt to quantify the effect of a correctly and accurately constructed airframe on the performance of the final product. It is important to note that it is entirely possible to build just about any configuration accurately and precisely with any of the methods investigated. To do so is only a matter of how much time can be invested to achieve an acceptable quality of construction. Hence, the accuracy FOM is a measure of how difficult and time consuming it will be to build a high quality airframe. Accuracy in construction directly effects mission performance and thus was given a weighting of 4.

6.3.1.4. Labor. This FOM is a direct measure of how much labor is required to build a quality airframe. Processes such as cutting ribs and foam cores are indicators of labor time. Manufacturing methods with high parts count and high hand labor time are scored lower than construction methods with fewer parts count and less touch labor. There is a limited amount of time available to construct and test a prototype aircraft, hence Labor Time was given a weighting of 4.

6.3.1.5. Assembly. The assembly FOM is an attempt to quantify the difficulty in the total number of assembly operations required to complete the aircraft and its components. Methods requiring numerous operations are scored lower than methods requiring fewer operations to complete an airframe. If an airframe is properly designed, the assembly time for each component will be reduced to a minimum, so it was felt that assembly should only receive a weighting of 3.

6.3.2. Concepts Screened and Rationale Behind Scoring. Each manufacturing method is discussed and the rationale behind its FOM scoring is discussed in the sections that follow. The final FOM scores are tabulated at the end of the discussion.

6.3.2.1. Built-Up Construction. This is the standard method for building model airplanes. Given enough time, it is possibly to produce exquisite work with this construction method. Unfortunately, the built-up method has many drawbacks. A high degree of skill from all construction team members is required to produce a quality airframe. Labor time is very high in this method due to parts count. Assembly requirements can also be high since it is important to carefully align and fit each individual part. Despite all of the drawbacks of this method, it is readily available, as no special tools are required.

6.3.2.2. Composite-Foam Core. This construction method is widely used in the model sailplane world, and was employed in out previous entry into the DBF competition. The composite foam core method, can, with skill, produce very accurate parts. There is, however, much room for error. A simple swipe of the sanding block can ruin a wing surface. Accuracy can suffer in this method if one is in a hurry (as we are). The parts count in this construction method is fairly low, so there is an improvement in FOM score seen over the built-up method. However, there are still numerous jigs, templates, and tools that must be made. The assembly FOM score is improved over built-up due to the lower number of actual airframe parts and the reduction of possible errors.

6.3.2.3. CNC Molded. This construction method at first seemed an unreasonable way to construct a DBF airplane, but as the team looked more closely into the method, the advantages began to show. Primarily, CNC molded construction produces extremely accurate results with little touch labor. If parts are correctly CAD designed, the producing molds from those drawings can be readily accomplished.

The labor involved in CNC molded construction is largely a matter of how much work you allow the CNC machine to do. This is largely dependent on how much machine time one can afford and what materials the molds are cut from. In general, the more machine time and the better the mold stock, the less hand work is required. Since our team had generous donations in both machine time and mold stock material, the CNC molded method received the best score for the labor FOM. Assembly of components with this method is mush simpler than with other methods, as all parts can fit together exactly as designed with little hand work (if properly designed).

6.3.3. Results of FOM Analysis. Table 6.1 below gives the final FOM rankings for each concept. CNC Molded construction was chosen.

Table 6.12. FOM Rankings For Construction Methods

	Accuracy	Labor	Assembly	
Weighting	4	4	3	**Results**
Built-Up	1	-2	2	**2**
Composite-Foam Core	1	0	1	**7**
CNC Molded	2	2	-2	**10**

6.4. Cost

Our team learned many lessons from the 2000-2001 competition. Among these lessons was that the construction method chosen is not necessarily the main driver in the expense of competing.

Although actual cost was considered in the design of our aircraft, it was not an overriding constraint. For example, we would not choose to use an all carbon wing and fuselage skin, as that would more than quadruple the cost of our airframe. In all, however, the airframe cost is not the major portion of the cost of competing in the AIAA DBF competition. Instead, the major expense is incurred in the purchase of the propulsion system and the flight systems and in traveling to the contest site.

Fortunately, we have received generous donations from private individuals, our department, and companies. This has allowed us not only to proceed with our desired construction method, but also to compete in the competition.

6.5. Skill Matrix

Each team member's knowledge and skills were assessed to determine the viability of even competing in the contest. The knowledge gained from this investigation allowed consideration many design options. The team skill breakup is shown in Table 6.2.

Table 6.13. Team Skill Breakup

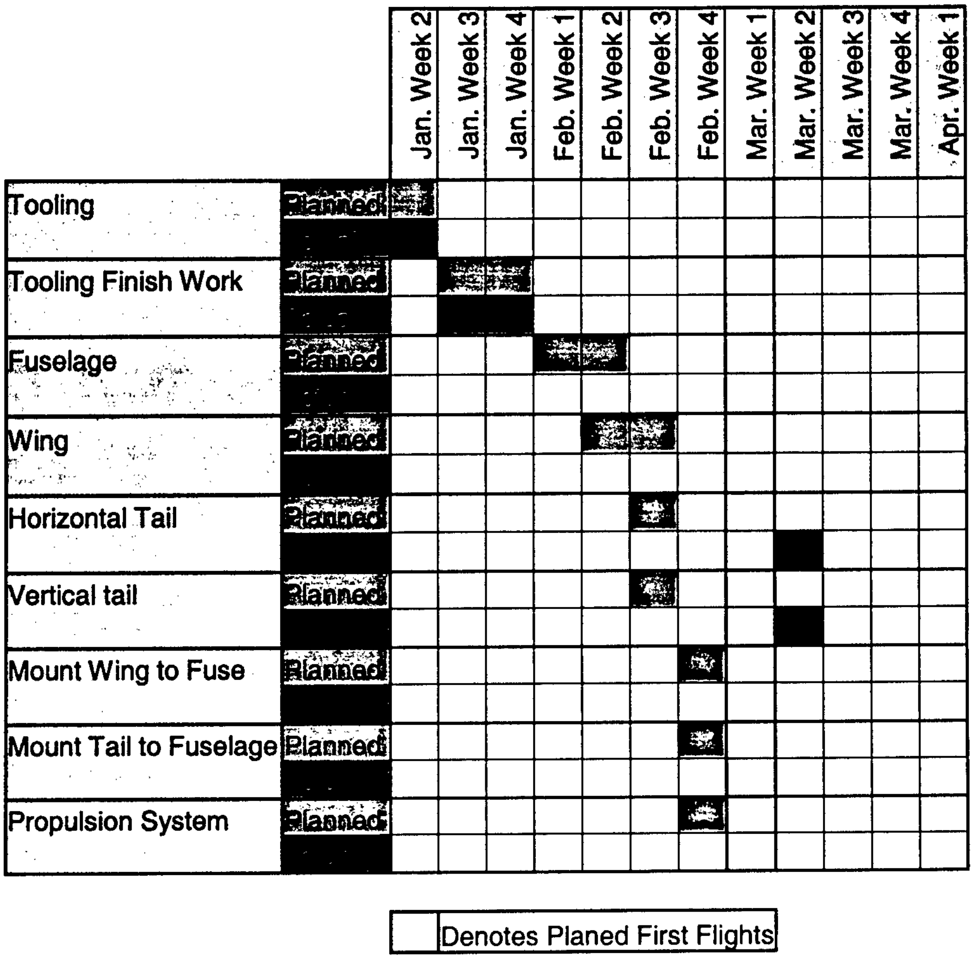
	CAM and CNC Machining	CAD	Built-up Balsa Construction	Composite Lay-up	Vacuum Bagging	Sanding/Shaping	Painting	Foam Hot Wire	Model Airplane Experience
Personnel Available	2	5	4	6	6	7	3	6	5

As can be seen, several team members have experience in R/C modeling, CAD, and other essential areas. A further breakdown of team member capabilities was determined to be unnecessary.

6.6. Scheduling and Manufacturing Milestone Chart

The *Milk Run* team manufacturing schedule and milestone chart is shown below. 24-hour access to our shop is available, so there were few scheduling conflicts; however, workspace for such a large aircraft is at a premium. This necessitated staggering of construction of major components such as the wing and fuselage as can be seen in Table 6.3 below. Unfortunately, there have been delays in the schedule, but an April 1st flight date is now planned.

Table 6.14. Manufacturing Milestone Chart



2002 Cessna ONR Student Design/Build/Fly Competition

Design Report

Cleveland State University

"The Flying Viking"

Table of Contents

1. Executive Summary

Cleveland State University students have banned together to uphold the student chapter of AIAA at the college. This development brought new memberships to the chapter as well as appointed officers to keep the chapter on the rise. This dedication also holds a tradition to participate in the CESSNA ONR Student Design/ Build/ Fly Competition.

The conditions of the competition are to design, build and fly a propeller driven, unmanned aircraft, that is powered electrically and radio controlled. The framework of the aircraft shall be within competition specifications that involve a sensible manufacturing cost, a high aircraft performance, as well as high-quality handling capabilities during flight. The aircraft will be required to complete at least three flight periods within ten minutes. The aircraft will have three flight missions in sequence within one flight period. These missions will include loading and unloading of a payload.

All teams will be judged by a scoring system. The team with the highest score will be awarded first place. Scoring will be determined by three categories, written report score, total flight score, and rated aircraft cost score. The written report score will be based on a design report submitted, covering all aspects of the teams progress and accomplishments of the competition. The report must follow the given outline provided by the competition rules, which will be scored, based on specific categories. The total flight score will be judged based on the ability to follow instructed flight procedures given by the competition. The rated aircraft cost score will be based on the construction and overall structure of the aircraft. These three categories will be used to establish an overall team score.

1.1.1 Conceptual Design

Unique designs, ideas and thoughts were thought out to see how we were going to approach the challenge. Gathering individual ideas and conceptual designs from each person, established the design goals. These ideas were discussed among the team to get a well round plan of attack. The team decided on one conceptual design plan to follow, with the understanding that design parameters can and will be changed, do to research, calculation, and affordable criteria to meet competition rules.

1.1.1 Design Alternatives

There were three main alternatives discussed among the team. Each alternative was very different from one another, and was presented by each sub team to the entire group. The alternatives include different wing configurations, number of wings, positioning of the wings, fuselage configurations, and number of motors. The variety of wing configurations researched was a conventional style, a tandem wing style, canard style and a flying wing style. The fuselage shapes and sizes depended completely on each design considered. Each fuselage design consisted of a vertical airfoil, which was modified for each alternative. The fuselage was to be constructed to carry the payload necessary for the competition. The landing gear style was

researched to determine the best-fit solution for the final design. The team researched whether there was a need for more than a single engine plane. All motors followed the guidelines of the competition, being that all motors considered were of the brushed variety. The motors researched were all Astroflight motors with available specifications for analysis. The motors were all tested with a variety of propellers to determine which best suited the needs for the competition.

1.1.2 Design Tools

The design team utilized several methods to determine the designs throughout the concept design phases. The team using a R/C modeling Aircraft guide (Lennon; Practical Techniques for Building Better Models) as a major design tool developed Microsoft spreadsheets to calculate basic parameters necessary for the development of the variety of concepts investigated. Each aspect of the design, wing, fuselage, Landing gear, motor, controller, and optimization were all modeled in Microsoft Excel and various program tools available through the Internet. Each design parameter were investigated using the R/C modeling principles and from past design team experiences.

1.1.3 Conceptual Design Results

Between the three alternatives that team researched, a concept was chosen to perform design calculations. The investigations performed by the design teams during this phase yielded that the optimum plane would be that of a tandem wing style plane with a elliptical, moderately cambered wing planform, and a wing area of 800 in^2. The fuselage was developed to contain up to 24 softballs. The wing would have a span of 7 ft and fixed landing gear would be used for landing type. A single Astroflight cobalt-90 engine turning an 18x6-10 propeller would power the aircraft. The optimization program developed in excel predicted that the aircraft configuration would have a rated aircraft cost of 15.26. Three other alternatives were considered as well.

1.2.1 Preliminary Design

In the preliminary design, alternatives from the conceptual design were examined and rated. Based on the results of this rating, the tandem and conventional models were eliminated. The canard was found to be the best model for the given specifications. The preliminary design focused on variations to the canard design.

1.2.2 Design Alternatives

After the decision was made to use the canard model, design alternatives for wings, tails, cargo, and landing gear were assessed. This assessment was done using Excel, Electricalc, and JavaFoil.

1.2.3 Design Tools

The above mentioned design tools were used in the following manner. Excel was used to determine weight and wing loading. JavaFoil was used to estimate flight characteristics. Finally, Electricalc was used to analyze the thrust and drag.

1.2.4 Preliminary Design Results

Results were tabulated for manufacturing processes, wing and power loading, and configurations. The results were then finalized in the detailed design.

1.3.1 Detailed Design

During the detailed design phase, the ideas developed in the conceptual design and then refined in the preliminary design were further refined to produce the final aircraft configuration with the optimal performance and lowest cost.

1.3.2 Design Alternatives

The design alternatives were narrowed down to a final configuration in the preliminary design phase; at this point the only alternative that survived the first two phases of design is the pusher-canard design.

1.3.3 Design Tools

The design tools used in the detailed design were similar to the ones used in the preliminary design. JavaFoil was used to predict the lift coefficient of the airfoil sections, Excel was used to calculate the rated aircraft cost and ElectriCalc was used to predict the motor performance and maximum flight time.

1.3.4 Conceptual Design Results

The final plane configuration was a pusher canard type aircraft. The aircraft had a fixed tricycle landing gear. The fore wing area was 70 percent of the aft wing area and the vertical stabilizers were attached the wing tips of the aft wing. The design used one motor with one propeller and drew power from a 29-cell battery pack.

2. Management Summary

2.1 Design Team Architecture

It was decided that the team members would be divided up into individual groups of two people. We established areas of responsibilities that we need to cover in order to produce a well-rounded aircraft. Each member was responsible for two areas, which involved a different team member, as a partner, for each area of responsibility. This helped us interact with each other as a team and provided overlapping ideas and thoughts.

Six diverse groups were established to meet our goal. Each of the group is named as follows: Fuselage, Wings, Motor and Controller, Landing Gear, Optimization, and Analysis and Report. Each group followed specific competition criteria that related to there area of concentration. Development of conceptual goals was presented from each individual group to the whole team in scheduled meetings for review and design alternatives. The 2001-2002 Cleveland State University Design, Build, and Fly design team is comprised of six individuals. Four team members being seniors (mechanical engineering), and two members being underclassmen

(mechanical engineering). The break down of the team and their respective responsibilities are detailed in table 2.1.

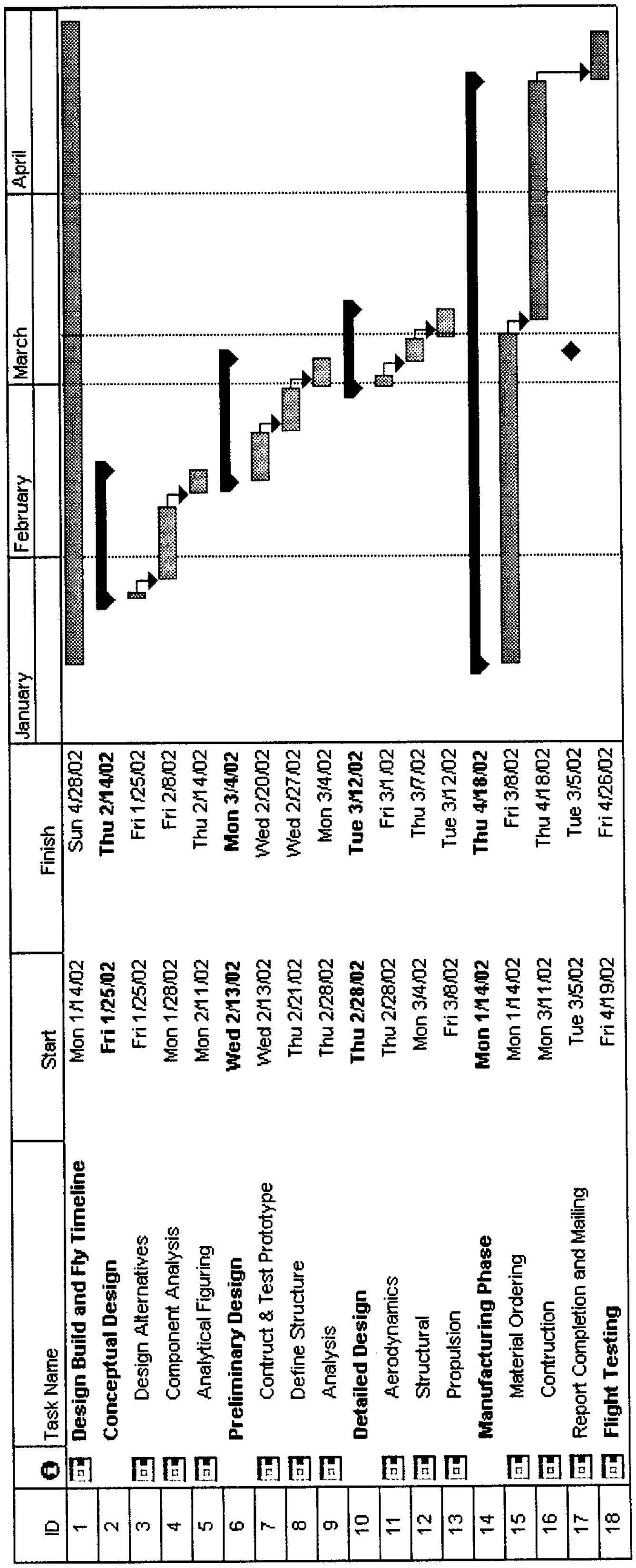
Fady Bishara-**Team Leader and Chief Designer**					
Wings	**Power Plant**	**Fuselage**	**Vertical Fins**	**Landing Gear**	**Construction**
Fady Bishara	Fady Bishara	Kurt McClain	Kurt McClain	Greg McGinnis	Fady Bishara
Doug Wright	Greg McGinnis	Greg McGinnis	Doug Wright	Doug Wright	Greg McGinnis
					Kurt McClain
					Doug Wright
					David Paoletta
					Paul Wenyandt

Table 2.1: Table of architectural structure of design team.

Through the guidance of the team leader, the team was divided into the teams mentioned in table 2.1. Each group's tasks were assigned according to importance. This design team containing all new members to the design competition the experiences of each individual couldn't be taken into consideration when assigning their respective task.

2.2.1 Management Overview

The Flying Viking team determined a schedule for all tasks and assigned each a tentative completion date (milestone). The Team Leader was responsible for the assurance of completion and to oversee all development progress of each individual section. The design team met each week to discuss progress and any current and future problems with their respective sections. The design team met periodically with faculty advisor Dr. E. Keshock. These meetings were devised to discuss the projects progress and developments. The following figure is a time line of the project and its milestones.



ID	Task Name	Start	Finish
1	**Design Build and Fly Timeline**	Mon 1/14/02	Sun 4/28/02
2	**Conceptual Design**	**Fri 1/25/02**	**Thu 2/14/02**
3	Design Alternatives	Fri 1/25/02	Fri 1/25/02
4	Component Analysis	Mon 1/28/02	Fri 2/8/02
5	Analytical Figuring	Mon 2/11/02	Thu 2/14/02
6	**Preliminary Design**	**Wed 2/13/02**	**Mon 3/4/02**
7	Contruct & Test Prototype	Wed 2/13/02	Wed 2/20/02
8	Define Structure	Thu 2/21/02	Wed 2/27/02
9	Analysis	Thu 2/28/02	Mon 3/4/02
10	**Detailed Design**	**Thu 2/28/02**	**Tue 3/12/02**
11	Aerodynamics	Thu 2/28/02	Fri 3/1/02
12	Structural	Mon 3/4/02	Thu 3/7/02
13	Propulsion	Fri 3/8/02	Tue 3/12/02
14	**Manufacturing Phase**	**Mon 1/14/02**	**Thu 4/18/02**
15	Material Ordering	Mon 1/14/02	Fri 3/8/02
16	Contruction	Mon 3/11/02	Thu 4/18/02
17	Report Completion and Mailing	Tue 3/5/02	Tue 3/5/02
18	**Flight Testing**	Fri 4/19/02	Fri 4/26/02

Figure 2.1: Project Timeline

3. Conceptual Design

During the conceptual design phase, the team members researched alternative concepts for the plane. These concepts were run through design calculations using Microsoft Excel and the style of plane was chosen for further considerations. The team then investigated more into this concept to determine what type of alterations could be made to it. Based upon the alterations possible for the chosen concept, the design team determined the best concepts of the chosen style based on manufacturability and figures of merit.

We approached the idea of investigating two types of wing configurations. The conventional style and the canard style. Each have there own advantages and disadvantages. The canard wing style has a less tendency to stall or spin. The wings share the load of the aircraft, which makes for a reduced wing area. The canard wing configuration provides a smaller and lighter aircraft. This is because the required lift can be divided between the two wings. A disadvantage of this wing style is that the fore plane must be much heavier or loaded so to control the stall speed. This is because stall speed of the fore wing is much higher than the stall speed of the main wing. The canard wing style will take off and land faster than the conventional aircraft. This makes a more significant landing length, which can effect the flight time.

3.1 Design and Configurations

Important characteristics were chosen during the concept phase. The design team to determine the best possible configurations of these key components investigated each characteristic. Then each characteristic was chosen based on its rated cost contribution and figures of merit. If a component configuration was deemed un-flyable or difficult to manufacture it was dismissed as a possibility.

3.1.1 Airfoil Configuration

The design team investigated the wide variety of airfoil configurations that are possible in the industry. The goal was to determine the best configuration for the mission profile of the competition. It was decided to go with the canard style of aircraft. Different types of wing styles for the canard aircraft were researched. The style of wings we considered is the rectangular, elliptical, taper and sweptback wings. Each have there own benefits and downfalls to our needs. The rectangular wing is the easiest wing to construct. Each rib of the wing is of the same size and shape. The wing also has a single chord wise curvature. The elliptical wing is the ideal wing for flight. It has the lowest induced AoA, the least amount of drag, and has a less tendency of stalling. Its downfalls are it is difficulty to construct. Each rib is different and wings have double curvature chord wise and span wise. The tapered wing can be as good as the elliptical wing if constructed properly. The wing has lower root bending moments. The wing provides a good deal of strength at the root. It is a lighter and much stronger wing than the rectangular wing. The ribs are also different sizes throughout the wing. The sweptback wing has a smaller tip chord, which causes early tip stalls. The tips are behind the center of gravity. This type of wing has increased

drag and lower lift the larger the sweptback angle. This type of wing is a preferred type of wing for stunt flying.

Although the addition of the second wing will raise the rated aircraft cost the advantages from outweighed the disadvantages. The conventional style aircraft would necessitate a longer wingspan compared to the canard style, which would not only increase rated aircraft cost, but increase wing loading as well. The wing loading is a major consideration when determining a wing configuration. The aircraft landing speed and the wing loading are closely related. The higher the wing load is the greater the landing speed becomes. Another advantage to the canard wing style plane, is the location of the center of gravity, it is basically in the center of the aircraft configuration allowing for a possible increase in payload. A moderately cambered elliptical planform was chosen for further investigation during this phase.

3.1.2 Fuselage Configuration

The fuselage is a key component in the aircraft. Not only is it a main support for the aircraft's apparatuses, but also it potentially needs to support a payload. The fuselage is the corner stone to the assembly of the aircraft, because it houses the wings and tail. There are a couple types of fuselages such as a vertical and horizontal configuration.

The initial consideration was a vertical style fuselage. The vertical configuration would be more space efficient to allow for the greater payload. The design would also be easier to fabricate for the aircraft construction. The downside to a vertical configuration is its increased vertical surface area making it more susceptible to a crosswind action.

The second consideration is a horizontal style fuselage. The horizontal configuration would reduce drag, but limit the space necessary for a payload cargo. The team decided that this would not suit our needs for the competition with a tandem wing style aircraft, due to its low lift characteristics.

3.1.3 Tail configurations

The tail configuration is important for stabilization of the aircraft during all aspects of the mission profile. There are several styles of tail configurations that can be researched and used. The uses of these structures are based upon type of aircraft and the mission profile. The possible types include conventional, a T-shaped tail, and a canard.

The conventional style is the standard configurations for most aircraft. It consists of a vertical tail configuration combined with a horizontal airfoil stabilizer. The conventional is a configuration that has been a proven success by past teams.

The T-shaped tail consists of a vertical tail as in the conventional style, but the airfoil stabilizer is place at the top of the tail instead of locating it near the bottom. This configuration would be more complex to manufacture. From this premise, the team dismissed it as a possibility.

The canard configuration has a few advantages to its style. The major advantage is it is mainly free from stall-spins at low altitudes, which usually end up in a crash. With a canard style it also reduces the size of the aircraft dividing the required wing area between the two lift surfaces. When analyzed using the Aircraft Cost Model Optimization calculations, the canard configuration gave the lowest cost rating. Refer to Figure 5.5 for an explanation.

The canard and conventional style tails were chosen for further evaluation in the next design phase.

3.1.4 Landing Gear

Landing gear is an important part to the design of the aircraft. The landing gear has two major functions for the aircraft. First, the landing gear provides adequate clearance between the propellers and the ground. Secondly, the landing gear allows the aircraft to be able to properly rotate during takeoff and functions beyond. Also, the landing gear can be designed to absorb the shock of the landing. The two main landing gear alternatives are either retracting or fixed. Also, the location of the individual landing gears will have to be selected. The landing gears may be located in a tri-cycle fashion, or in a dual fashion with tail draggers.

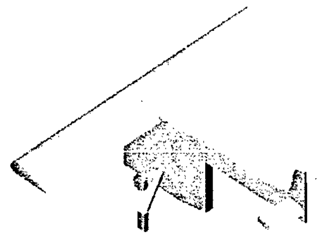
In considering a landing gear type weight is a factor as it has been in all configurations for the aircraft. Among the weight considerations another important factor is the rated aircraft cost. Keeping this in mind the design team investigated both retractable and fixed configurations for the conceptual design phase.

3.1.5 Power Plant

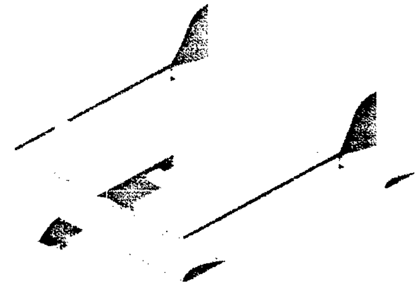
The power of the aircraft is generated from the motor, batteries, and the propeller. The type of motors investigated for the aircraft were Astroflight motors in particular the cobalt-90 model. It was determined using Electricalc, during this phase, that a two motor configuration wouldn't be needed for the sorties being run during the competition. According to the resultant calculations, the team felt that a single motor plane would be sufficient enough to compete in the competition. Furthermore, a single engine plane would reduce the overall rated aircraft cost. The battery cells chosen were that of nickel cadmium and according to competition rules would not exceed the 5-pound limit.

3.2 Configuration Alternatives

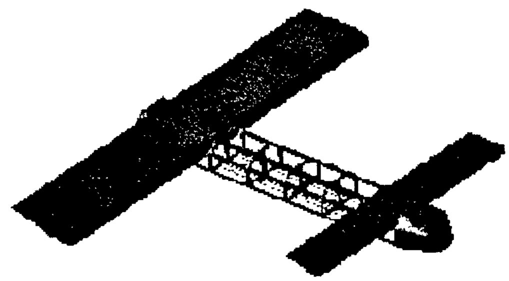
The Flying Viking design team, after much research, decided that there were three alternative ideas to consider. The alternatives consisted of a conventional, canard, and tandem wing styles. These three alternatives all contain possible sub-alternatives that consist of different wing configurations, tail configurations, and fuselage bracing (the tandem concept). These configurations would be sorted through to determine the best possible concept to analyze further for the future design phases.



Conventional Concept



Tandem Concept



Canard Concept

Figure 3.1: Above are three alternative aircraft designs chosen by the design team for investigation during the concept phase.

3.3 Technology Selected

WingFoil and JavaFoil will be used to analyze wing profiles. Selection of a wing profile will depend on its drag characteristics; lift capability, stability, and ease of manufacture. For the Conceptual stage, 3 different Eppler profiles are being examined. The profiles are the E205, E211, and E214. Refer to the Preliminary Design section 4.1.1 for profile selection.

3.4 Figures of Merit

Two major figures of merit were considered contributors when analyzing the concepts. The F.O.M.'s have been divided into the following categories maximum payload and the rated aircraft cost. The payload sortie is based upon maximum number of softballs the aircraft can carry during the given mission. The payload range is between 10-24 softballs. The softball rows cannot be staggered and must be a single ball high. The second figure of merit is the rated aircraft cost. The rated aircraft cost is a contest requirement tool that scores the aircrafts based on the outlined criteria placed in the competition rules. The team placed it amongst the figures of merit for the project as a design tool. The rated aircraft cost requirements are good tools to use in the concept design phase to eliminate concepts that wouldn't receive a high rating.

3.5 Ranking Chart

The ranking chart was used to determine what aircraft design we wanted to proceed with. The chart was divided into significant categories, which have a major impact on the competition score and construction capabilities.

Conceptual Model Number	Wing Configuration	Wing Planform	Dihedral	Fuselage Configuration	Number of Softballs	Estimated Weight W/out Payload	Estimated Flight Time	Rated Aircraft Cost	Estimated Flight Score	Estimated Score	Decision
1	C	E	D	C	16	10.75	8:53	12.965	2.48	25.79	N
2	A	E	D	C	16	9.75	8:53	12.865	2.48	25.99	Y
3	T	E	D	C	20	12.25	9:30	13.115	2.74	28.17	Y
4	C	R	D	C	16	10.25	8:53	12.915	2.48	25.89	N
5	A	R	D	C	16	9.25	8:53	12.815	2.48	26.09	Y
6	T	R	D	D	20	11.75	9:46	14.065	2.66	25.55	N
1	C	E	A	C	16	10.75	9:48	12.965	2.24	23.38	N
2	A	E	A	C	16	9.75	9:48	12.865	2.24	23.56	N
3	T	E	A	C	20	12.25	10:25	13.115	2.50	25.69	Y
4	C	R	A	C	16	10.25	9:48	12.915	2.24	23.47	N
5	A	R	A	C	16	9.25	9:48	12.815	2.24	23.65	N
6	T	R	A	D	20	11.75	10:46	14.065	2.41	23.18	N

Wing Planform
E = Eliptical
R = Rectangular

Wing Configuration
C = Conventional
A = Canard
T = Tandem

Wing Angle
D = Dihedral
A = Anhedral

Fuselage Configuration
C = Conventional
B = Dual Boom

Decision
Y = Good
N = No Good

Table 3.1: Ranking Chart

4. Preliminary Design

Throughout the conceptual design, three programs were used invariably. They were Excel, Electricalc, and JavaFoil. The use of these programs was continued in the Preliminary design. For instance, Excel was used to optimize calculations for the Rated Aircraft Cost. The program Electricalc was used to analyze the power plant. Finally a program called JavaFoil was used to determine and analyze the optimum airfoil. These programs were used in conjunction with one another, through an iterative process, to confirm that one of the three choices of plane configuration was indeed the choice that would have the best chance to win the competition. As was alluded to by the ranking chart, the canard was the configuration chosen to examine further. Therefore, it was the goal of the preliminary design to more closely examine the characteristics of the canard.

First, a small-scale prototype was built and tested. Upon prototype approval, the following items were investigated as figures of merit: manufacturability, fuselage configuration, and power plant assessment. Next, the construction, material and arrangement of the fuselage, wings, tail, and, landing apparatus were resolved. Finally, calculations were performed to support choosing the canard over the conventional and tandem models mentioned in the concept stage.

4.1 Prototype

A small-scale (approximately ¼ scale) prototype was constructed of the canard. The prototype was constructed from balsa wood. This plane was a simple glider with no motor. Although simple, the prototype served its purpose by proving it was capable of flight. Initially, there had been some concern about the team's experience with a two-wing design. This experience was non-existent. Although, after the prototype proved a success, the decision was made to continue pursuing the canard design.

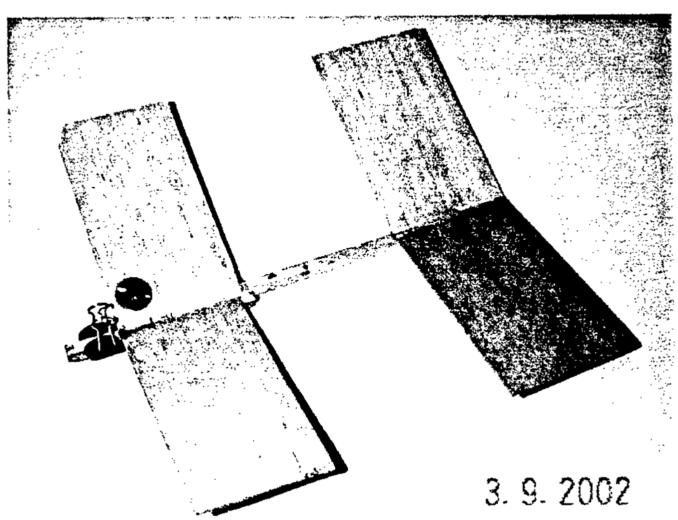


Figure 4.1: Prototype.

4.2 Figures of Merit

Figures of merit are quantitative representations of the analysis that has taken place in order to establish a best choice for the item being examined. The following sections will provide figures of merit for their respective topics. As an overall theme to the project, the teams approach was to maintain a certain level of simplicity. The reason for this is that in general, complexity increases the time, cost, and knowledge needed to perform the design.

4.2.1 Manufacturing

To ease the burden of manufacturing/assembling the plane, many items were researched to determine if they could be bought off the shelf rather than constructed. The limiting factors to this method were cost, weight, and applicability to the plane. Generally if the part being examined fit our needs we bought it and incorporated it as was needed. In other cases, we bought the part and modified it to our needs. If the part did not exist, was not cost efficient, or a better part could be constructed, then we did not buy it.

The manufacturing was completed in a step-wise fashion. By splitting the team into two groups, the manufacturing was completed in a more efficient manner. One group began on the wing construction, while the other group began on the fuselage.

4.2.2 Fuselage

In regards to the fuselage configuration, the main concerns were ease of loading and unloading softballs, center of gravity, weight distribution, and smallest possible surface area. Three ideas were put forth for rapid loading and unloading of the softballs. These were the angled slot, the slotted cylinder, and the dual glove box door. These three concepts were judged on ease of loading, speed of loading, ease of manufacture, and cost. The angled slot was determined to be the easiest to construct and the most cost effective. The slotted cylinder was determined to be the quickest loading and unloading. However, it was discovered that the slotted cylinder violated the rules for loading and unloading. The dual glove box configuration was easy to load and unload as well. However, the complexity and cost of the configuration lead to its demise. Hence, the angled slot was the choice of record. The angled slot is depicted in the following figure. The figure shown depicts a lengthwise section through the fuselage. Therefore, the eight softballs shown are only half of the cargo. Total cargo will be sixteen softballs. The weight is estimated to be 96 oz.

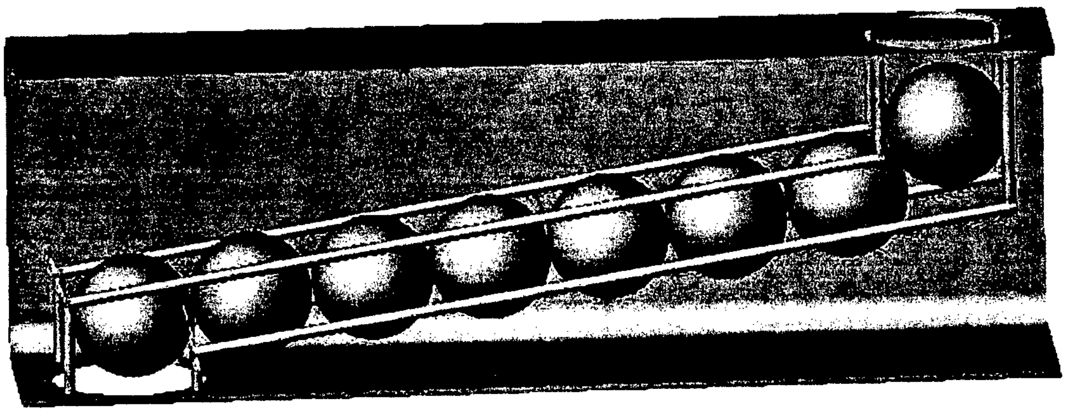


Figure 4.2: Fuselage loading and unloading configuration.

4.2.3 Power Plant

In order to extract the optimal amount of power (while not exceeding 40A), the motor selection, battery selection and configuration, and propeller selection were analyzed. In performing this analysis the following parameters were established: Time of flight needed, weight of Flying Viking (loaded and unloaded), drag, power, and speed. The program used for this analysis was Electricalc. Upon entering or changing the afore mentioned parameters of our design, Electricalc would calculate the results. Next, the results were sorted based on their relative importance. Of primary importance was the amount of time the plane could fly without draining the batteries. The preliminary estimate of time needed to complete the flight mission was 8 min, 53 sec. To add a factor of safety, a ten-minute mission time was used. Further investigation of the power plant is displayed in the analysis section 4.4.

4.3 Material Selection and Configuration

The materials selected for the different components of the plane were to all have one thing in common. They all had to have a high strength to weight ratio. This philosophy was used to build the lightest plane possible while still maintaining structural integrity. Some of the materials selected include Rohacell foam, balsa wood, and carbon fiber rods.

4.3.1 Fuselage

The main support of the fuselage was constructed of Balsa trusses in a two compartment rectangular shape. Cross member support for cargo area, servos, batteries, motor and controller were added where necessary. Also, the fuselage was equipped with a removable top. This removable top provided access to the various control items and batteries. The cargo chambers also were configured with removable lids for loading and unloading of the softballs. These lids were designed to have threaded inserts. Refer to Figure 4.1 for detail of softball entry.

The exterior of the fuselage was then sheeted with balsa plywood. This sheeting enhanced the torsional rigidity of the plane. Prior to sheeting, the corners were sanded to an acceptable radius. MonoKote was then added to produce a smooth finish.

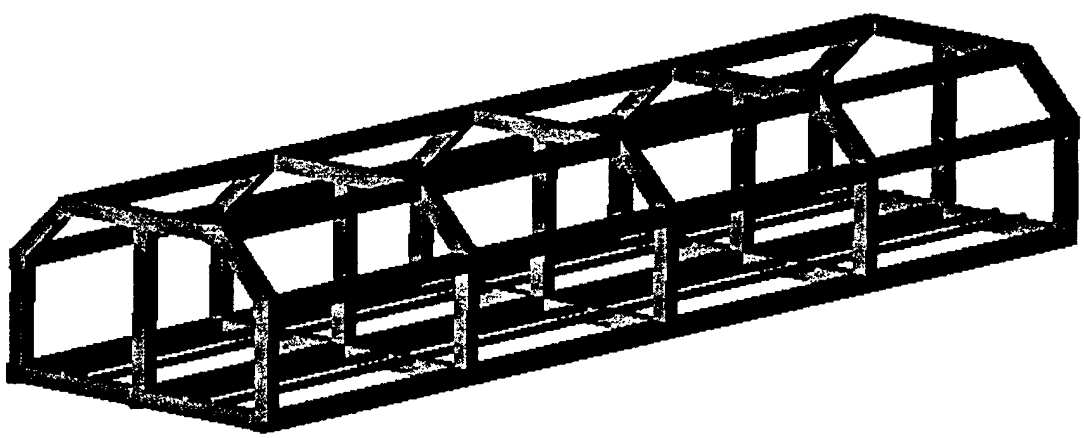


Figure 4.2: Fuselage

4.3.2 Wings

The two wings, fore and aft, were constructed as separate entities from the fuselage. This was done for compact traveling purposes. The wings were constructed as follows. Ribs were cut from Rohacell foam to match the chosen airfoil. A hole was then drilled through each rib. The spar (carbon fiber tube) was inserted into and through each rib. Upon completion of the spar and rib frame, the wing was sheeted with balsa plywood. This sheeting accounted for torsional rigidity.

4.3.3 Vertical Tail

The vertical tails or fins were constructed in the same manner as the wings.

4.3.4 Landing Gear

The landing gear was constructed of music wire. The music wire was coiled so as to absorb the shock of the landing. Calculation for this landing force can be found in the Analysis section 4.4. For stability reasons, a tri-cycle type landing gear was used. Also, to protect against damaging the aft wing and fins, tail draggers were installed on the vertical fins. The length of the landing gear was designed so that the propeller would have 4" of ground clearance minimum. Also, the front landing gear was constructed so as to have steering capability. This aided in taxiing the craft on the runway.

The wheels selected for the landing gear were adjustable to coincide with the maximum amount of force that they must dissipate. Inflating or deflating the wheel accomplished this dissipation of force.

4.4 Analysis

The goal of the analysis was to prove that the selection of the canard was in fact a valid selection. To accomplish this, the analysis was focused on establishing calculations to back up the statements made in the Conceptual and Preliminary sections. For in the Conceptual design, the crafts were chosen based on the amount of lift that they could produce. With this in mind, the conventional design was limited because both of its wings did not provide lift. The tail produced a downward moment to counteract the main wing and hence decrease its overall lifting capabilities. The tandem and canard were assumed to have equal lifting capabilities. The choice between the canard and tandem was made based on their respective fuselage configurations. The canard provided for easier access to the cargo based on the loading configuration chosen. Therefore, the canard was analyzed for wing and power loading in the following sections. Also, statements were made to distinguish tail (fin) and landing gear configurations.

4.4.1 Wing Loading

In order to analyze the wing loading capabilities of the canard, an estimated loaded weight was established. This weight was then distributed over the wing surface area to determine the amount of wing loading. The weight of the loaded canard was estimated to be 22 pounds. The wing area was estimated to be 1200 square inches. Refer to the following figure for a graph of the wing loading.

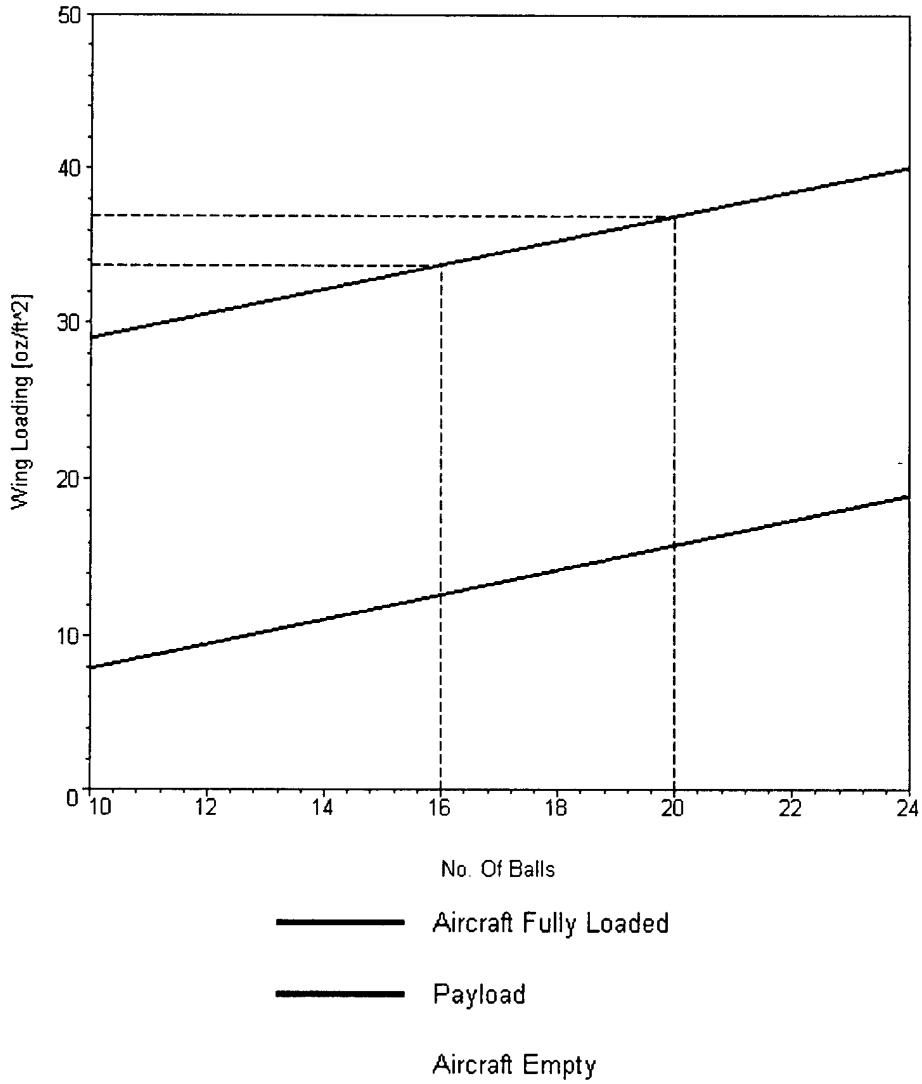


Figure 4.4: Wing Loading

4.4.2 Power Loading

Due to the fact that both wings provided lift for the canard, the motor didn't have to work as hard to keep the plane at level flight. Therefore, more of the motor's energy could contribute to the forward propulsion of the plane. To examine the power loading of the plane more closely, the services of Electricalc were enlisted. The following parameters were input into the program: number and type of cells, motor type, prop type, and estimated drag. The objective of using Electric was to produce a relationship between the power and relative. Refer to the figures below for loaded and unloaded representations of amount of power needed to achieve a particular speed.

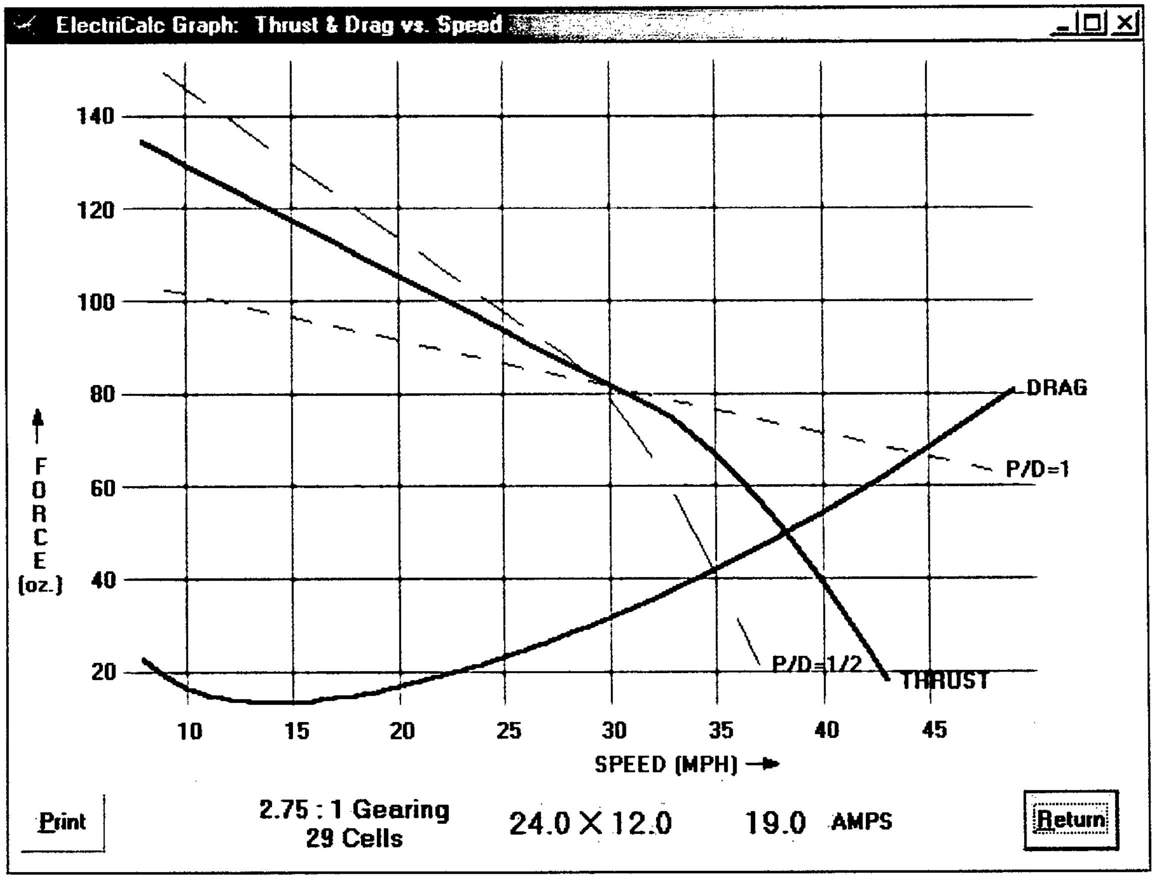


Figure 4.5: Plane is not loaded in this figure

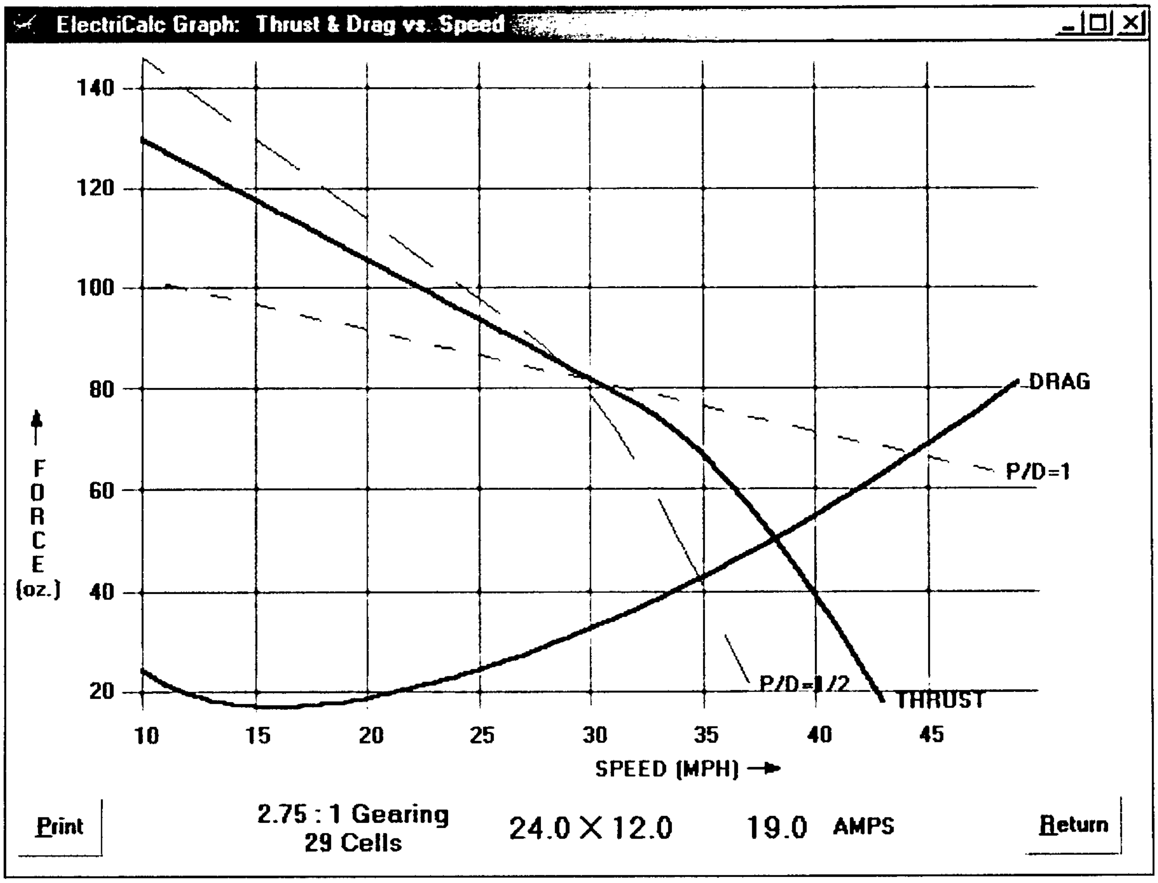


Figure 4.6: Plane is loaded in this figure

From the graphs it was noted that the thrust to speed ratio did not change much whether the plane was loaded or not. This is because once the plane achieved the required thrust to produce lift; the remaining thrust was devoted to producing speed.

4.4.3 Vertical Tail

The vertical tails or fins are designed to be consistent with that of the classical canard. The vertical tails were extended downward from the ends of the aft wing. There were no controls on the vertical tail. In order to size these vertical tails, a procedure outlined by Andy Lennon was followed. The first step performed involved an educated guess as to the size of the tails. Next, the profile area of the entire plane was measured and cut out of cardboard (including the tails). This cardboard model was then balanced to determine the location of the model's center of lateral area (CLA) and also its center of gravity (CG). Upon doing this, it was apparent that material had to be subtracted from the tail in order to achieve the final balance. Material was removed until the model's CG (and CLA) was at 25% of the tail moment arm (TMA) as specified. Aspect Ratio of the tail was also calculated. The dimensions were then finalized for the detailed design drawing set. This tail design provided the Flying Viking with good spiral stability. According to Lennon,

spiral stability is achieved when SSM (Spiral Stability Margin) is 25% of VTMA (Vertical Tail Moment Arm).

4.4.4 Landing Gear

The landing gear was established by examining two different configurations. The two different designs both involved a tricycle style configuration, however the difference between the two was whether or not the landing gears would be on the wing or the fuselage. In examining this closer, it was decided that the landing gears would be located on the fuselage. The reasoning behind this decision was that the wings would have to be reinforced in order to compensate for the stress of landing. Reinforcing of the wing would add unnecessary weight to the plane. Also, small tail draggers were added to the vertical tail to protect the plane in the event of an awkward landing.

5. Detailed Design

The detailed design phase was a spin off from the preliminary phase. As a team we held an initial meeting to begin final calculations, final decisions and our final aircraft model. The basis of these decisions, were based on three major categories: aerodynamic design, structural design and propulsion system design. Each category was studied, and final solutions were decided as described throughout the detailed design. After which, the take off performance, predicted mission performance, and rated aircraft cost were determined.

The component selection was very much based off our preliminary aircraft structure. Our final detailed design structure was developed from our detailed component selection. Optimizations and calculations were developed from each area. The propulsion system made up of the motor and propeller, and the structural design area which included aerodynamics and weight factors. These were areas of concern to develop higher take off performance, handling capabilities, aircraft weight and balance, as well as a low aircraft cost. The overall score is highly determined by the number of softballs that the aircraft can carry. This made each area of concern a much more difficult task. We had to decide on the number of softballs we wanted to try and carry. It became a trade off to what components we wanted and needed to use. The more softballs we wanted to carry, the larger the aircraft needed to be as well as the motor, batteries, and overall structure which plays a big part in weight factor. Using less softball's results in less mass, but it also brings down our overall score. Final structural decisions were made and our detailed final aircraft design was done in Solidworks.

5.1 Aerodynamic Design

The aerodynamic design of the airplane consisted mainly of designing the wings and ensuring that they provided adequate lift with the lowest amount of drag possible. First, we chose a wing loading of 33-36 oz/sq. ft when plane is fully loaded based on our team members' past experience. Second, we calculated the required wing area based on the estimated gross weight

from Table 5.1, which came out to be 1093 square inches. The wing design process consists of two major steps, the first being the planform proportions and the second being the airfoil selection. Paragraph 5.1.1 discusses the planform proportions and paragraph 5.1.2 describes the airfoil selection.

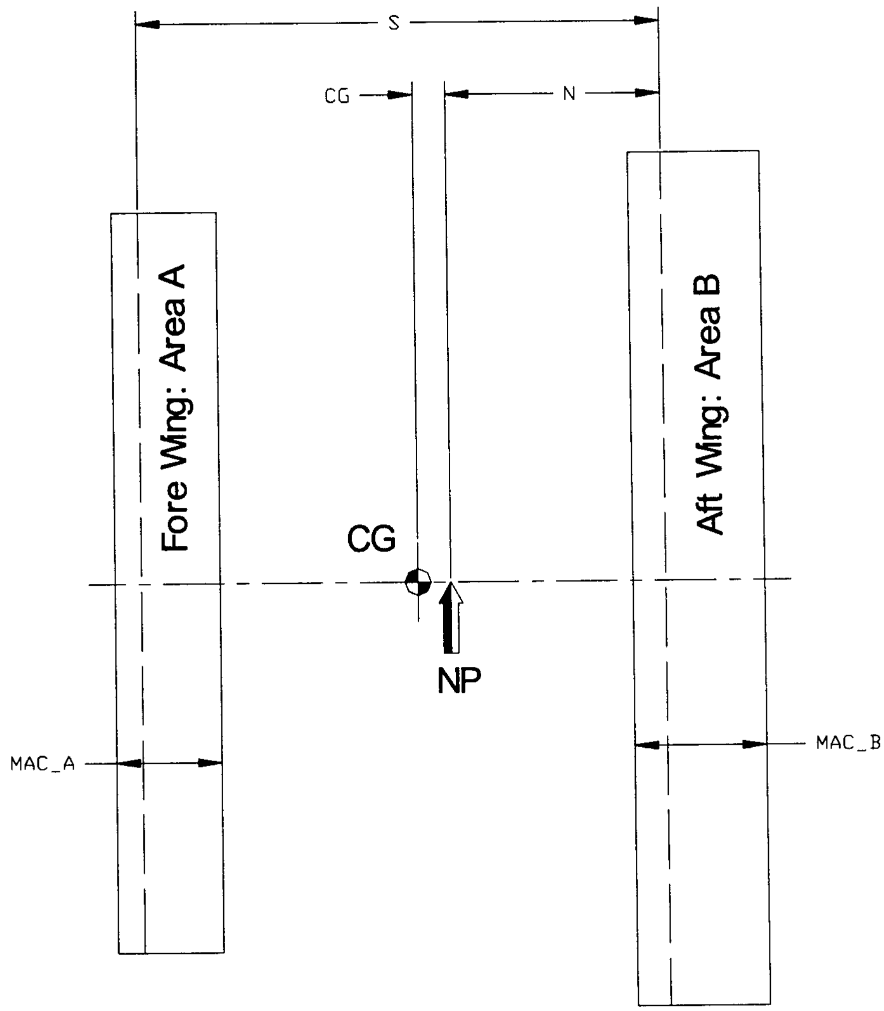
Center of Gravity				
Total Wieght of Airplane				
	Mass Properties		Weight Properties	
Fuselage	4	Lbm	128.8	Lbf
Engine	2	Lbm	64.4	Lbf
Aft Wing	0.8			
Fore wing	1	Lbm	32.2	Lbf
Horizontal Tail		Lbm	0	Lbf
Vertical Tail	0.4	Lbm	12.88	Lbf
Body	1	Lbm	32.2	Lbf
Pay Load	8	Lbm	257.6	Lbf
Batteries	5	Lbm	161	Lbf
Total Weight of Plane	22.2	Lbm	689.08	Lbf

Distance of Component from Reference		
note: Reference is the cone or tip of airplane		
Fuselage	2.500	ft
Engine	5.000	ft
Aft Wing	4.200	
Fore wing	0.300	ft
Horizontal Tail		ft
Vertical Tail	4.200	ft
Body	2.500	ft
Pay Load	2.500	ft
Batteries	1.000	ft
Center of Gravity	**2.312149533**	ft

Table 5.1: Weight and Balance Worksheet

5.1.1 Wing Planform Proportions

In order to determine the wing planform proportions, we developed a Maple worksheet based on the formulas provided by Lennon [1] to present graphically the relationship between the areas and the location of the center of gravity (CG.) Figure 5.1 shows the final wing layout with the fore wing area at 70 percent of the aft wing area.



Where:
NP is the neutral point
CG: center of gravity
MAC_A: mean aerodynamic chord of area A
MAC_B: mean aerodynamic chord of area B
N: distance from 25% MAC_B to the NP
S: separation between the two wings from 25% MAC_A to 25% MAC_B

Figure 5.1: Wing Area Distribution and Layout.

The Maple worksheet calculates the location of the neutral point (NP) based on formula 5-1.

$$N = \frac{Area_A \cdot S}{Area_A + Area_B} \tag{5-1}$$

Once N was known, the location of the center of gravity had to lie ahead of the NP by a distance of 25 percent of the aft wing mean aerodynamic chord (MAC.) This distance is called the static margin and it greatly affects the stability of the aircraft. We chose a separation distance (S) of 40 inches between the forewing and the aft wing to accommodate two rows of eight softballs each. Then, we set the forewing area equal to 70 percent of the aft wing area, which yields a fore wing area A of 449 in^2 and an aft area B of 643 in^2. But in order to determine the MAC and wingspan of each wing we have to set an aspect ratio for each wing. Following Lennon's [1] aspect ration recommendation of 6 to 8 for moderate-speed sport type aircraft, we chose a fore aspect ration of 7 and an aft aspect ration of 6.45. Furthermore, to obtain reasonable chord dimensions we slightly altered the wing areas that we have calculated previously to 448 for the forewing and 645 for the aft wing, Table 5.2 shows the dimensions of each wing using this data; the Maple worksheet calculated a distance N of 16.4 inches ahead of the 25 percent MAC_B line. This places the CG 2.5 inches (25 percent of MAC_B) ahead of the NP that is 18.9 inches from the 25 percent MAC_B line. Thus, almost exactly at the midpoint of the distance S. This is a great advantage when it comes to payload distribution since the payload can be carried between the two wings with the same number of balls ahead of and behind the CG without altering its position from the unloaded state to the fully loaded state.

	Fore Wing Area A	Aft Wing Area B	Units
Area	448	645	$Inches^2$
Aspect Ration (AR)	7	6.45	Unit less
Mean Chord (MAC)	8	10	Inches
Wing Span	56	64.5	Inches

Table 5.2: Fore and Aft Wing Dimensions.

5.1.2 Airfoil Selection

The final step in the process is to choose an airfoil section for both fore and aft wings. Since this is a canard type configuration, caution must be taken when selecting the airfoils for the fore and aft wing. Lennon [1] outlines the selection process as follows:

- The front wing must stall first before the main wing. If that were not the case, the result of an aft wing stall first would result in a violent nose pitch up leading to the fore plane stalling, which can cause crash at low altitudes.
- The main (aft) wing must arrive to the angle its angle of zero lift before the forewing does. Otherwise, if the fore plane stops lifting first the result would be a violent nose pitch down as the aft wing still lifts creating a strong pitching moment.

To satisfy these criteria we must compare the coefficient of lift (C_L) versus the angle of attack alpha (α) for the Reynolds Number (Re) at which the plane will be flying. JavaFoil has the ability to produce these curves given the correct Re, so we must first calculate the Re using the formula 5-2.

$$\mathrm{Re} = \frac{\rho \cdot V^2 \cdot L}{\mu} \tag{5-2}$$

Where: Re: Reynolds number

ρ: density of the air

L: characteristic length, in this case the respective MAC

μ: dynamic viscosity of the air

By substituting our values into equation 5-2, we get 186,600 for the forewing and 233,250 for the aft wing.

With Lennon's [1] guidelines in mind, it seems reasonable to pick Eppler 214 for the forewing and Eppler 197 for the aft wing. Eppler 214 has a lower angle of zero lift than Eppler 197, and therefore will satisfy the criteria set by Lennon [1]. On the other hand Eppler 214 stalls at an angle of attack around 9° whereas Eppler 197 stalls at 12°, which, once more satisfies the airfoil selection criteria. Please refer to Figure 5.2 for a plot of the Eppler 214 airfoil along with its lift versus angle of attack, and Figure 5.3 for the Eppler 197 along with its lift versus angle of attack graph.

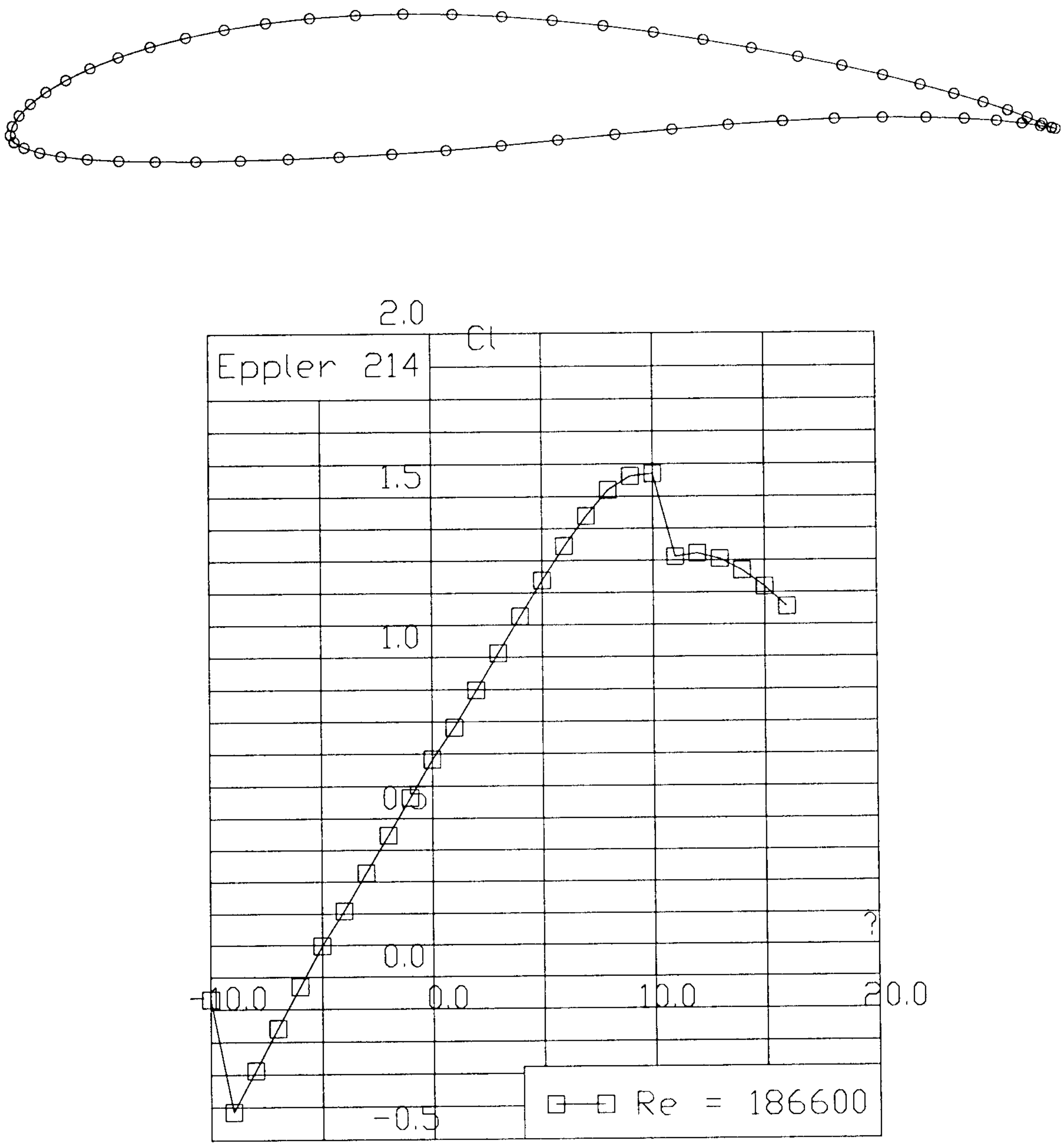
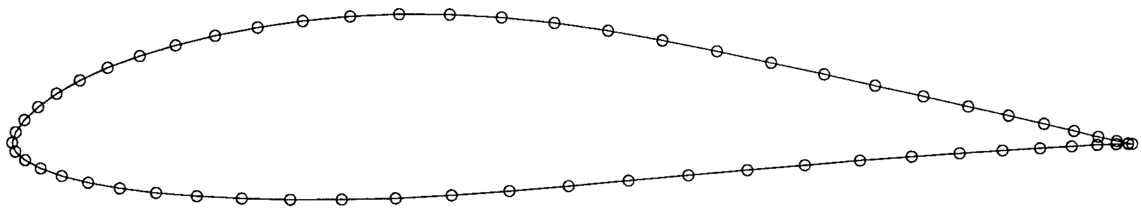


Figure 5.2: Eppler 214 airfoil and lift versus angle of attack (AoA) for the Eppler 214.



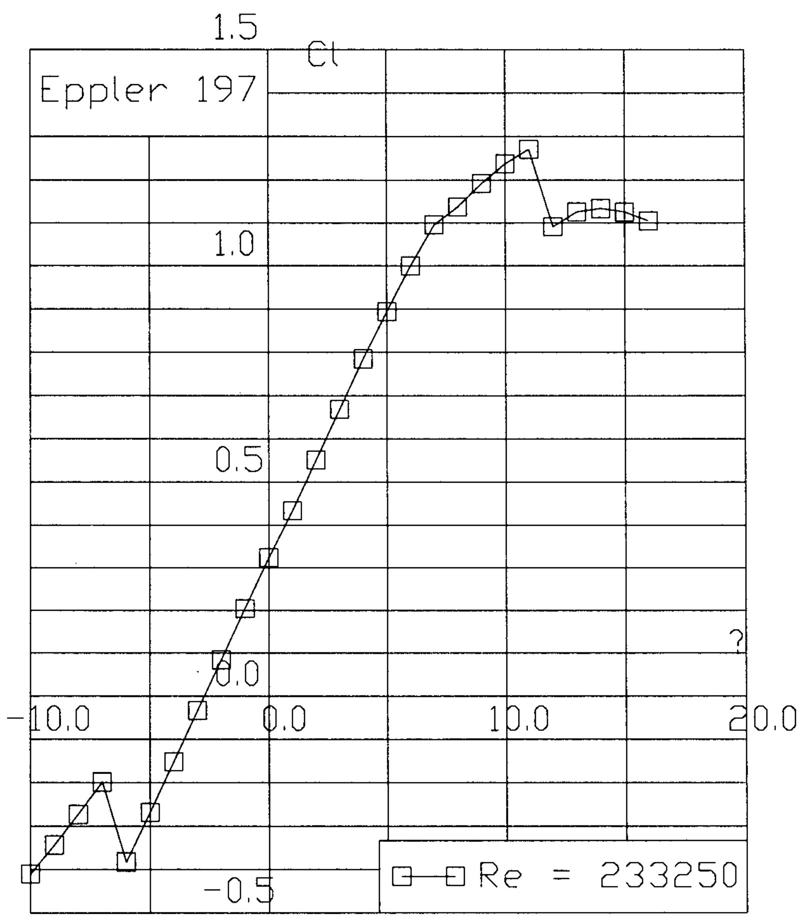


Figure 5.3: Eppler 197 airfoil and lift versus angle of attack (AoA) for the Eppler 197.

5.2 Propulsion Design

The propulsion system for this aircraft consists of an electric motor and propeller. The motor was chosen based on the maximum gross weight of the aircraft using AstroFlight's selection chart. Our final selection was the AstroFlight Cobalt 90 motor with the superbox gear reducer for added torque. The propeller chosen was a zinger 24 x 16 pusher wood propeller. This propeller choice was made based on the maximum efficiency amperage of the motor, cruise time, max speed, and thrust. Electricalc was used to optimize these variables. Electricalc simplified this process by allowing the data from multiple selections to be sorted based on optimal characteristics desired. Speed and thrust were the most critical elements; while still allowing for acceptable cruise time and amperage. The following table shows the selection alternatives. Unfortunately, the optimal propeller discovered, the 24 x 24, could not be used. The reason for this is that a retailer was unable to be found. The 24 x 24 propeller could have been manufactured, however it was not cost effective. The highest pitch propeller available that did not handcuff the budget was the 24 x 16.

ElectriCalc Performance List

---PROP---		GEAR	#		PROP	MOTOR	MIN	CRUISE	SPEED		---CLIMB---	
D	P		CELLS	AMPS	WATTS	KRPM	TIME	TIME	STL	TOP	RATE	HEIGHT
24.0	24.0	2.75	29	28.0	628	6.78	6.1	11.0	22	52	826	5055
24.0	21.0	2.75	29	25.9	591	6.95	6.6	11.8	22	51	826	5448
24.0	22.0	2.75	29	26.6	604	6.89	6.4	11.2	22	51	830	5330
24.0	23.0	2.75	29	27.3	616	6.83	6.3	11.2	22	51	830	5197
24.0	20.0	2.75	29	25.2	578	7.01	6.8	12.2	22	50	819	5553
24.0	19.0	2.75	29	24.5	564	7.07	7.0	12.2	22	49	807	5636
24.0	18.0	2.75	29	23.7	549	7.13	7.2	12.9	22	48	790	5696
24.0	17.0	2.75	29	22.9	533	7.20	7.5	12.6	22	47	769	5729
24.0	16.0	2.75	29	22.1	516	7.27	7.7	13.0	22	45	742	5733
24.0	15.0	2.75	29	21.3	498	7.34	8.0	13.6	22	44	709	5699
24.0	14.0	2.75	29	20.4	479	7.41	8.4	13.3	22	42	672	5628

Print File Sort

13 combinations
11 within limits, 11 listed.

Select Cancel

Figure 5.4: Propeller selection

5.3 Structural Design

The final structural design of the airplane followed the concept discussed in the preliminary design. Since we did not have the capability to run any finite element analysis on the model we over-designed the wings to handle much more load than they would actually be subjected to. The design was mainly based on previous experiences of some of our team members.

5.3.1 Wings

The wing infrastructure consisted of a main hollow carbon fiber spar. Since no wing dihedral was used, the same spar can run from one wingtip to the other without having to be joined somewhere in the middle. ½ inch Rohacell foam ribs were spaced 6 inches apart and epoxied to the spar. The whole wing was then covered with 1/16th inch balsa to give the wing the final shape. The balsa skin also acts as a structural member, this is known as stressed skin construction. The elevator control surfaces are on the forewing and the ailerons are on the aft wing.

5.3.2 Vertical Stabilizers

The vertical stabilizers were placed on the aft wing pointing down. They serve two purposes: maintain lateral stability and act as tip plates. The vertical stabilizers did not have any means of active control (rudders) to simplify construction, save weight and minimize the rated aircraft cost.

5.3.3 Fuselage

The fuselage was constructed entirely out of balsa wood due to the ease of manufacturability. The fuselage also employs the stressed skin construction-technique, the whole fuselage is covered with 1/16th inch sheet balsa. ¼ x ½ inch balsa pieces act as cross members for the fuselage skeleton. The motor is in the rear of the fuselage (pusher configuration.) To offset to weight of the motor, the battery pack is placed ahead of the center of gravity. The payload is carried internally in guided bays side by side with a maximum capacity of 8 balls per bay.

Performance Analysis

The performance of the final design was evaluated using ElectriCalc. The program allowed us to optimize the battery pack, select the propeller, estimate the maximum battery time at cruise condition, and evaluate take-off and climb out performance. The final aircraft configuration is shown in Table 5.3.

Wings	
Fore Wing	
Airfoil	Eppler 214
Area	448 in^2
Span	56 in
MAC	8 in
Total Falpevator Area	67.2 in^2
Aft Wing	
Airfoil	Eppler 197
Area	645 in^2
Span	64.5 in
MAC	10 in
Total Airleron Area	
Total Vertical Stabilizer Area	216 in^2
Center of Gravity Location from 25% MAC_B Line	18.9 in
Maximum Softball Capacity	18
Gross Weight (fully loaded)	16 lbf
Landing Gear Type	Fixed Tricycle
No. of Motors	1
Propeller	24 x 16, 2 Blade
Predicted Rated Aircraft Cost	12.89

Table 5.3: Final Aircraft Configuration.

The fully loaded aircraft is capable of a climb-out performance of 719 ft/min at 28° and maximum cruise speed of 45 mph. The takeoff speed is 25 mph and landing speed is around 35 mph. The total estimated mission time was 8.8 minutes including 2 minutes for loading and unloading the payload.

During the design process we assumed that the wings are over-designed based on previous experience and therefore will be able handle the 2.5 g loading test. This will be proven by flight testing the aircraft and by lifting it from the wingtips.

Using the final aircraft configuration we were able to calculate the rated aircraft cost using the model outlined in the competition rules. The excel spreadsheet shown in Figure 5.X and Figure 5.X calculated a rated aircraft cost 12.89.

Aircraft Cost Model Optimization			
	Rated Aircraft Cost = (A*MEW+B*REP+C*MFHR)/1000		
	Rated Aircraft Cost =	12.89	
A	Manufactures Empty Weight Multiplier		$100.0
MEW	Manufacturers Empty Weight		
	Actual Weight of Plane w/out Payload		10.00 lb
B	Rated Engine Power Multiplier		$1,500.0
REP	Rated Engine Power		
	Total number Engines	1.00	
	Total Battery Weight	5.00 lb (max 5lb)	
	(1+.25*(#engines -1))*Total Battery Weight		5.00
C	Manufacturing Cost Multiplier		$20.0 per Hour

Figure 5.5: Total Rated Aircraft Cost spreadsheet.

MFHR	Manufacturing Man Hours				
	MFHR = Σ WBS hours			219.50	hr
	WBS 1.0 Wing(s)			110.33	hr
	WBS 2.0 Fuselage			54.17	hr
	WBS 3.0 Empenage			20.00	hr
	WBS 4.0 Flight Systems			25.00	hr
	WBS 5.0 Propulsion Systems			10.00	hr

WBS 1.0 Wing(s) (# of wings and total surface area)

Fore Wing					
Span	8 hr/ft x	4.67	ft	37.33	hr
Chord	8 hr/ft x	0.67	ft	5.33	
Cntrl Surf	3 hr/surface	2	Surfaces =	6.00	hr
				48.67	hr
AFt Wing					
Span	8 hr/ft x	5.38	ft	43.00	hr
Chord	8 hr/ft x	0.83	ft	6.67	
Cntrl Surf	3 hr/surface	4	Surfaces =	12.00	hr
				61.67	hr
			Total =	110.33	hr

WBS 2.0 Fuselage

	10 hr/ft x	5.42	ft =	54.17	hr

WBS 3.0 Empenage 1. Vertical Surface (no active control)
2. Vertical Surface (with active control)
3. Horizontal Surface

Vert w/Ctrl	10 hr/surf x	2	Surfaces =	20.00	hr
Vert wo/Ctrl	5 hr/surf x	0	Surfaces =	0.00	hr
Hoizontal	10 hr/surf x	0	Surfaces =	0.00	hr
				20.00	hr

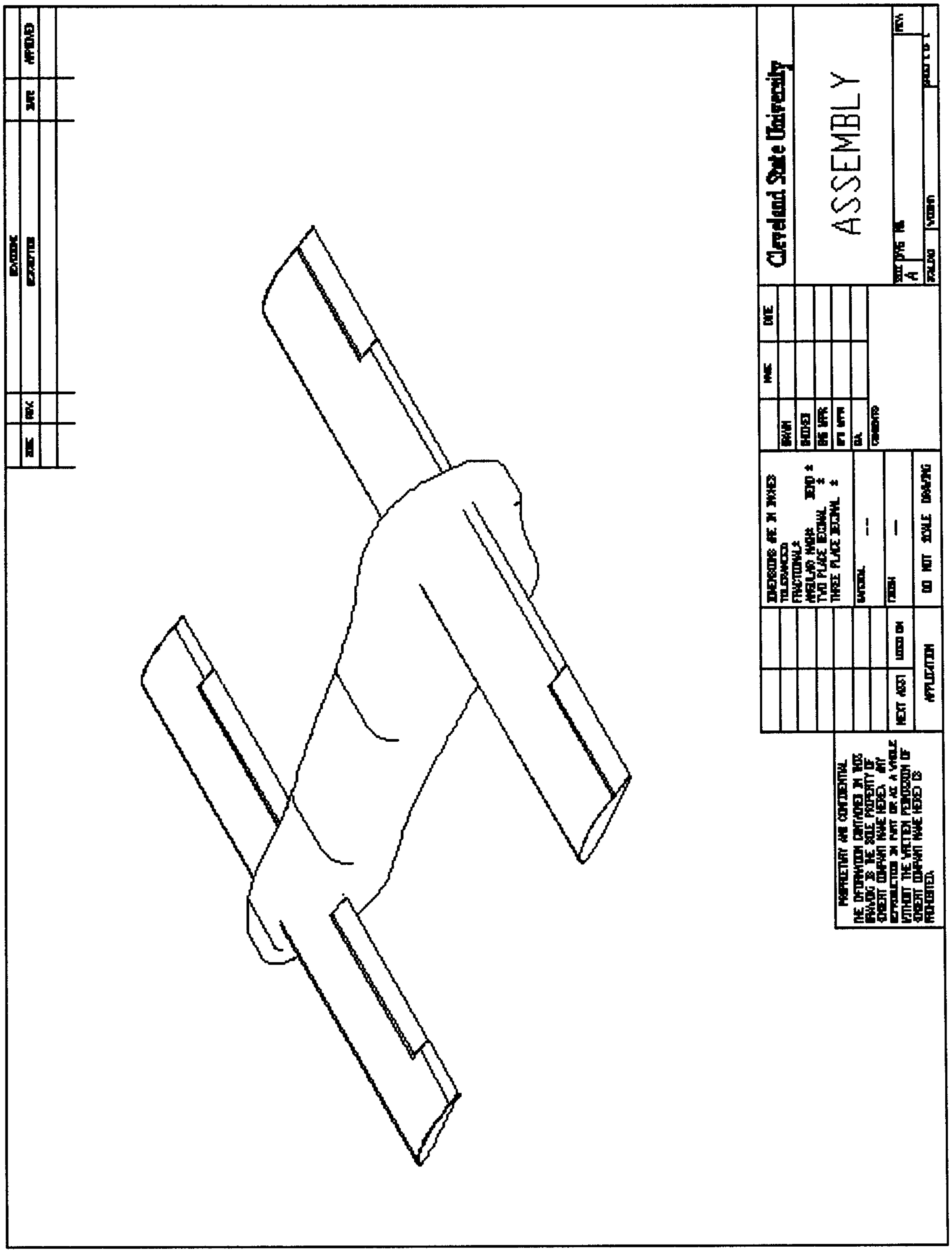
WBS 4.0 Flight Systems

	5 hr/control x	5	control =	25.00	hr

WBS 5.0 Propulsion Systems

	5 hr/engine x	1	engine =	5.00	hr
	5 hr/prop x	1	prop =	5.00	hr
				10.00	hr

Figure 5.6: Total Rated Aircraft Cost spreadsheet continued from figure 5.5, input data in green cells and output is displayed in the yellow cells.



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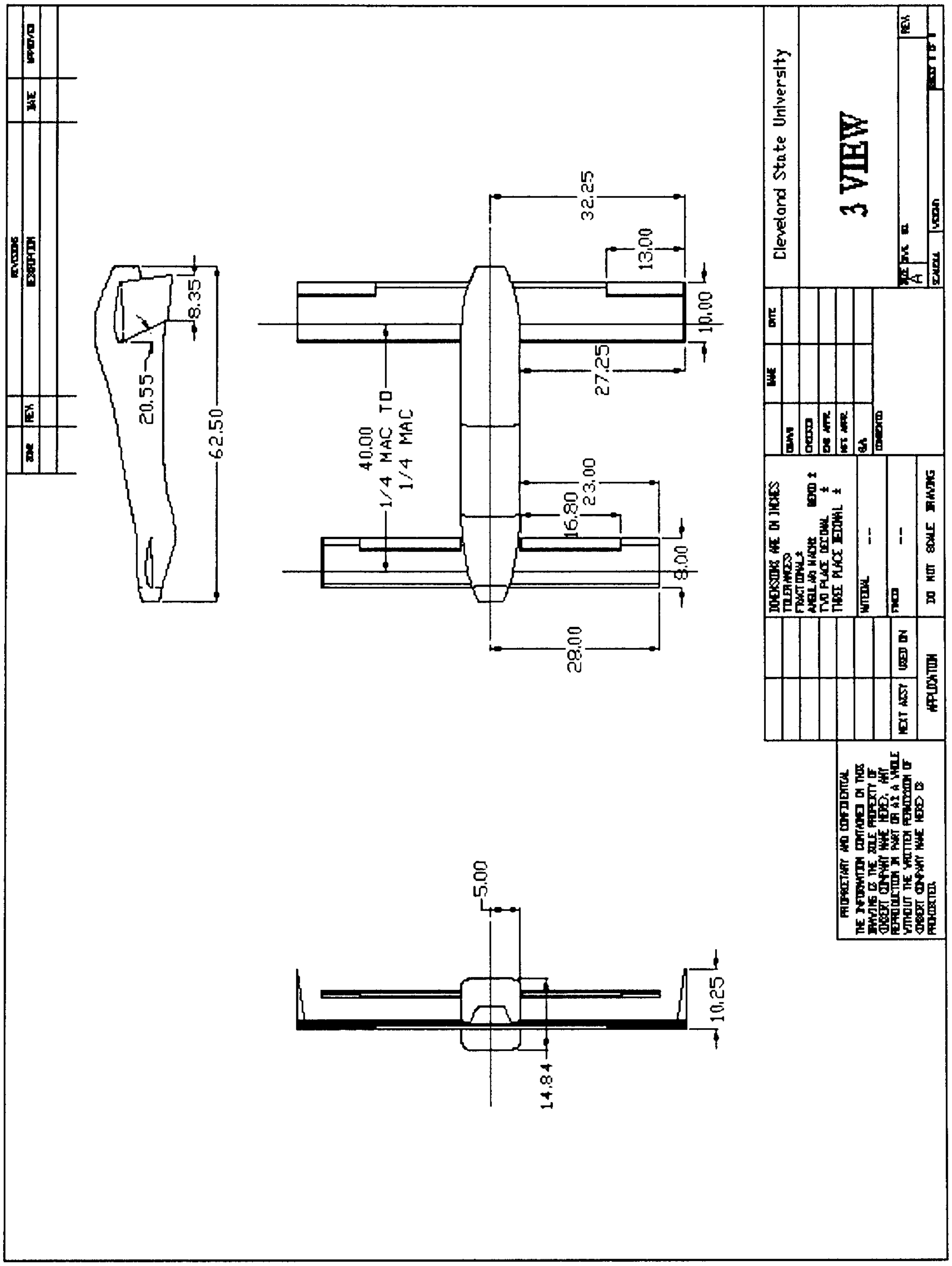

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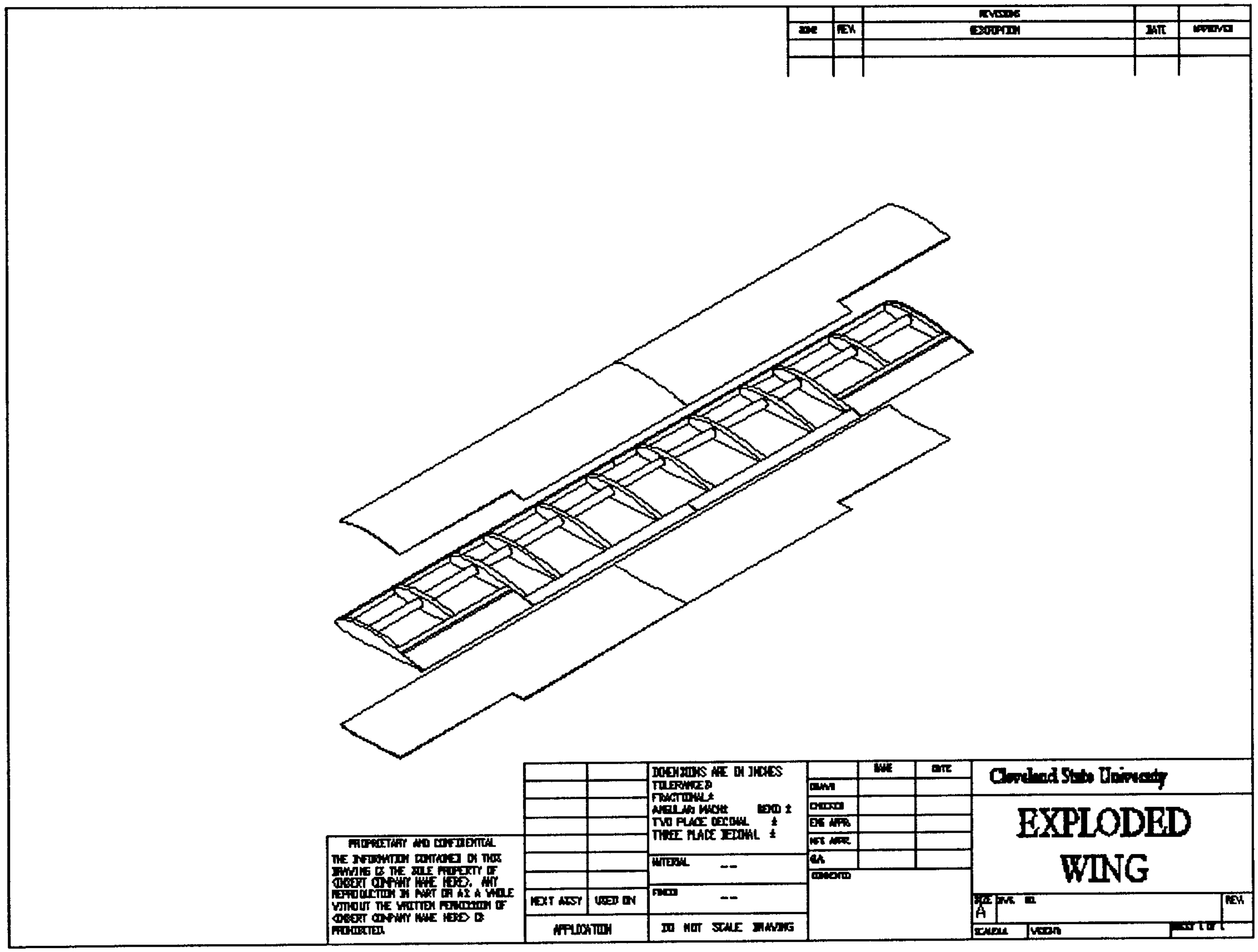
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6. Manufacturing Plan

The analysis and design work has been completed, a manufacturing plan must be devised for the project. The manufacturing plan entails procedures investigated and their figures of merits, and a manufacturing schedule. The following section outlines the design teams research and selection to construct the manufacturing plan.

6.1 Figures of Merit (FOMs)

A list of figures of merit was developed specifically for the manufacturing plan. This list helped the team select the type of processes it would use in the construction of the aircraft. The figures of merit were separated into four categories: Availability, Required Skill Level, Construction Time, and Cost. These FOMs were generalized during the development of the design. The manufacturing process was selected from the FOMs and the process that was determined to be the most suitable for the final design.

6.1.1. Availability

The availability of the materials and processes to the design team is a major factor during the manufacturing phase of the project. The design team approached each method of manufacturing and materials to be used with a pre-determined idea of the methods being used to manufacture the aircraft. The major component that determined the type of materials being used for construction was how available the material was to the team, if it wasn't readily available it was deemed to be an unusable material. Secondly, if the manufacturing processes or machinery wasn't available it was dismissed as a possible method.

6.1.2. Required Skill Level

To manufacture a component it takes a required skill level for each manufacturing process. If the process required a skill level above and beyond that of any member in the design team it was dismissed as a manufacturing process. If a process needed to be outsourced it was also automatically dismissed due to time constraints.

6.1.3. Construction Time

The time allotted for the construction phase was that of tight time constraint. The time available to the team for construction and testing was approximately one month. While investigating procedures, the team took into consideration the time available for the construction of the aircraft, and if the construction of an individual component were to take more than a week that process was rejected.

6.1.4 Cost

The final FOM that was a determining factor of the manufacturing process was the cost of the materials and any machinery needed for the construction of the aircraft. Due to budget constraints on the design team a budget plan was set forth to detail the funds available for the manufacturing phase of the project. Each process and bill of materials was drafted up and if any process were to cost above and beyond the manufacturing budget it was to be dismissed.

6.2. Investigated Processes

The design team investigated three methods for manufacturing processes. First, the team investigated the traditional style for R/C Modeling which is that of lightweight wood and cloth or plastic skin covering. Secondly, the design team considered using carbon fiber composites. Lastly, the team investigated using metal alloys as a material and process for construction.

6.2.1 Traditional Method

The traditional method is a method most common in the world of R/C modeling today. The materials used are typically balsa wood and cloth of plastic sheeting for skin. Alterations can be applied to use foam for the airfoil ribs in substitution for the balsa as well. These materials are inexpensive and readily available to the public. The required skill levels for this type of manufacturing process are low and can be easily taught if needed. The time to construct the aircraft using this method is very minimal in hours, and can be completed quickly.

6.2.2 Carbon Fiber Composites

The Carbon Fiber manufacturing process is alternative that is beneficial in most ways, but costly in others. This process involves using structural foams that provide shape for the components of the aircraft. The foams add to the compressive strength of the aircraft structure. The laminates used will be that of carbon fiber which add tensile strength to the structure as well. The availability of the materials is limited and vendors are difficult to locate for them. The cost on the materials and manufacturing processes involved are also costly and add to a designer's tight budget. The foam cutting requires a skill level knowledge of how the foam is to be cut and the laminates require special handling. The construction time of such a method can be lengthy and tedious.

6.2.3 Metal Alloys

Using metal alloys such as aluminum as an approach to the construction add to the integrity of the structure. Furthermore, aluminum is also readily available material source as well. The disadvantage is that of weight, cost and construction time. Using an alloy though generally considered a light substance is a weight disadvantage when approach this project. The costs of the materials are also expensive compared to other manufacturing method. Finally, the manufacturing of the aircraft using metals will be lengthy and requires a great amount of skills to form.

6.3 Results

As dictated by the figures of merit outlined in this section a manufacturing process was chosen. The design team chose to construct the aircraft using the traditional methods with the alteration of foam as the airfoil ribs instead of the balsa wood. This design will be lightweight and very cost efficient with respect to the budget. The skills required for this method are minimal and any difficult task will be taught.

6.3.1 Wing Assembly

The wing is to be constructed and assembled using foam for the airfoil ribs and a carbon fiber rod to support each rib. The carbon fiber rod is lightweight and will be assembled through the centers of the ribs and adhered with epoxy. The airfoils will be cut with a hot wire foam cutter in which a wire is heated and ran through the foam to cut it. Since, a rectangular wing configuration is being used this process will be a generally easy task to complete. Once the ribs have been cut and fixed to the carbon fiber rod, the wings can be sheeted with their skin. Before this can take place, the servomotors must be framed in and mounted. When the sheeting commences the servos will be cut out. The sheeting will consist of a layer of balsa wood adhered to the wing configuration with a sheeting of plastic over it to add to the integrity of the structure.

6.3.2 Fuselage

The fuselage will be constructed with a mix of plywood and balsa. The plywood is used for the higher stressed areas and the balsa for the less stressed areas and the sheeting. The payload carriage will be constructed using PVC pipe shells. All of the controls, battery cells, and motor will be fastened in their appropriate places using bolts and epoxy. The landing gear will consist of piano wire in a triangular formation. The finished fuselage will be sanded the covered with sheeting to add to the aerodynamic quality of the structure.

6.3.3 Manufacturing Schedule

The following timeline list the components of the aircraft being constructed, the duration allotted for construction, and their respective milestones. Figure 6.3.1 displays the orderly fashion to which these tasks were most important.

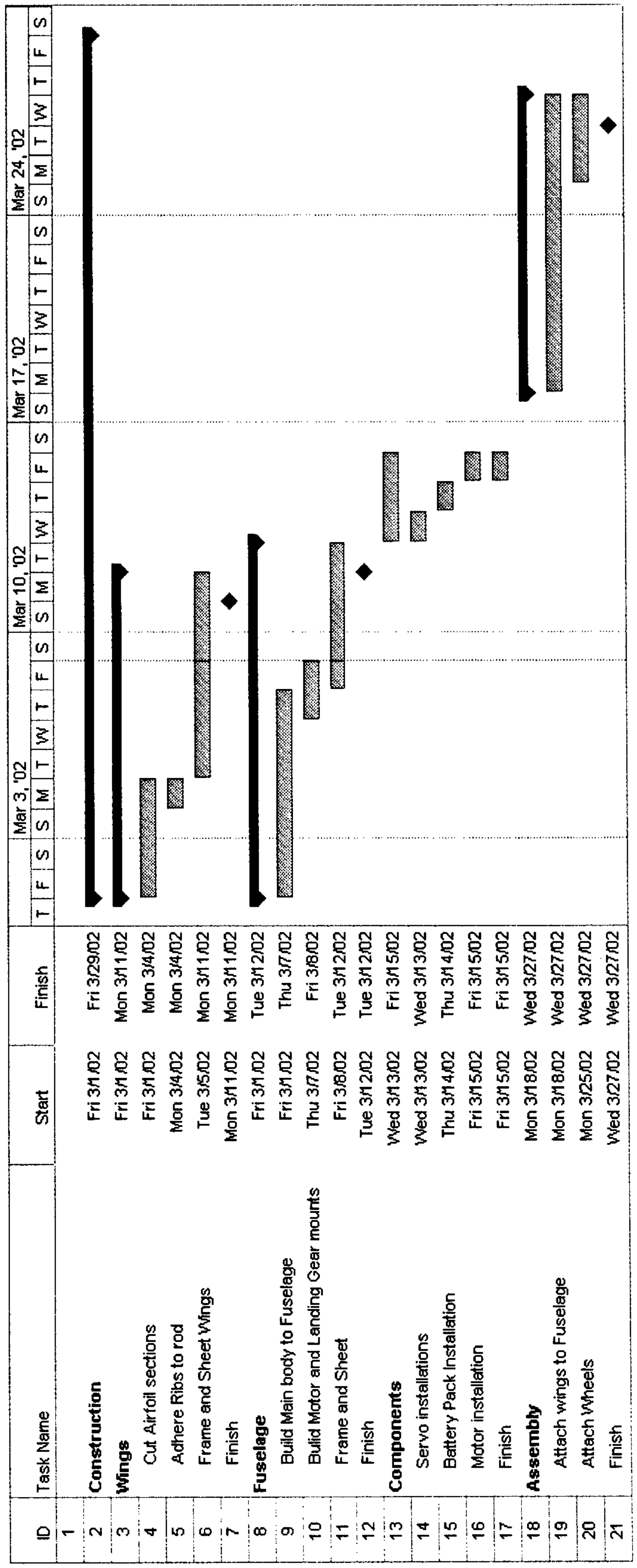

Figure 6.1: Manufacturing Timeline

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[1] Lennon, Andy, R/C Model Aircraft Design, 1996, Air Age Inc.

[2] Fox, Robert and McDonald, Alan, Introduction to Fluid Mechanics, 5th Edition, 1999

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2001-2002

AIAA Student Design/Build/Fly Competition

Turkish Air Force Academy

"CONQUEROR"

Designed by :
Mustafa ILLEEZ
Ferdi METE
Sıtkı AKSAN
Kürşad HARMANCI
Erkan BENLI

Table of Contents

1. Executive Summary

The project "Conqueror" began as the license (bachelor's degree) thesis of Mustafa ILLEEZ and Ferdi METE. The topic of the thesis was designing an unmanned aircraft and the project advisor proposed AIAA's design/build/fly competition requirements to be taken the same as the ones we would use in our project. Then, Mustafa and Ferdi decided to attend to competition because it would be a good opportunity to present their thesis and would surely be a good motivation. However, two undergraduate students would most probably be unable to finish Conqueror on time, since they had no experience on producing phase and the deadline for presenting the report was only six months ahead. Three very skilled members of school's model airplane club: Sıtkı AKSAN, Kürşad HARMANCI and Erkan BENLİ could be the best ones to take mission in such a team and that is what they did.

After founding the team next step would be a very timely and detailed planning. Because our team was the group which had the least members. That was a disadvantage certainly every member would study much more when compared a member of a more crowded group. However it would be much easier to communicate and meet in a smaller group. We surely needed a very careful planned and timed program to follow that was what we did at the beginning. Group decided on the tasks to be completed and everybody was given a special task to complete in a pre-determined time.

1.1. Conceptual Design

1.1.1. Design alternatives

At start up the design team members proposed very different configurations to complete the mission and perform the demands of the competition. We had options of wing planform, wing configuration, wing vertical location, airfoil, fuselage shape, tail arrangement.

In wing planform decision phase we had rectangular, elliptical, tapered and non-linear wing planform alternatives. Certainly we had to decide if we would configure our plane as a monoplane or biplane. For a biplane configuration wing placing was also a very important task to be concerned. While we were making analyses on different candidate designs we had to decide on wing vertical location. Would it be a high, low or mid wing location. Certainly it was very important to choose the most suitable airfoil for "Conqueror". That was one the subject that team members hesitated to determine. Fuselage shape was also a very important factor for the performance of the aircraft. We had three main ideas about it: a rectangular shape, an elliptical shape, and a geometry like two(or more) joint balls. That was not all about the fuselage, we also had to find out the most suitable configuration of placing the balls in it. Because the fuselage geometry would be dependent on it. Team members would have to decide on type of tail. We had four main options: v-tail, t-tail, boom-mounted tail, conventional tail. Design of landing gear, propulsion system and determination of propeller were discussed in preliminary design phase.

As it is listed briefly above there were many possibilities to be taken into account so our limitations in capabilities, limitations of the competition and the need to choose the most suitable aircraft with ease in construction and with least cost possible determined the final configuration. Team always had to find the optimal solutions to problems that were encountered in design phase.

1.1.2. Design tools overview

Our school had never attended such a competition so we didn't have a background specialized on AIAA Competitions. However the academic education taken by the members was a satisfactory for such a project. Team members first decided to find out information about the "design" in general and the footsteps of it. In addition to that logically used previous years reports since it would be beneficial to see what was done to be successful. After Mustafa and Ferdi decided to start the conceptual design they found out that it very important to have some useful softwares to make correct and very fast calculations. The design alternatives listed above were analyzed with the help of softwares below:

- Nvfoil
- Microsoft Excel
- Yukselen
- Autocad R14

Yukselen: Yukselen was a program coded in microsoft developer by Prof.Dr. Adil YUKSELEN. It had great help to us to compare different wing planforms by using Prandtl Lifting Line Theory. It had supplied us the lift distribution and compare it with elliptical lift distribution. We could change the geometric inputs and even compute for a certain airfoil. Stall characteristics for a certain wing could be obtained with this program and we were able to see whole output in graphics and see on the planform shape as well.

Nvfoil: It is a software for the calculation of the principal aerodynamics parameters of airfoils; they may be defined analytically or by lists, and are studied in a 2-D non viscous flux, at an angle of attack assigned, and at M = 0. It's possible to assign the geometry of the airfoil analytically (E.g.: airfoils Naca 4-digits or 5-digits, or elliptic), or reading by lists, or copying from other applications. In those events, the additional points are obtained by a cubic spline interpolation, using a Chebishev distribution, that is gathering them around the leading and trailing edge. At user's order, or at every data variation, the airfoil geometry is updated, e then the (Douglas-Neumann or Oeller routine is ran, for the calculation of Cp, at the angle of attack desired. Then, the aerodynamics integral parameters, Cl, Cm (in respect of 1/4 of the chord) and Cd (for a valuation of the numeric errors), are calculated. It is also possible to correct the obtained results with the effect of the compressibility, of the boundary layer, or of a plain flap at the trailing edge.

MS-Excel: Excel is a very common program used for different purposes. Three of team members had used excell especially in aerodynamics I and II, flight mechanics, hydraulics lessons. During the conceptual design phase we used excel sheds and macros developed for "Conqueror" very much. From the previous years contest reports we were able to acquire some statistics to give us an idea about the size and characteristics of attendant planes with the help of excel sheets arranged. We used excel books developed to calculate first sizing of alternatives, compare them and obtaining some necessary data about the aerodynamic characteristics of different configurations. Even the wing placing for biplane configuration was able to be analyzed with our excel sheets. In our project excel software is used extensively.

Autocad R14: Autocad was used to draw design alternatives in three dimensions in order to see and compare the alternative configurations. Those drawings had great help to team while explainig the ideas during the meetings and in that way we could see what each member thought for the concept.

Aircraftdragestimator: Aircraft drag estimator was a software which requires data about the geometry and flight of aircraft in order the compute drag of whole aircraft. Since it would not be easy to calculate drag for each part of the airplane it gave us an idea quite reliable. Besides that this program was able to calculate the necessary power required for flight. Certainly it wouldn't be an exact calculation but we had an estimation at least with a safety factor outputs of this program became very useful in conceptual design phase.

1.1.3 Results:

The programs listed in design tools overview, limitations of teams capabilities and contest rules, certainly the judgement of team members were the basics of choices made "Conqueror" appear. These basics analyzed different wing, tail, payload, propulsion system combinations and the plane which might have the best score was an elliptical mono wing design. However, team members which were responsible of production objected to that decision because of the time remained was not enough for such a construction. A tapered, high-mono wing and a rectangular biplane configuration were chosen as last two candidate designs. Both of them would be designed to carry 24 balls. The placement of the balls was chosen as 2x12 (two rows) Determining the propeller was out of question in conceptual design phase and no detailed calculations were made on landing gear.

1.2. Preliminary Design

Team completed three main tasks during preliminary design. One of them was deciding the airplane among two alternatives. Others were design of landing gear and choosing the proper propeller. Besides them some important subjects about the construction methods and materials were made in preliminary design.

1.2.1 Design Alternatives

There were two main alternative configurations selected during conceptual design. A biplane with rectangular wings and a monoplane configuration with tapered wings. For propeller we had tried many different propellers with the help of programs "prop selector", "extended prop selector" and analyzed propellers of similar planes. For the design of landing gear we analyzed landing gears of similar planes and we had used some different books which are listed in references section. Proper and most efficient propulsion system was designed and analyzed with the help of software explained in design tools part.

1.2.2. Design tools overview

Softwares used during preliminary design are listed below:

- Motorcalc 6.04
- MS Excel
- Prop Selector
- Extended Prop Selector
- Autocad R14

Motorcalc 6.04 : It is a program for predicting the performance of an electric model aircraft power system, based on the characteristics of the motor, battery, gearbox, propeller or ducted fan, and speed control. You can specify a range for the number of cells, gear ratio, propeller diameter, and propeller pitch, and it produces a table of predictions for each combination. Program will predict weight, current, voltage at the motor terminals, input power, output power, power loss, motor temperature, motor efficiency, motor RPM, power-loading, electrical efficiency, motor RPM, propeller or fan RPM, static thrust, pitch speed, and run time. By producing a table of predictions, it lets you determine the optimum propeller size and/or gear ratio for your particular application. It can also do an in-flight analysis for a particular combination of components, predicting lift, drag, current, voltage, power, temperature, motor and electrical efficiency, RPM, thrust, pitch speed, propeller and overall efficiency, and run time at various flight speeds. It will also predict stall speed, hands-off level flight speed, optimal level flight speed, maximum level flight speed, rate of climb, and power-off rate of sink. MotoCalc's graphing facility can plot two parameters against any other (for example, lift and drag vs. airspeed). By using this highly developed software we found out the most suitable motor and cell combination.

MS Excel: Two design alternatives were compared with excel sheets based on the scoring of the contest. Their estimated weights and preliminary sizing of their geometry were taken into account. While making those estimations the materials that would be during production were

discussed and decided. Excel was also used to find out the landing gear configurations used for aircrafts those have similar geometry.

Propselector and **extendedpropselector:** Although propeller selection could be made by using motorcalc 6.04 software team wanted to compare the results of that software with propselector and a more detailed version of it, extendedpropselector.

Yukselen: Yukselen was a program coded in microsoft developer by Prof.Dr. Adil YUKSELEN. It had great help to us to compare different wing planforms by using Prandtl Lifting Line Theory. It had supplied us the lift distribution and compare it with elliptical lift distribution. We could change the geometric inputs and even compute for a certain airfoil. Stall characteristics for a certain wing could be obtained with this program and we were able to see whole output in graphics and see on the planform shape as well.

Aircraftdragestimator: Aircraft drag estimator was a software which requires data about the geometry and flight of aircraft in order the compute drag of whole aircraft. Since it would not be easy to calculate drag for each part of the airplane it gave us an idea quite reliable. Besides that this program was able to calculate the necessary power required for flight. Certainly it wouldn't be an exact calculation but we had an estimation at least with a safety factor outputs of this program became very useful in conceptual design phase.

Autocad R14:

1.2.3. Results

The mono plane configuration was chosen and decided to be the best configuration and it was expected to take the highest score. For landing gear a tricycle type is chosen. Third cycle is placed in front of the main gear for better landing performance and easier handling during taxi and take off. It was not easy sometimes to find necessary or the most suitable material. For example while choosing electrical motors for "Conqueror" team made very detailed analyses of propulsion system and battery combinations. However we used the motor we could acquire. Some of our calculations were made for further analyses and they were thought to be good references for next year's Design/Build/Fly Contest team.

1.3. Detailed design:

The aircraft chosen to be the one to build was optimized during detailed design phase. One of the most important concerns of this design phase to design the airframe and make structural analyzes of main components. Certainly the aircraft performance was calculated and optimized in order to fit the requirements of the contest. Stability analyzes were performed in order to find out if the flight quantities of "Conqueror" were proper for a safe flight.

1.3.1 Design alternatives

Wing and tail area, position of the wing (longitudinal), empennage sizing, wing and tail location according to the fuselage, placement of propulsion system and other components, airframe structure sizing were the main topics of design alternatives during detailed design phase. Propulsion system analyzes were done again because there were changes to be made after detailed design in the geometry of the "Conqueror". Besides that team was aware of the fact that in this year's scoring battery weight would effect rated aircraft cost so it was very important to find calculate the correct engine/battery configuration. Besides the software "Motorcalc" experiments on engine/cell outputs were made. Flight performance of "Conqueror" is reanalyzed by using the data obtained during optimization phase in detailed design.

1.3.2. Design tools overview

The softwares used in calculations made during detailed design are listed below:

- Stability estimator
- MS Excel

- Motorcalc
- Winfoil 2.2
- Autocad R14
- Yukselen

Stability estimator: we calculated the pitch stability, required power values with the help of this very useful software.

MS Excel: We used again excel program for sizing and computing the final geometry of wing and tail with excel. Flight performance of "Conqueror" is analyzed again with excel sheets prepared during conceptual design.

Motorcalc: We had used this software to find the proper combination of engine and battery during conceptual design. Since we would not be able to change the motor of "Conqueror" we made analyses with this software to find the minimum battery weight for a ten minutes safe flight.

Winfoil: This software was so one of the design tools of conceptual design. During detailed design phase it is used for the frame design of the wing and tail.

Autocad: We had used this software during preliminary design to see two different candidate designs better and compare them more objectively. During detailed design, three dimensioned (3D) drawings of last configuration were prepared and detailed two dimensioned (2D) drawings especially for great help of the detailed drawings in building phase.

Yukselen: this software is used again because of the possible changes in wing geometry. Best taper ratio which will be the closest to elliptical lift distribution in general is analyzed with this program and stall characteristics of the wing became significant.

1.3.3. Results

"Conqueror" was ready to be built after the detailed design. Structural analyzes showed that it would have safe endurance characteristics. Final flight analyzes showed us that designed aircraft will have proper flight characteristics for the contest's flight mission. The stability analyzes proved the plane to have good handling properties. All the calculations and data obtained during conceptual, preliminary and detailed design were checked again and the results were discussed finally. It is decided that design was ready to build.

2. Management Summary

Mustafa İLLEEZ and Ferdi METE were the senior cadets in Turkish Air Force Academy Aeronautics Department. As graduation thesis and to take their bachelor's degree they were asked for designing and building an unmanned aircraft (UAV). Advisor of the thesis Prof.Dr. Süleyman TOLUN proposed to use the requirements of design/build/fly contest of AIAA as characteristics of the project. After that it was decided to attend to competition. Mustafa and Ferdi were inexperienced on the production and flight of R/C model planes. Fortunately our school had a model airplane club and three members of the club were included to team. Sıtkı AKŞAN was a senior and Kürşad HARMANCI was a junior cadet in aeronautics engineering. Erkan BENLİ was a sophomore in electronics engineering.

2.1. Architecture of the design team

As mentioned in executive summary team formed to build "Conqueror" was consist of the fewest students among the teams attending the contest. Team came together just after being formed for the purpose of designing the plane which would be able to get the highest score. Certainly one of the major limitations of the design was the production capabilities of our model plane club where "Conqueror" would be constructed. Another important limitation was the approaching deadlines for proposing the written reports and being ready for flight contest. Because of those facts team decided to form a division of assignment areas a very effective but not very classical one. A very good planning was done and everybody had to do his duty

perfectly on time. Since Mustafa, Ferdi and Sıtkı were roommates and classmates they were almost twenty four hours together. Besides that every week whole team came together checked assignments completed and distributed new missions if necessary. If necessary decision are made and problems and new ideas are discussed. Team agreed on that that type of brain storms would be very helpful for the developing and executing new ideas in the project. Another method used to make the team work more efficiently was that every member was volunteer to duties that he was skillful at. Because of that instead of an assignment oriented organization a member oriented one was preferred. One of the greatest advantages of the group was the perfect coordination and cooperation. Everyone had to work hard and every member was ready for it.

Mustafa İLLEEZ : He was the leader of the group he had formed the team and informed the school administration about the project. He also got in touch with other teams participating the contest for information exchange especially to learn about softball. He downloaded necessary softwares for the team from internet or school computer network.

Ferdi METE: He was always with Mustafa and helped him to complete his tasks. He prepared all the drawings during preliminary design, detailed design and rearranged them while the written report was getting prepared. His creativity helped the team a lot when unexpected problems were encountered.

Sıtkı AKŞAN: As he was the administrator of model plane club his relations with the members of the club caused Kürşad and Erkan to be in the team. Besides that his personal relations with the companies selling materials for small aircrafts like ours.

Kürşad HARMANCI: Besides his great effort in manufacturing process he designed and analyzed landing gear of "Conqueror". In design phase he always reminded the team the limits of manufacturing capabilities.

Erkan BENLİ: His talent and experience of manufacturing and flying R/C models had great help to complete the manufacturing process on time. Since he was an electronics engineering student he was responsible of whole propulsion system. He was the pilot of "Conqueror".
A detailed table of assignment of team members will be presented in Table 2.1.

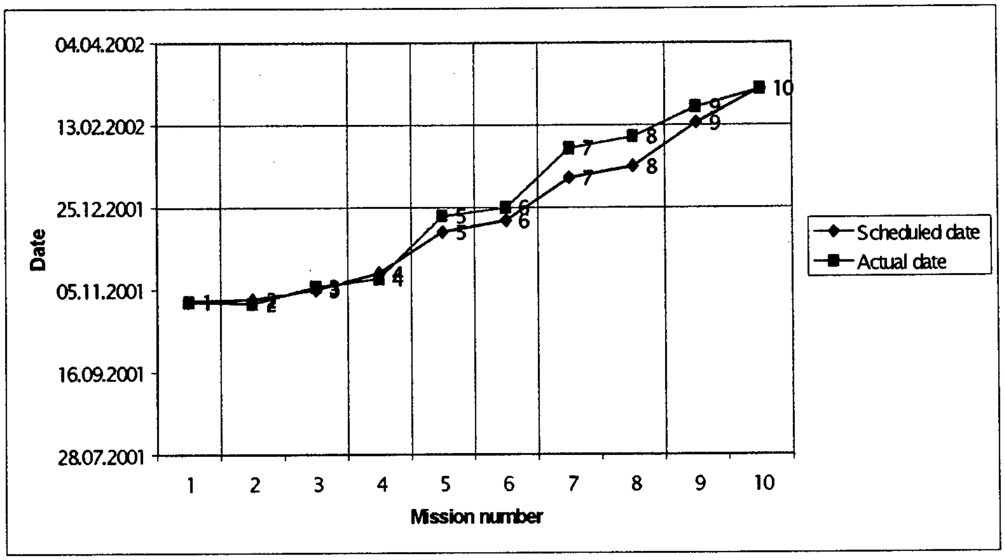
Mustafa İLLEEZ	Ferdi METE	Sıtkı AKŞAN	Kürşad HARMANCI	Erkan BENLİ
Aerodynamics • Wing design • Airfoil selection • Tail design Structure • Wing endurance	Aerodynamics • Wing design • Fuselage design Structure • Wing endurance	Logistics • Material selection • Obtaining necessary materials Production	Aerodynamics • Design of landing gear Structure • Landing gear endurance Production	Propulsion • Motor and battery selection Flight tests Production

Table 2.1. Assignment division in general

However it must be added that those were the main responsibilities of the team members. Each member of the group had to gather information about everything useful and related to design and manufacturing. Members helped each other and told their opinions directly to each other. Decisions related to contest were taken in weekly meetings. For example if the subject of the meeting was landing gear Kürşad (because his assignment was it) explained his ideas, his alternatives and his requirements. After that team's brain storming and decision process begins.

2.2. Schedule for assignments

Sıtkı, Erkan and Kürşad had experience on manufacturing so they could estimate a specific time while the team was preparing a timetable for the missions to be completed. However lack of experience made the follow the footsteps of other teams reports of whom were in contest site. Certainly out schedule would have to be different because of exams and holidays. Team members completed some of the tasks even before their deadlines but unfortunately some delays occurred unexpectedly about obtaining the necessary materials.



	Event and Number	Scheduled Event Completion Dates	Actual Event Completion Dates
1	Application for the contest	29.10.2001	29.10.2001
2	Data collecting for the concept of design	30.10.2001	28.10.2001
3	Analyze previous succesful design	05.11.2001	07.11.2001
4	Determinination of alternatives and figures of merits	15.11.2001	12.11.2001
5	Conceptual design	10.12.2001	20.12.2001
6	Determinination of alternatives and figures of merits	17.12.2001	25.12.2001
7	Preliminary design	12.01.2002	30.01.2002
8	Determination of alternatives and figures of merits	19.01.2002	06.02.2002
9	Detailed design	14.02.2002	24.02.2002
10	Preparation of the written report	07.03.2002	07.03.2002
11	Obtaining necessary materials for manufacture	15.02.2002	

Figure 2.1. Design and Development Milestone Chart
A schedule of project completion deadlines was provided at the first group meeting. The actual completion dates of the events are depicted in the same figure. As we can see above actual time of completing the tasks were very close and sometimes even before the deadline. However 11th event was obtaining the materials of construction parts which would be bought from abroad countries were not acquired yet while the proposal report was being written.

3. Conceptual Design

In conceptual design team determined the main parameters of design and main alternatives for the design were inspected. Those alternatives of components were combined in order to obtain different plane configurations. After that those alternatives were analyzed and two possible configurations were chosen for further studies of preliminary design phase. Excell sheets developed to compare different configurations' estimated rated air craft cost, flight score and total score. Aerodynamic efficiency and ease of manufacture is also considered.

3.1. Design Alternatives

Determination of design alternatives was almost ready from the first meetings of the team. In order to make a proper assignment division, assignments would have to be determined and they were the design parallel to design parameters in general. First those parameters in general analyzed and some judgements were made and than combinations of those judgements became our candidate planes. Candidate planes were compared within the pre-determined criterions in other words figures of merit. In that way the alternatives which would be eliminated and which would be used at further analyzes became distinct.

3.1.1. Wing planform

Aerodynamically the major component of the plane are wings. It is certain that wing planform has significant effect on the flight characteristics of the airplane. Three general planforms namely: elliptical, tapered and rectangular and few non-linear planforms were considered during conceptual design phase.

At the beginning team decided that a sweep angle would not be necessary because the cruise speed of such an aircraft will not require such a configuration. Besides that wing twist is not considered in this project.

Rectangular planform was considered because of the ease in construction and it was easier to make calculations on a planform that has constant chord length. However team was aware of the fact that lift distribution of a rectangular planform was different from an elliptical lift distribution.. According to literature (for further information look references section) a wing that has untapered (rectangular) wing has 7% more drag to lift than an elliptical wing of the same aspect ratio.

A tapered wing is easier to manufacture than an elliptical wing. With determining a correct taper ratio it was possible to have less than 1% higher drag to lift than an elliptical wing. In last years contest some of the teams eliminated tapered wing at the very beginning of the design because of the claim of poor handling characteristics. However no scientific data was given. To find the proper taper ratio and analyze stall characteristics of that planform program Yukselen was used.

An elliptical wing is aerodynamically the most efficient wing. However because of the handicap of manufacturing difficulties very few of planes could have elliptic wings.

A tapered wing is easier to manufacture than an elliptical wing. With determining a correct taper ratio it was possible to have less than 1% higher drag to lift than an elliptical wing. In last years contest some of the teams eliminated tapered wing at the very beginning of the design because of the claim of poor handling characteristics. However no scientific data was given. To find the proper taper ratio and analyze stall characteristics of that planform program Yukselen was used. Team agreed on that a careful analyze would make "Conqueror" able to have good stall characteristics.

Delta wing and tandem were also some of the alternatives but because of the lack of the experience of manufacturing team and the general opinion of considering those planforms not to be necessary to meet the requirements of the contest they are not analyzed further. Some non-planar wing planforms were also considered however at the end of conceptual design they were eliminated because of having no time and no experience to make necessary experiments and calculations for extreme aircraft configurations. Such a great attempt would be above the aim of such an undergraduate student study.

3.1.2. Wing configuration

We had two main alternatives for wing configuration, except non-planar, delta and tandem wing configurations which were eliminated at the end of conceptual desig, a monoplane and a biplane configuration.

When a monoplane which had the same total wing area as a biplane was considered it would have 41% greater span. That would provide a reduction in drag due to lift about 31% when compared to the biplane of equal wing area. Briefly, a monoplane configuration would be aerodynamically more

efficient.

However, biplane had its own advantages, too. Our plane would fly in low speeds and a biplane would provide that speed without complicated highlift devices. In our judgement the most important benefit of biplane was thought to be its span. It would not have excessive span and that would provide structural efficiency.

Since a safe flight was always a figure of merit we decided to make more detailed analyzes for wing configuration. Decision would be clear during the preliminary design phase.

3.1.3. Wing vertical location

This alternative was only for mono wing configurations. We had three main alternatives to locate the wing. Low, high and mid wing configurations were considered.

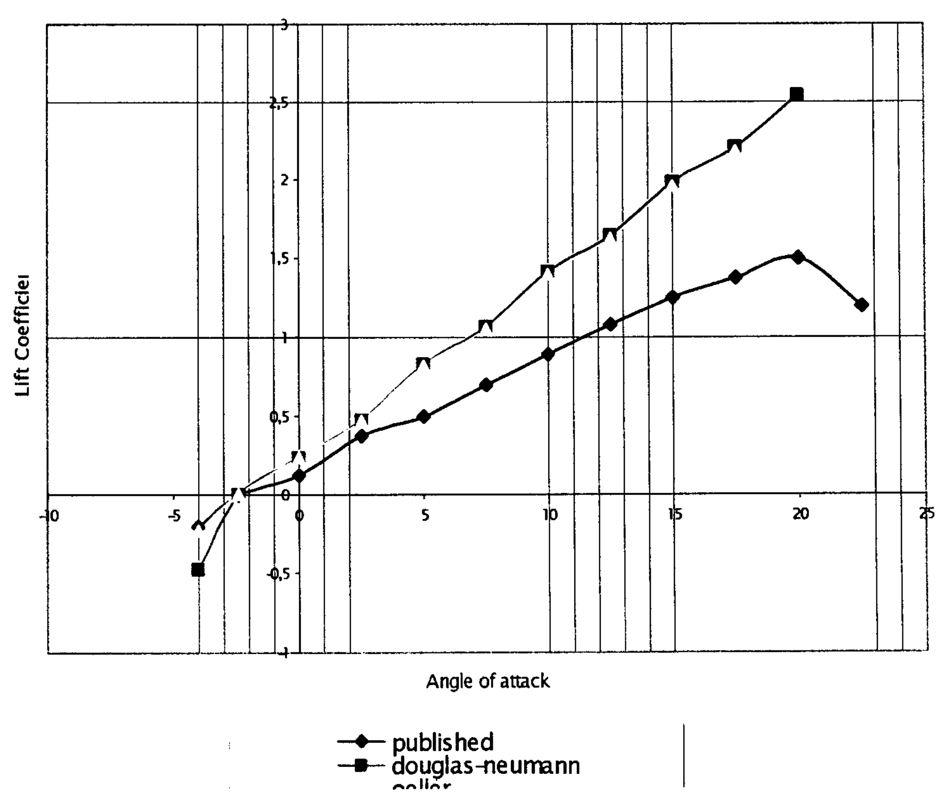
Contest flight performance requirements indicated that except for 360 degree turns our plane was not asked for great acrobatic maneuverability. This acceptance and the structural problems to be considered lead the team eliminate the mid wing configuration.

With a low wing configuration break downs were probable to occur after a bad landing. If the propellers were located above the plane of the wing to minimize landing gear length, interference effects of the wing and propeller would increase. Those were some of the reasons for eliminating low wing location.

3.1.1. Airfoil selection

In the literature presented in references section six airfoils were recommended and mentioned as the most popular airfoils used in R/C models. They were Clark Y, Göttingen 389 Davis, N.A.C.A. 2409. Besides them because of being a flat bottom airfoil PT-40 was also added to our analyzes. Nvfoil software was used to make calculations and it contained two options to perform the analyzes one of them was Doglas-Neumann Method and the other was Oeller Method. However, they had given different results when compared with the published data especially at high angle of attacks. We couldn not obtain any data about the stall characteristics of airfoil with Nvfoil program. As published data, "Summary of Low-Speed Airfoil Data Volume III" written by a group authors from Department of Aeronautical and Astronautical Engineering of University of Illiniois at Urbana-Champaign (U.I.U.C.) and a graphic of Massacuhetets Institute of Technology (M.I.T.) were used for different airfoils.

NACA 2409



Göttingen 389

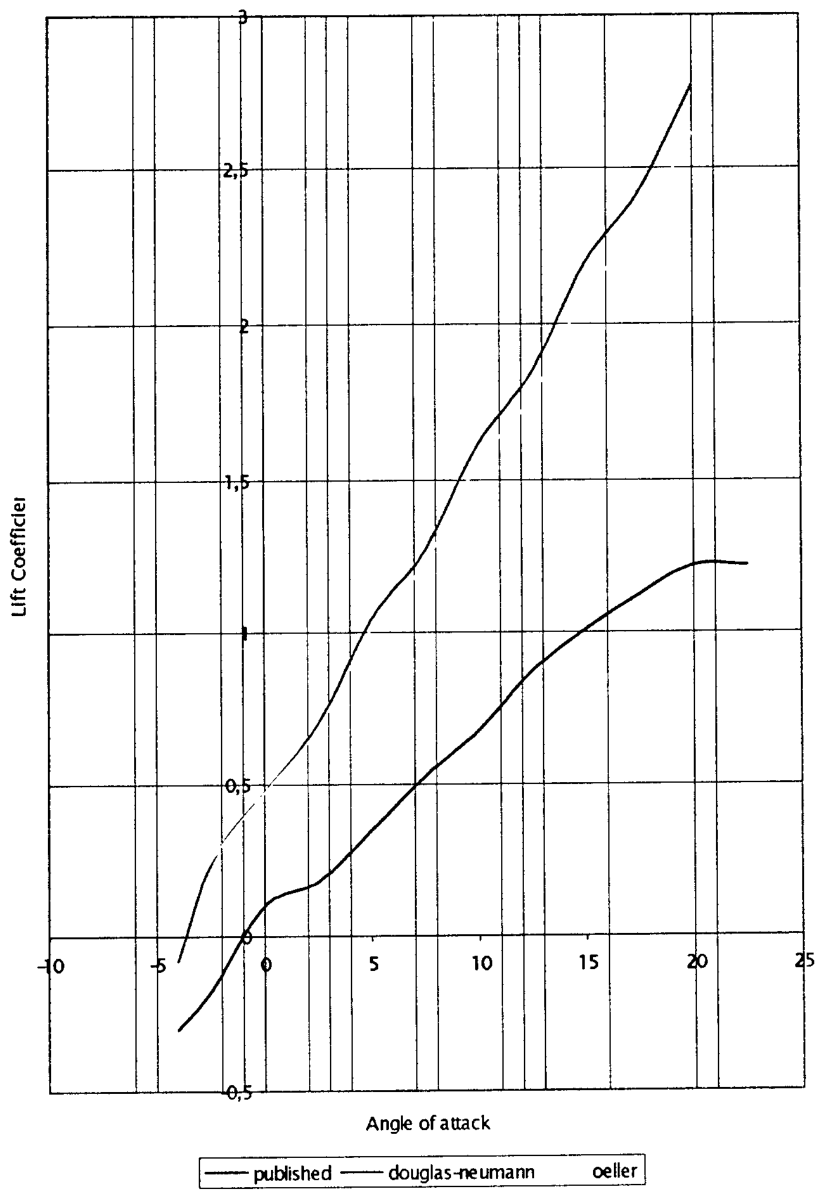


Figure 3.3. Graphic of lift coefficient and angle of attack for Göttingen 389.

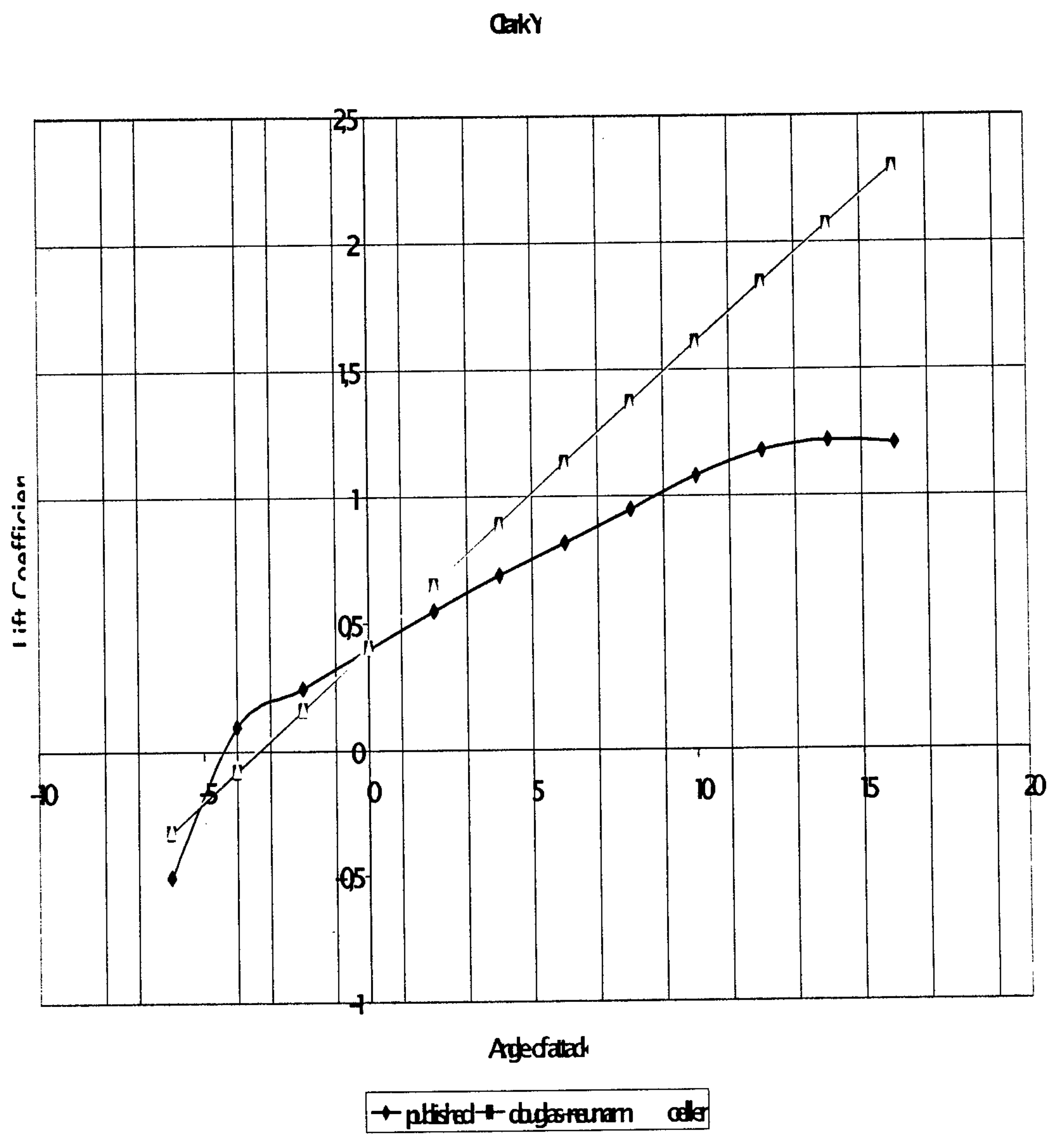


Figure 3.4. Graphic of lift coefficient and angle of attack for Clark Y.

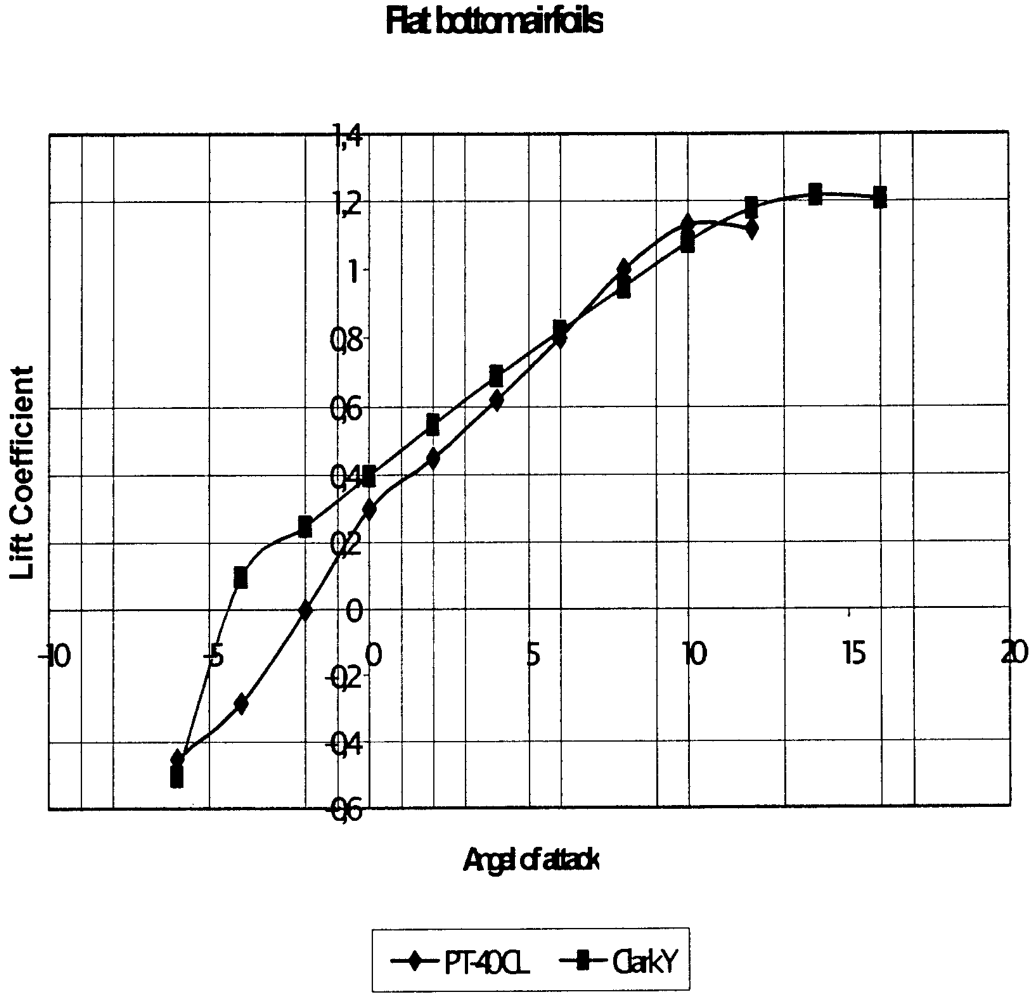


Figure 3.5. In our final graph we compared two flat bottomed airfoils. Although they had similar lift coefficients at low angles of attack Clark Y was a little better and stall characteristic of PT-40 was very sharp and it would stall at a lower angle of attack.

Finally team members decided to use Clark Y as wing airfoil of "Conqueror". Its aerodynamic efficiency and previous experiences of manufacturing department with that airfoil were effective issues on that decision.

Clark Y had been used in aviation extensively for decades and it had long been a member of the successful group of airfoils used on R/C aircraft. It had a flat lower surface and 12% thickness. Those specialties would ease the wing construction significantly. Our plane would require a moderately high lift coefficient and Clark Y would supply low drag at $0,5<C_L<0,75$.

3.1.2. Fuselage shape

We had three main alternatives for shaping the frontal surface of fuselage. First was an elliptical geometry. Second alternative was a rectangular geometry which was the easiest one to manufacture. Fourth was a geometry formed by the rows balls. Third alternative for the fuselage shape was covering the fourth one in order to make stronger and decrease the skin friction. Extra drag caused by the increase of frontal area would be lessened by the decrease in skin friction. Rectangular fuselage geometry was eliminated because of poor aerodynamic efficiency and leaving too much unnecessary space that was forming extra area and drag. Fourth one was creating extra skin friction and was not a proper option when location of cells and other flight systems were considered. Among those alternatives the third one was chosen by the team. Because of choosing a high wing configuration, fuselage would have to meet greater loads and endurance would be a dominant factor. It was stronger than the elliptical fuselage option.

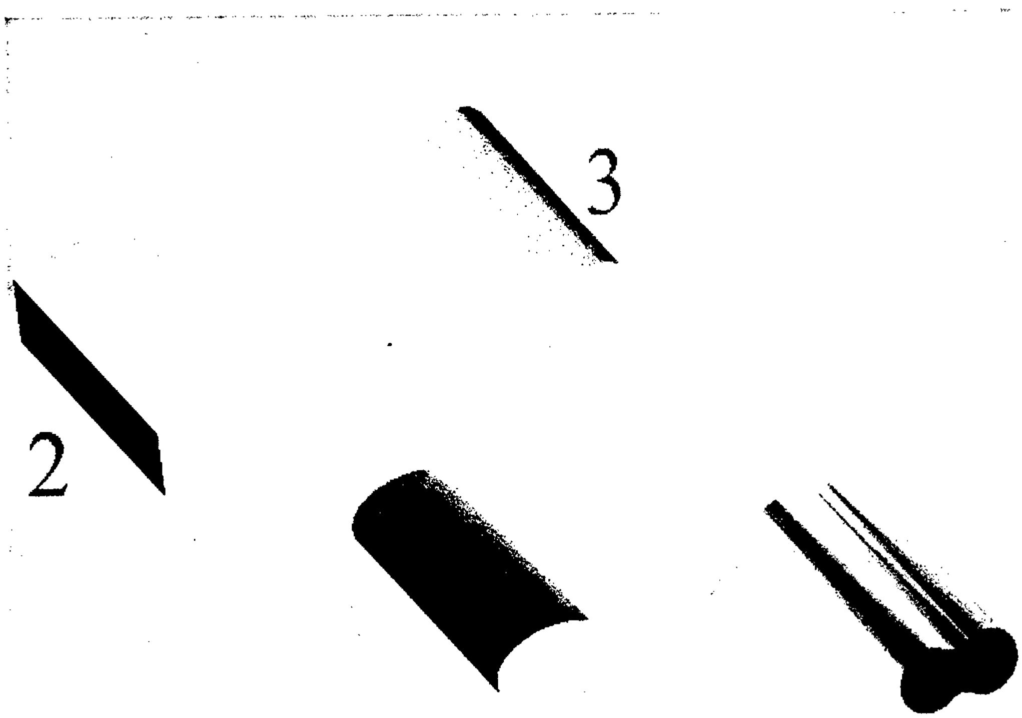


Figure 3.6. Main alternatives for fuselage shape are showed above.

3.1.5.Tail arrangement

Tail arrangement would be dependent on the general configuration of the airplane. For a twin fuselage configuration a boom-mounted arrangement would be proper. For a biplane or monoplane configuration conventional, v-tail or t-tail options could be suitable. Certainly these arrangements would have advantages and disadvantages.

A t-tail arrangement would be clear of the wing wake and propwash so the horizontal tail size would be reduced. This also reduces fatigue for both the structure and the pilot. However it would be heavier because of the need for extra strength of vertical tail that would support horizontal tail.

A conventional tail would supply adequate stability and control at the lightest weight and almost

70% of the aircraft had been using that arrangement.

V-tail was considered to reduce the wetted area. Because the forces on horizontal and vertical tails would be the projections of the force exerted upon the v-tail surfaces. Two main disadvantages of V-tail were their control-actuality complexity and adverse roll-yaw coupling risk.

A boom-mounted tail tail would be heavier but would allow pusher propellers at the same time. Usage of a boom tail would provide the possibility of locating the engine close the center of gravity. For a twin fuselage biplane configuration it could be a suitable tail arrangement.

3.2. Figures of merit

Combination of design alternatives presented above formed many different aircraft configurations. In order to analyze and choose the best manufacturable configuration for the contest team determined some criterions. Those criterions in other words figures of merit were as following: safety, ease of design and manufacture, aerodynamic efficiency (it would have impact on geometry and speed consequently rated aircraft cost and flight score), pilot control. Team decided these parameters to be equally important in order to make the proper design and perform a successful flight during contest.

3.2.1. Safety

In our design we did not considered safety only as a parameter for the safety of team members and viewer(other team members) but also the safety of the plane during flight. Because any kind of accidents or breakdowns during flight contest would lead big problems. We would most probably be unable to repair and even before contest any kind of repairs or changes in design would cause time trouble. A stable flight was one of the preliminary concerns of the team.

3.2.2 Ease of design and manufacture

During first meetings of the team some very interesting ideas and non-planar aircraft configurations were proposed. Lack of enough data for the calculations of such configurations would cause team to have many experiments to do and form some complicated computational fluid dynamics (CFD) codes. After discussing that type of design options team members decided that to be above the aims of such a student study especially when number of the team members were considered (only five). That decision caused ease of design to be considered as a figure of merit.

A configuration easier to produce would let the team have enough time for flight tests, modifications and optimizations on "Conqueror". Certainly, our manufacture department had declared some of the limitations of the facilities of our model plane club and some complicated designs had to be eliminated.

3.2.3. Aerodynamic efficiency

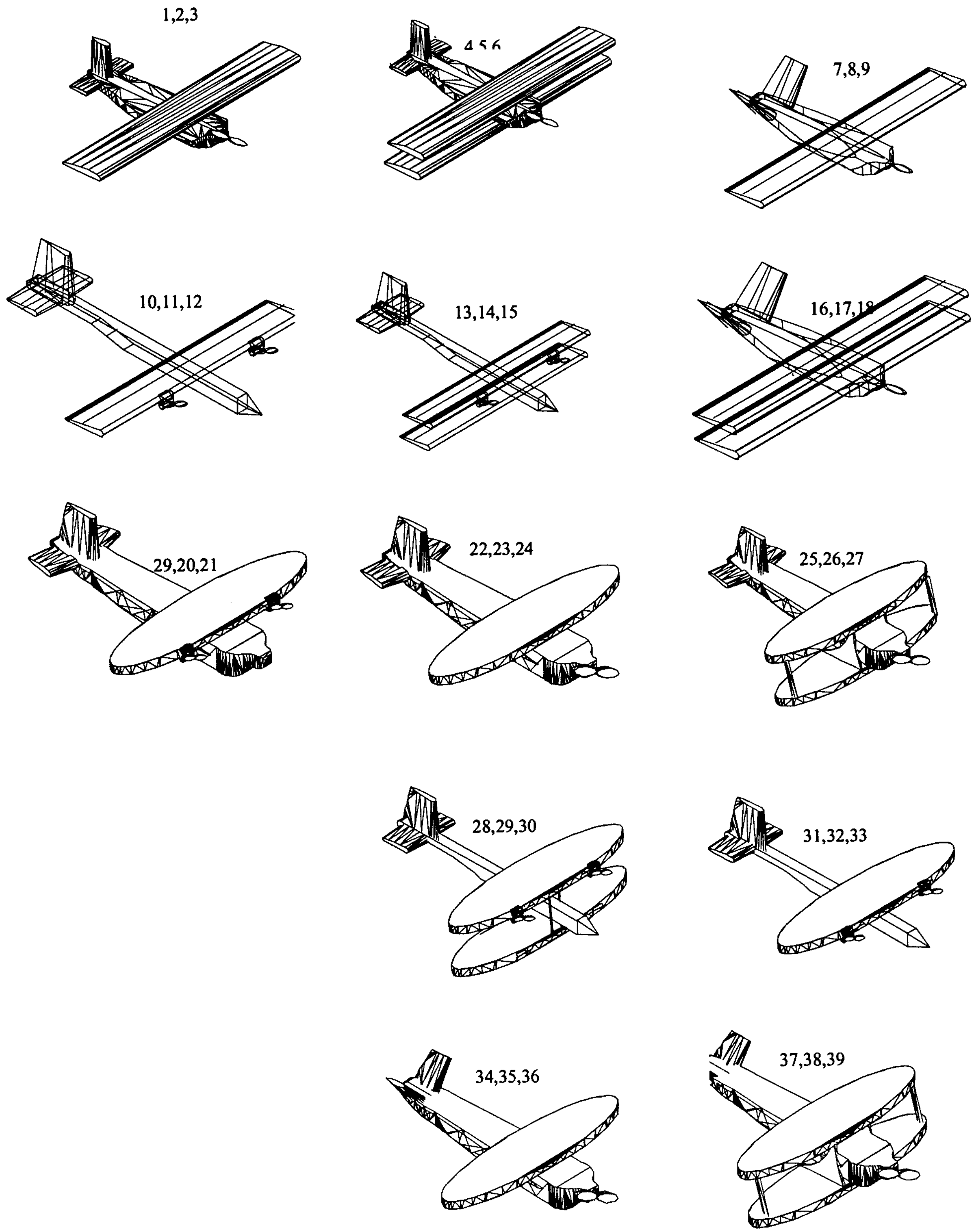
Aerodynamic efficiency of the configuration would effect the sizing of the airplane, propulsion system, stability and control of the airplane directly. Instead of considering all these parameters individually team decided to take the dominant factor, aerodynamic efficiency, as a figure of merit. Sizing and propulsion system would be the main factors of rated aircraft cost so would effect total score directly.

3.2.4. Pilot control

Besides the aircraft performance pilot will was a preliminary concern. Erkan who had an experience of R/C model airplanes manufacture and flight had not had an AMA license. He would try to acquire his license until the flight contest. His directions and requirements about the handling and flight characteristics of the design were important for a safe flight. Besides that team had to design "Conqueror" easy to be controlled by any pilots even not familiar with it. Because in case of being Erkan unable to obtain the AMA license team would have to have the help of a pilot from contest place as indicated in contest rules.

3.3. Configuration selection

After determining the figures of merit and design alternatives among the configurations


1,2,3
4 5 6
7,8,9
10,11,12
13,14,15
16,17,1
29,20,21
22,23,24
25,26,27
28,29,30
31,32,33
34,35,36
37,38,39

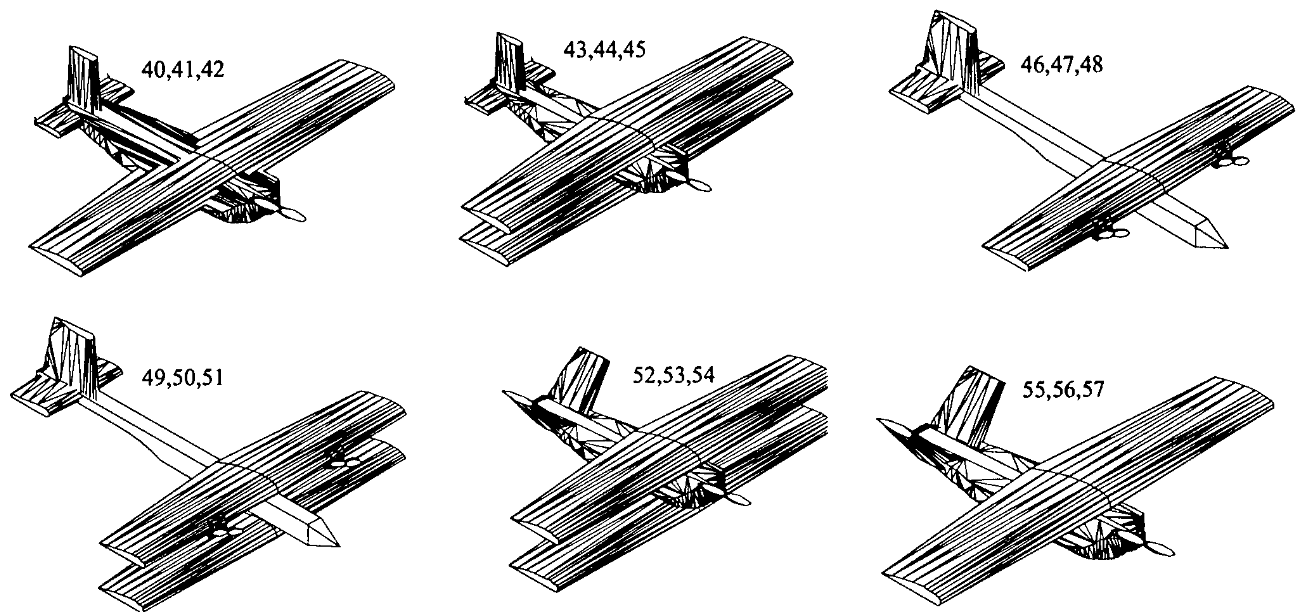


Figure 3.7. Some of the alternative configurations considered during conceptual design showed.

3.4. Results

After making the analyzes and judgements above team chose two two configurations as candidate designs and analyzed them in a more detailed way. One of them was a biplane configuration and the other was a monoplane configuration.

Wing planform was fixed as tapered wing. A rectangular wing would be the easiest one to manufacture but its aerodynamic efficiency was the worst. The best aerodynamic planform was an elliptical one however difficulties of manufacturing caused its elimination. A tapered wing with a correct taper ratio was less than one percent worse than an elliptical wing of equal aspect ratio. However the main hesitation of the team for a tapered wing planform was the stall characteristics. Team who had prepared the best scored report of 2001 Design/Build/Fly contest had eliminated tapered wing at the very beginning of design phase because of the tip stall risk. But lack of any calculations or detailed information led the team to study on stall characteristics of a tapered wing. Prof.Dr. Adil YUKSELEN who had developed a software related to subject let the team use his software. Team agreed on that besides a lift distribution very close to elliptical lift distribution a wing configuration with good stall characteristics was possible to design. Software was based on the Prandtl-Lifting Line Theory.

Non-planar geometries and elliptic wings were eliminated since out capabilities would not be enough to study on and manufacture such geometries. Besides that lack of obtaining enough data for very different configurations would oblige the team perform many experiments, wind tunnel tests etc. Such detailed studies were not possible due to lack of time.

Number	Safety	Ease of manufacture	Arodynamic efficiency	Pilot control	Total score	Decision
1	3	5	2	3	13	Elminated
4	5	4	3	2	14	Elminated
7	4	5	2	3	14	Elminated
10	5	5	0	2	12	Elminated
13	4	4	1	1	10	Elminated
16	3	4	2	2	11	Elminated
40	4	4	5	3	16	Elminated
43	3	3	4	2	12	Selected
46	5	4	3	2	14	Elminated
49	3	3	4	1	11	Elminated
52	5	3	4	4	16	Selected
55	4	4	5	2	15	Elminated

Table 2.1. Each plane above represents three concepts as tail type was not absolute yet. Non-planar concepts and elliptic wings were eliminated and did not exist on the table above.

4. Preliminary Design

During conceptual design team made an enumeration and determined two concepts for further analyzes that will be realized during preliminary design. Figures of merit and design alternatives of conceptual design were not enough to meet the demands of preliminary design. More detailed analyzes including estimation of rated aircraft cost, determining the load (contest rules allowed teams ability to carry a number of balls differing from 10 to 24), sizing and making calculations for both of the alternatives.

4.1. Design alternatives

Preliminary design required more detailed and more accurate calculations. During that phase team worked on two concepts simultaneously. It might seem to be a waste of time. However team had structural doubts for large spans and as it was mentioned in conceptual design phase one of the major concerns of the team was a safe flight. That was the reason for the attempt to have two alternative configurations and comparisons and judgements would be completed at the end of preliminary design. Design alternatives are listed and explained briefly below:

4.1.1. Aspect ratio

Wing aspect ratio would effect lift and drag coefficients of the chosen airfoil and flight performance (for example gliding characteristics). As stated above calculations and analyzes of the preliminary design alternatives were realized for both of the configurations (monoplane and biplane).

In the literature for the airplanes which had similar characteristics that were made evident with contest rules aspect ratio was recommended to be between 6 and 8. Calculations, tables and graphics were prepared in order see the effects of aspect ratio varying with angle of attack on drag and lift coefficients of chosen airfoil, Clark Y. Some conclusions reached from those analyzes were:

- A low aspect ratio wing would stall at a higher angle of attack, but with higher induced drag.
- A high aspect ratio wing had lower induced drag, was more efficient, but would stall at a lower angle of attack.

Another problem team had to deal with aspect ratio was that very high aspect ratio would mean larger span for required wing area. A very large span (although there were no limitations at contest rules) would cause structural problems and team would have great difficulty during transportation.

All these calculations and limitations forces the team to find an optimal solution. For mono wing an aspect ratio of 7 was decided. In case of biplanes for a single wing the same taper ratio was used and total aspect ratio of the biplane was 3,5.

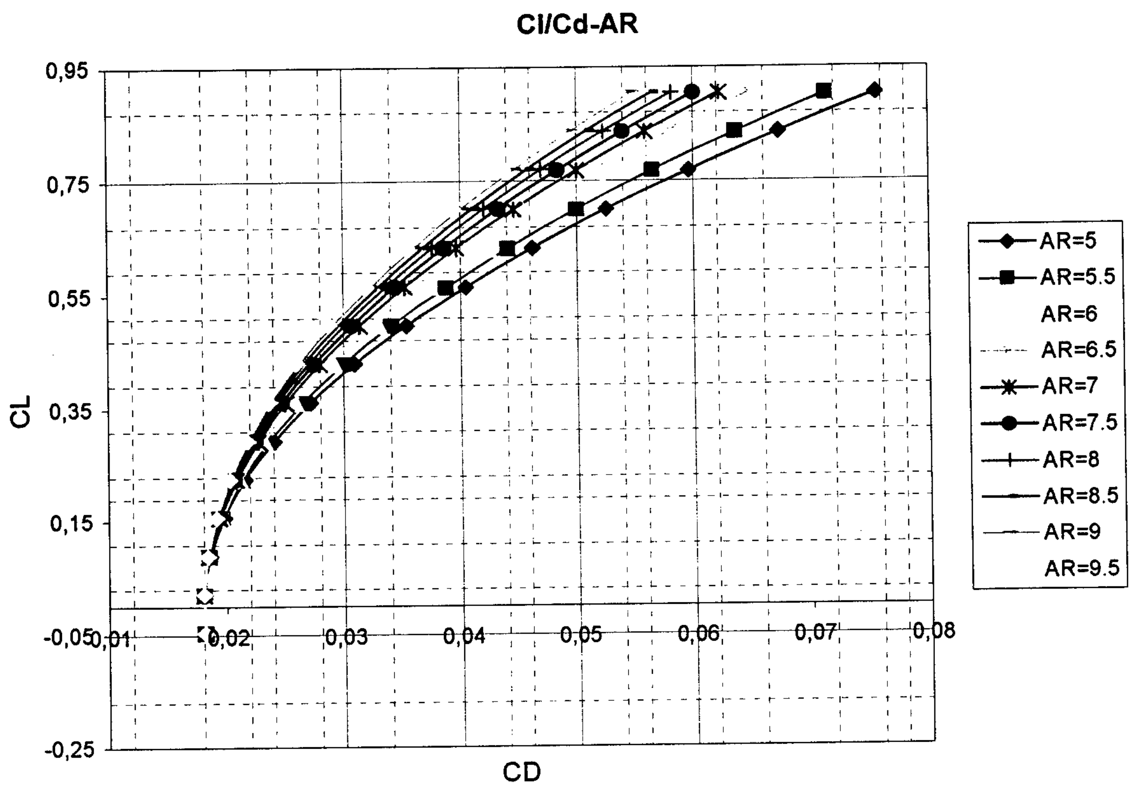


Figure 4.1. Effect of aspect ratio on lift and drag coefficient values of Clark Y. As we can see in the chart after the aspect ratio of 7 Cl/Cd value doesn't increase as it does in smaller aspect ratios.

a_{00}=0,068	a_{00}=0,068			Angle of Attack				
AR	CL	-3	-2	-1	0	1	2	3	4
3	0,067513	0,157305	0,224818	0,292331	0,359844	0,427357	0,49487	0,562382	0,629895
3,5	0,067582	0,157466	0,225048	0,29263	0,360212	0,427794	0,495376	0,562958	0,630541
4	0,067634	0,157587	0,225221	0,292855	0,360489	0,428123	0,495757	0,563391	0,631025
4,5	0,067674	0,157682	0,225356	0,293031	0,360705	0,428379	0,496054	0,563728	0,631403
5	0,067707	0,157757	0,225464	0,293171	0,360878	0,428585	0,496292	0,563998	0,631705
5,5	0,067733	0,157819	0,225552	0,293286	0,361019	0,428753	0,496486	0,56422	0,631953
6	0,067756	0,15787	0,225626	0,293382	0,361137	0,428893	0,496648	0,564404	0,632159
6,5	0,067774	0,157914	0,225688	0,293463	0,361237	0,429011	0,496786	0,56456	0,632334

		Angle of Attack									
AR	C_{DI}	-3	-2	-1	0	1	2	3	4	5	6
5	0,9007	0,0018	0,0036	0,0061	0,0092	0,0129	0,0173	0,0224	0,0281	0,0344	0,0414
5,5	0,8847	0,0016	0,0033	0,0056	0,0085	0,0120	0,0161	0,0208	0,0261	0,0320	0,0385
6	0,8691	0,0015	0,0031	0,0053	0,0080	0,0112	0,0151	0,0194	0,0244	0,0299	0,0360
6,5	0,8540	0,0014	0,0029	0,0049	0,0075	0,0106	0,0142	0,0183	0,0229	0,0281	0,0338
7	0,8392	0,0014	0,0028	0,0047	0,0071	0,0100	0,0134	0,0173	0,0217	0,0266	0,0320
7,5	0,8247	0,0013	0,0026	0,0044	0,0067	0,0095	0,0127	0,0164	0,0206	0,0252	0,0304
8	0,8106	0,0012	0,0025	0,0042	0,0064	0,0090	0,0121	0,0157	0,0197	0,0241	0,0290
8,5	0,7967	0,0001	0,0000	0,0004	0,0012	0,0024	0,0041	0,0061	0,0087	0,0116	0,0150

C_{DI} Katsayının Kanat Açıklık oranına (AR) ve Hücum Açısına (a) Göre Değişimi

C_{Do}	0,018	Angel of Attack								
AR	-4	-3	-2	-1	0	1	2	3	4	5
5	0,0186	0,0198	0,0216	0,0241	0,0272	0,0309	0,0353	0,0404	0,0461	0,0524
5,5	0,0185	0,0196	0,0213	0,0236	0,0265	0,0300	0,0341	0,0388	0,0441	0,0500
6	0,0185	0,0195	0,0211	0,0233	0,0260	0,0292	0,0331	0,0374	0,0424	0,0479
6,5	0,0185	0,0194	0,0209	0,0229	0,0255	0,0286	0,0322	0,0363	0,0409	0,0461
7	0,0184	0,0194	0,0208	0,0227	0,0251	0,0280	0,0314	0,0353	0,0397	0,0446
7,5	0,0184	0,0193	0,0206	0,0224	0,0247	0,0275	0,0307	0,0344	0,0386	0,0432
8	0,0184	0,0192	0,0205	0,0222	0,0244	0,0270	0,0301	0,0337	0,0377	0,0421
8,5	0,0184	0,0192	0,0204	0,0221	0,0241	0,0267	0,0296	0,0330	0,0368	0,0411

4.1.2. Structural weight and wing area estimation

Rules of the Design/Build/Fly Contest of 2002 related with payload was different from previous years' contests. However while estimating structural weight team used the statistical data of previous years to compare with the first estimations realized with the help of the experienced manufacturing department members. Besides the attendants of previous contest different aircrafts of model airplane club of Turkish Airforce Academy were analyzed and compared.

			Monoplane		Biplane	
		Unit Weight(gr.)	Number	Total(gr)	Number	Total(gr.)
Airframe	Wing	750	1	750	2	1500
	Fuselage	2500	1	2500	1	2500
	Tail	500	1	500	1	500
	Landing gear	1000	1	1000	1	1000
	Strut	150	0	0	1	150
Propulsion	Motor	500	1	500	1	500
	Cell	65	30	1950	35	2275
Avionics	Servo	43	3	129	3	129
	Receiver	40	1	40	1	40
	Cell	89	1	89	1	89
	Total Structural Weight		**7458gr.**		**8683gr.**	

Table 4.1. Estimations for the components of airplane based on the previous experiences and previous successful similar aircraft.

Softball weight = 0,186 kg.		**Dry Operating Weight) (kg)**		**Wing Area**	
#Balls	Payload(kg.)	Monoplane	Biplane	Monoplane	Biplane
10	1,86	9,32	10,54	0,796972693	0,901747489
12	2,23	9,69	10,92	0,828790019	0,933564815
14	2,60	10,06	11,29	0,860607345	0,965382141
16	2,98	10,43	11,66	0,89242467	0,997199466
18	3,35	10,81	12,03	0,924241996	1,029016792
20	3,72	11,18	12,40	0,956059322	1,060834118
22	4,09	11,55	12,78	0,987876648	1,092651444
24	4,46	11,92	13,15	1,019693974	1,12446877

Table 4.2. In this table we made the first estimations of take off weight for each load alternative (10-24 balls) and for the cruise flight wing areas were determined for monoplane and biplane alternatives.

Wing loading of the design was estimated to be a value of 11,6917431 kg/m^2 (3,047894001 lb/ft^2). The average wing loading of the aircrafts in Design/Build/Fly Contest of 1998 was 3,691032 lb/ft^2. That compare indicated that our wing loading was less than other successful configurations that had similar size. No data was available for attendants of other years about wing loading so only the statistics of 1998 used for wing loading. A graphic showing the statistical data for the 1998 Design/Build/Fly Contest is presented below.

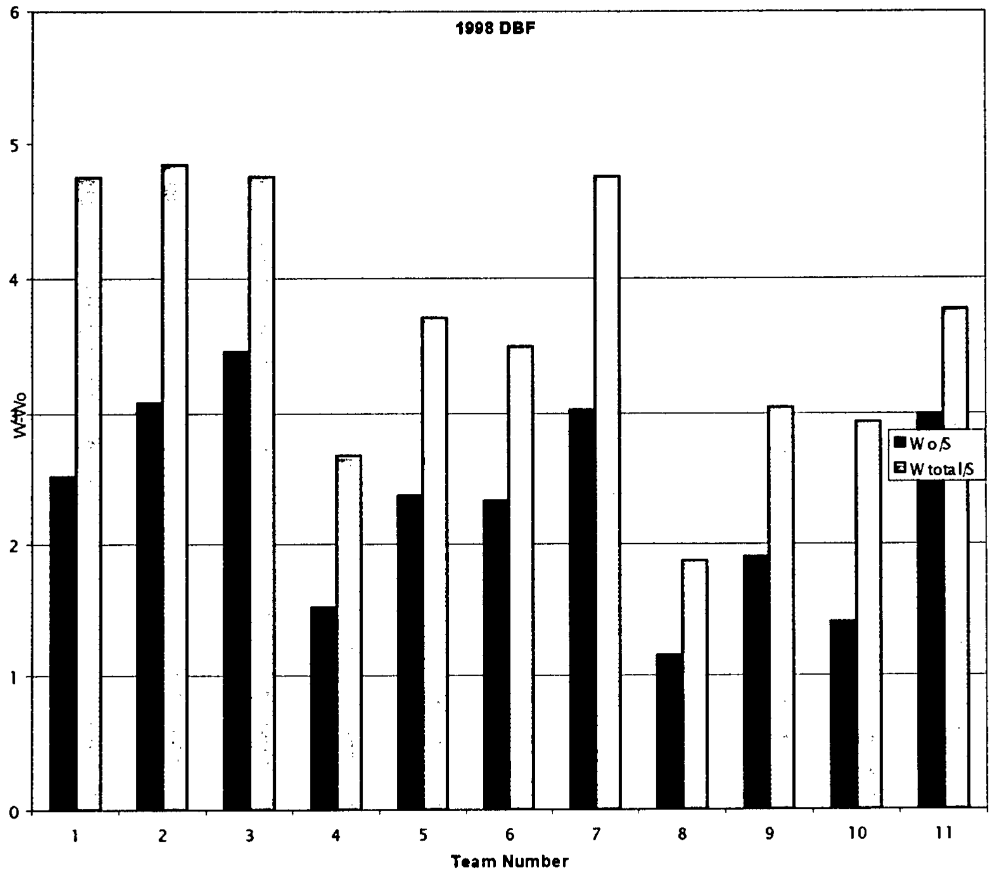


Figure 4.2. Table shows the wing loading values of 1998 Design/Build/Fly Contest. While preparing this chart only the planes that could complete the flight stage of contest were taken into account.

4.1.3. Taper ratio

Taper ratio was an important design parameter to be investigated by the team to acquire good flight characteristics with "Conqueror" a lift distribution very close to elliptical lift distribution and a safe stall progression was aimed. In A.G. Andy LENNON's book "R/C Model Airplane Design" a taper ratio of 0.35 was calculated to be the most efficient but in "Aircraft Design: A Conceptual Approach" of Daniel P. RAYMER a taper ratio of 0.4 was defined as ideal for unswept wings.

In order order to analyze stall progression and lift distribution at the same time program "Yukselen" was used. Program was drawing the planform with using your inputs like aspect ratio, wing span, airfoil characteristics, panel taper ratios lengths and outputs like wing lift coefficient, drag coefficient, lift distribution and how much it differs from elliptical distribution and their visualization on wing planform were taken.

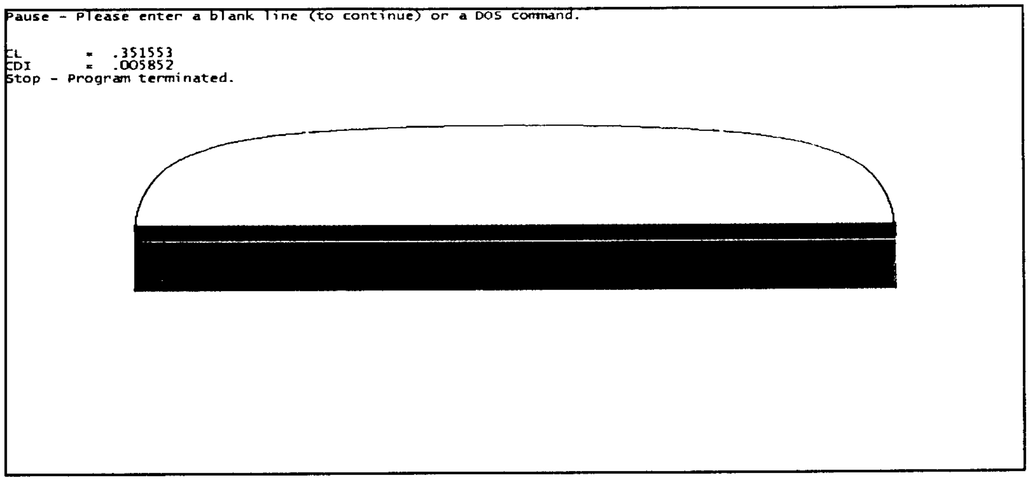


Figure 4.3. The output graphic of program Yukselen for a rectangular wing. Brown colored line shows the elliptical load distribution, green colored line above the planform is the load distribution of current wing and the green line under the wing indicates the stall development. Lift coefficients and induced drag coefficients for the zero angle of attack are also presented.

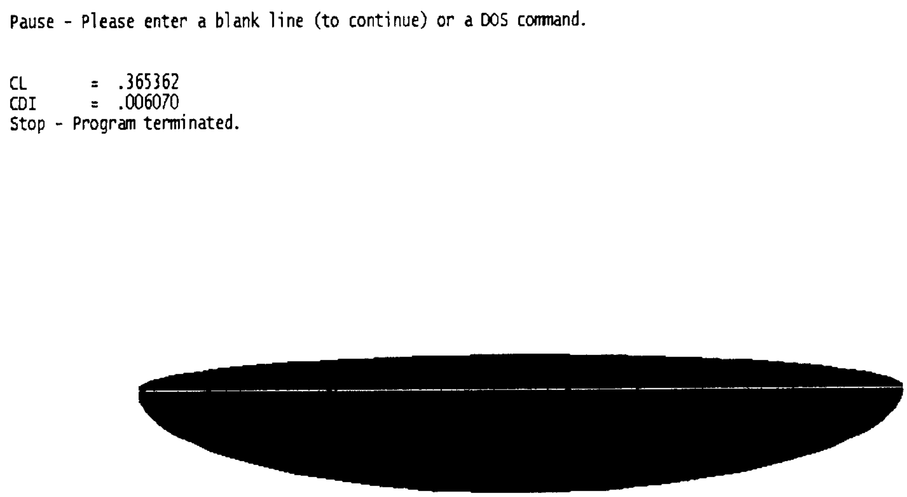


Figure 4.4. **Green line above the wing indicates the lift distribution for an elliptical wing. The**

green line below shows the stall development. It was aerodynamically the most efficient wing.

```
Pause - Please enter a blank line (to continue) or a DOS command.

CL      = .368365
CDI     = .007185
Stop - Program terminated.
```

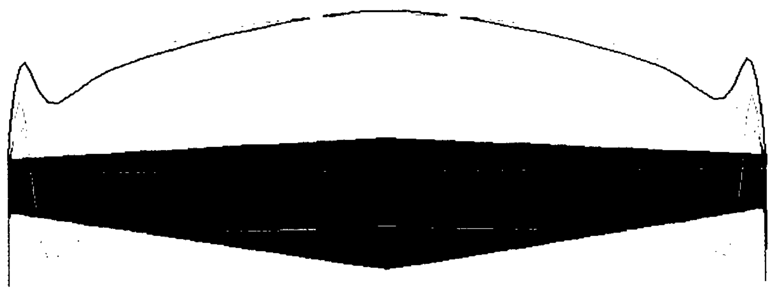


Figure 4.5. ***That output was a very important lesson we learnt. Because in some books a taper ratio of 0,40 was recommended. However for our airfoil, wing span and flight characteristics a tip stall would occur as we can see in the figure.***

```
Pause - Please enter a blank line (to continue) or a DOS command.

CL      = .361425
CDI     = .005992
Stop - Program terminated.
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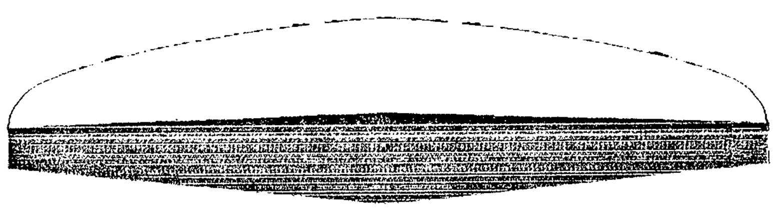


Figure 4.6. A taper ratio of 0,45 would eliminate the risk of tip stall totally as we can see in the green line under the wing planform. Load distribution of the wing with that taper ratio would be almost the same as the elliptic wing. We can see both of the load distributions as a green and a red line over the wing planform.

Since stall progression was related with wing span, aspect ratio and airfoil characteristics using published data without calculation could cause mistakes. As we can see from the graphics obtained by program Yukselen a taper ratio of 0.4 would cause tip stall. Analyzes with that software showed that a taper ratio of 0.45 would make the lift distribution very clause to the elliptical lift distibution. Calculated lift and drag coefficients for the zero degree angle of attack were on graphics above.

4.2.4. Wing gap

For the performance of a biplane configuration wing gap was an important parameter. In the reference books wing gap was mentioned very generally and it was recommended to be more than average chord. Since no softwares were available for wing placing excell sheets were developed by the team. In order to find the optimal wing gap because if the gap was too much structural problems would rise and connection struts would cause additional drag and a too short gap would increase interference between two wings raising the overall drag. Because of their minimal effect on aircraft performance and in order to get simplicity at biplane wing analyzes stagger and decalage were not used in the configuration.

In order to find the Oswald Span Efficiency Factor (e) for the biplane configuration Prandtl's interference factor was used first however at high lift coefficients the drag polar broke away from the parabolic shape represented by a fixed value of K in equation $K=1/(\pi.A.e)$ however the e method ignored the variation of K with lift coefficient. Because of that a semi-empirical method called Leading-Edge-Suction Method was also used to estimate K besides the e method.

In stead of writing down all the detailed analyzes of this very useful study very brief graphics are presented as below. Some examples from the tables for the calculations will be presented at Appendix A.. It was not very easy to find out correct and scientific data for that part of our study so we hope that they will be beneficial for the next year's team if a biplane configuration is considered. Our first calculations were to determine the gap/average span. For upper wing and down wing, for average spans varying from 1meter (3,2808 feet) to 2 meters (6,5617feet) and for aspect ratios between 5 and 11, gap/average span values were calculated and Figure 4.7. was formed.

After finding out the gap/average span values, Pandtl's diagram was used to obtain the interference factor values for different gap/average span values that were not over the limit. Limit was to have structural endurance and it was determined by the manufacturing department. Results are were summarized at Figure 4.8.

Interference factors were determined and presented in Figure 4.9. In the formula the r value (area of shorte wing/area of longer wing) was accepted as one in order to make both of the wings identical.

Calculated interference values were used to determine lift/drag ratios and with the help of that chart a gap of 125% of average chord was decided. It would provide both aerodynamic efficiency and it would cause less problems than a gap 150% of average chord.

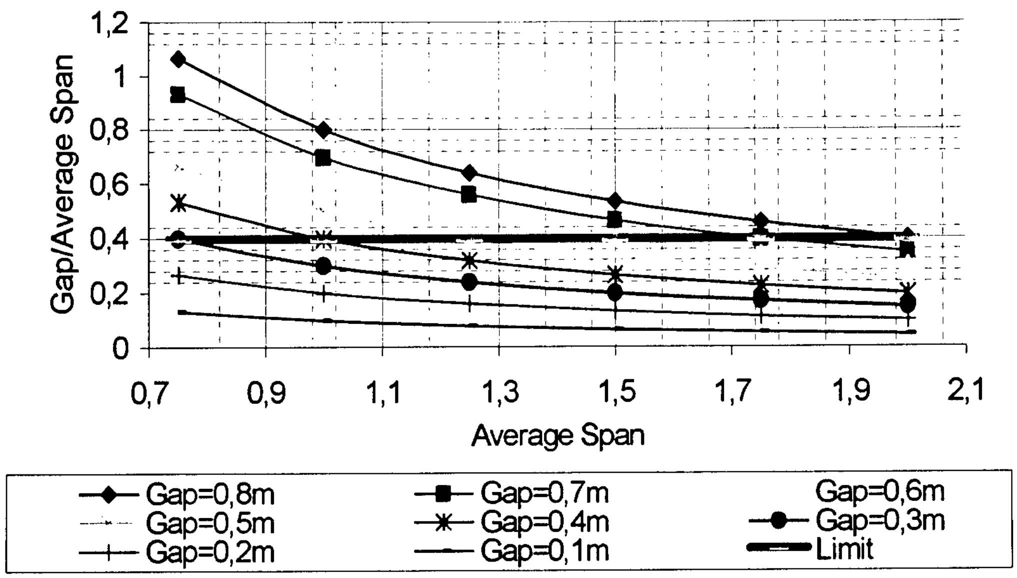
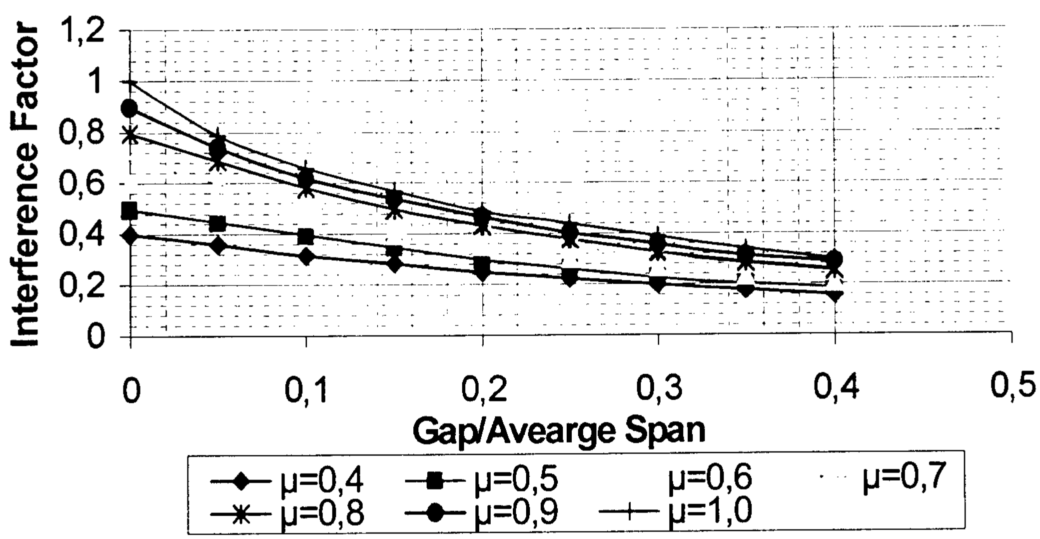


Figure 4.7. Calculated gap/average span values are shown in this chart. A pre-determined limit value is also indicated and for further analyzes values below that limit were used.

Figure 4.8. Interference factor (σ) values were obtained by using shorter span/longer span(μ) and gap/average span values.



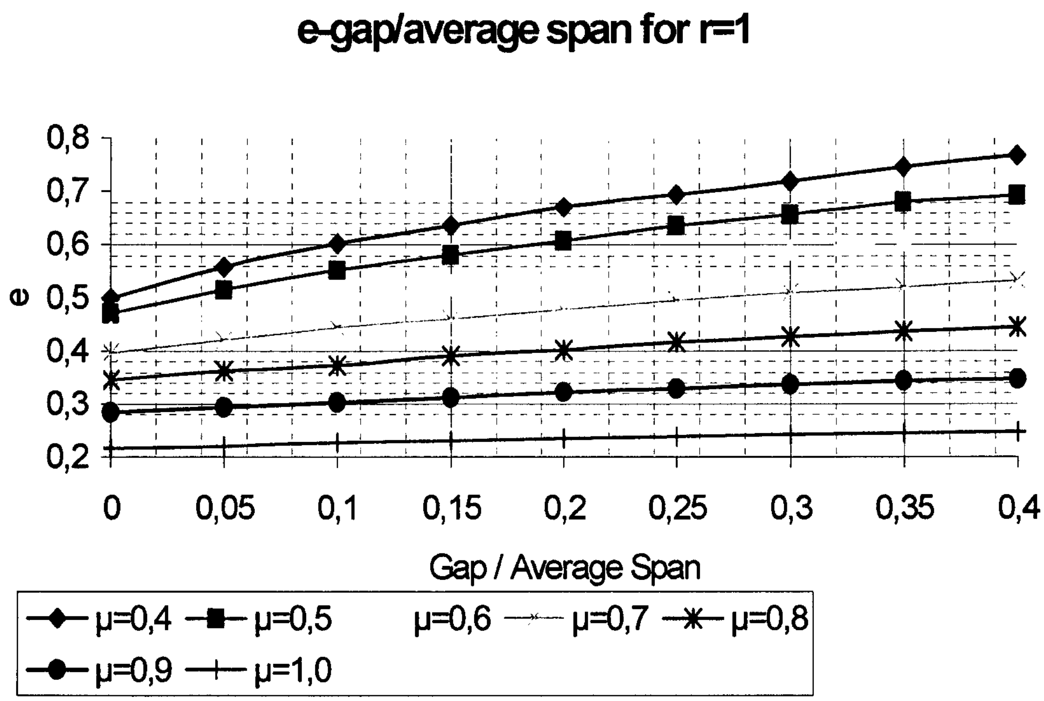


Figure 4.9. After determining the interference factor as shown in Figure 4.8. Oswald Efficiency Factor was determined with the Prandtl's formula : $e = (\mu^2(1+r2)) / (\mu^2+2\sigma\mu r+r^2)$.

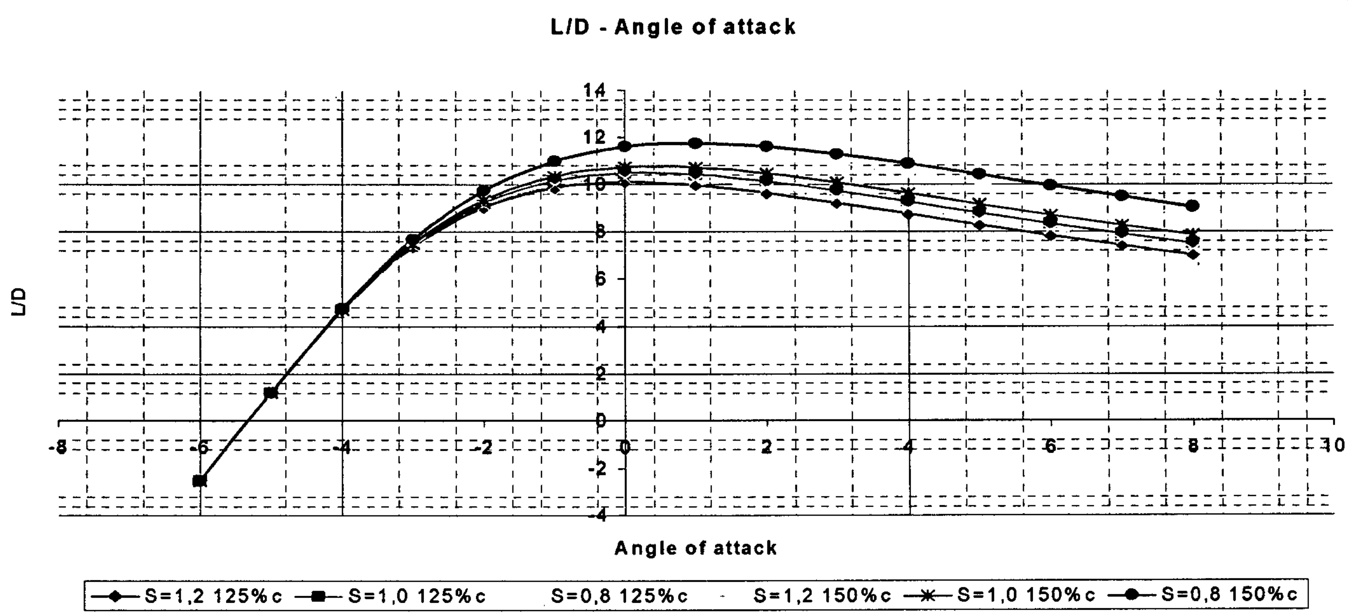


Figure 4.10. Drag coefficients were calculated above and lift/drag ratio for different wing areas and gaps are presented in this chart. As seen increasing the gap to 150% of average chord had minimal effect on lift/drag. In order not make any concessions from structural strength gap value was determined as 125% of average chord.

4.2.5. Tail sizing

Type for both configurations were determined as conventional tail. Aspect ratio of horizontal and vertical tail were taken as 3,2 and 1,6 respectively. Taper ratio for both tails was 0,5. Aspect ratio and taper ratio were determined by using historical data.

Besides being a very important component of the airplane, tail (especially horizontal tail), was an important factor for rated aircraft cost. At the very beginning of tail sizing we found out that for the monoplane design a tail aspect ratio of 4 would make a horizontal tail span about 75 centimeters. Although wing area or wing span were not considered during rated aircraft cost calculations a horizontal tail span larger than 25% of wing span would cause the tail admitted as a wing and rated aircraft cost would increase significantly. Both tail volume coefficient method and the method based on $S_{ht} = K \times MAC \times S_w / TAM$ gave similar results. We had to shorten the tail span however increasing the tail moment arm would cause the configuration be very long and rated aircraft cost would increase again.

Historical data for horizontal tail aspect ratio was varying between 3 and 5. We chose 3,2 instead of 4 and without an additional increase at empenange length we were able to have enough tail area.

Finally the aspect ratio was optimized and final tail dimensions were as follows:

Monoplane Tail Sizing								
c_{ht}= 0,5		c_{vt}= 0,04		$AR_{H.T.}$=3,2		$AR_{V.T.}$=1,6		
Horizontal Tail					**Vertical Tail**			
#Balls	**$c_{ave.}$**	**Span**	**Area**	**Elevator**	**$c_{ave.}$**	**Span**	**Area**	**Rudder**
10	0,18	0,56	0,100	0,062	0,19	0,30	0,0558	0,065
12	0,18	0,58	0,105	0,063	0,19	0,31	0,0587	0,067
14	0,19	0,59	0,110	0,065	0,20	0,31	0,0615	0,069
16	0,19	0,61	0,115	0,066	0,20	0,32	0,0645	0,070
18	0,19	0,62	0,120	0,068	0,21	0,33	0,0674	0,072
20	0,20	0,63	0,126	0,069	0,21	0,34	0,0703	0,073
22	0,20	0,65	0,131	0,071	0,21	0,34	0,0733	0,075
24	0,21	0,66	0,136	0,072	0,22	0,35	0,0763	0,076
Biplane Tail Sizing								
10	0,14	0,57	0,082	0,050	0,17	0,27	0,0462	0,059
12	0,15	0,59	0,086	0,051	0,17	0,28	0,0483	0,061
14	0,15	0,60	0,090	0,052	0,18	0,28	0,0504	0,062
16	0,15	0,61	0,094	0,054	0,18	0,29	0,0525	0,063
18	0,16	0,62	0,098	0,055	0,18	0,30	0,0546	0,065
20	0,16	0,64	0,101	0,056	0,19	0,30	0,0568	0,066
22	0,16	0,65	0,105	0,057	0,19	0,31	0,0589	0,067
24	0,17	0,66	0,109	0,058	0,20	0,31	0,0611	0,068

4.2.6. Landing gear design

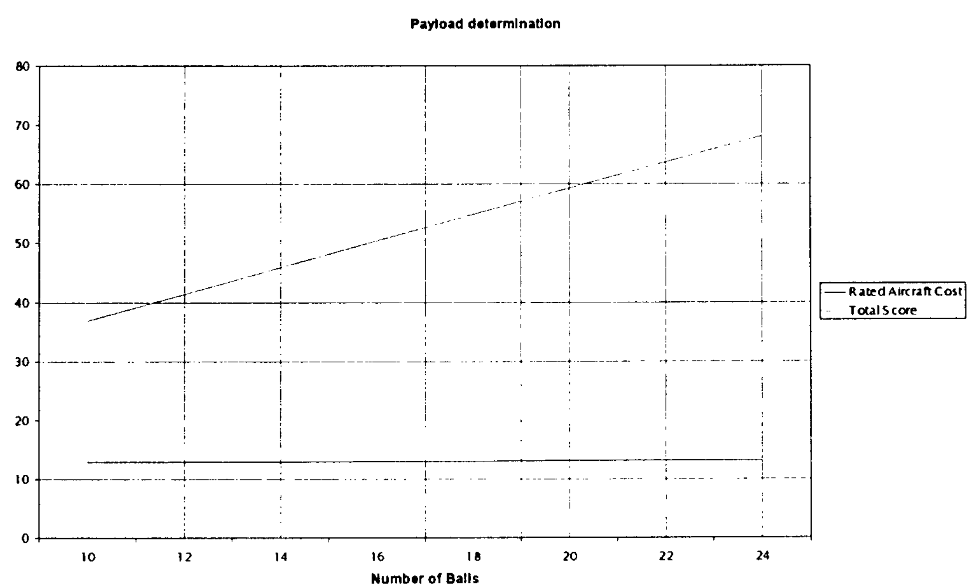
For monoplane configuration two major types of landing gears were investigated during preliminary design. They were conventional taildragger (tailwheel) and tricycle (nosewheel) type of landing gears. In a taildragger design two wheels would be located ahead and a steerable tailwheel well behind center of gravity. A tricycle landing gear was consist of a nosewheel ahead of center of gravity and main wheels behind. Judgement of the team was eliminating the taildragger configuration because of the possible instability of taildragger configurations. Very briefly, instability was a result of the obligation of locating the center of gravity at a higher point, appearance of a higher thrust line, existence of wheels well ahead of center of gravity. On the other hand a tricycle configuration would be more stable because of the correcting force couple formed by the tire friction and aircraft's momentum.

The landing gear length had to be set so that airplane would not hit the ground on landing. In order prevent such a crash (safety was the first concern in all design stages), length of landing gears (in static position) were determined in order to make the aircraft land at an angle of attack that gives 90% of the maximum lift. Reduction about 10% was a result of ground effect. It was called tail angle and for our design it was about 10 degrees. In order to diminish the nose-up attitude the angle off the vertical from the main wheel position to the center of gravity was determined as 20 degrees. The overturn angle was the angle from center of gravity to the main wheel where the main wheel was aligned with nose wheel and it was determined as 53 degrees. Strut-travel angle was decided to be about 7 degrees for better taxi control.

For the biplane configuration tricycle type of landing gears would not be suitable for proper landing conditions. That caused the team design a taildragger concept. For the taildragger concept taildown angle was determined as 12 degrees. Main wheel was decided to be 20 degrees front from vertical measured from the center of gravity. In order to prevent overturning main wheels were decided to be separated laterally 30 degrees from center of gravity.

4.2.7. Payload determination

In order to determine the payload or in other words number of balls that "Conqueror" would carry, excell tables and graphics were developed. Graphics proved that although wing span and wing chord increased (it caused an increase at rated aircraft cost) total flight score increased, too. According to the analyzes made team decided to carry maximum number of balls that contest rules permit.



4.2.8. Payload location

Contest rules limited payload location; balls could be arranged as a single vertical row and at least two horizontal rows. 2*12, 3*8, 4*6.... were possible configurations. Putting more balls at wing span axis would make a shorter fuselage possible. Rated aircraft cost of such a configuration could be less however the tail moment arm would decrease significantly and to get a stabile flight a greater tail area would be required. Besides that because of the increase of frontal area a greater power would be necessary to obtain a cruise speed safely above stall speed.

Location	2*12	3*8	4*6
Estimated Fus. Length	2	1,6	1,4
R.A.C.	13,18599935	12,70479548	12,57356189
Total Points	68,25421235	70,83939299	71,5787625

As we could see at the table, differences at the total points appeared because of payload location were minimal. Team judged that in order to have such a little increase would not compensate the loss of tail effectiveness. One or more disadvantage of locating the payload in more than two rows was that since the speed would decrease total flight score would decrease. Those were the main reasons for locating the balls as 2*12.

4.2.9. Final design selection

In order to find the configuration a compare of rated aircraft cost and estimated total score were used. For the monoplane configuration RAC=13,186 and total score=68,254212, for the biplane configuration RAC=15,483 and total score=58,127825 were the estimated values only to compare. While calculating those values written score was taken as 100 and total flight score as 9. However in order to make a judgement that compare was not the only parameter. Team was not sure if a very large wing span would be strong enough to resist the aerodynamic and structural loads in addition to wingtip test. After the approval of manufacturing department team decided that (certainly after making analyzes) a monoplane would be better for such a contest and team would be more successful with that configuration.

5. Detailed Design

Determinig the power loading was the first calculation made at the beginning of the detailed design. Propulsion system design began with selection of propeller. After that the aircraft performance was calculated and optimized in order to fit the requirements of the contest if necessary. One other important concern of the detailed design phase was to make structural analyzes of main components. Stability analyzes were performed in order to find out if the flight quantities of "Conqueror" were proper for a safe flight.

5.1. Performance Calculations

5.1.1. Power loading

Thrust to weight ratio would directly affect the performance of the aircraft. An aircraft with a higher thrust to weight ratio will accelarate more quickly, climb more rapidly, reach a higher maximum speed and sustain higher turn rates.

Using a table providing reciprocal values for power to weight ratio of different aircrafts presented in Reference 1, a P/W value of 0,13 watt/gr. (0,08 hp/lb.) was obtained. This was equal to a power loading of 7,692 g/watt (12,5 lb/hp).

Another method for estimating P/W was thrust matching. That method was based on the formula: P/Wo = a V_{max}^c . "a" and "c" coefficients were based on the statistics acquired from Taylor, J., "Jane's All the World Aircraft", Jane's, London, England, UK, 1976.

a=0,006 c=0,57

Vmax.(km/h)	P/W_o(watt/g)
72	0,068680077
81	0,073449308
90	0,077995488
99	0,082349945
108	0,08653719
117	0,090576846
126	0,094484912
135	0,098274648
144	0,101957193

Table 5.1. Power loading determination for different maximum speeds.

5.1.2. Wing loading

The wing loading is the weight of the aircraft divided by the area of reference wing. Wing loading affects stall speed, climb rate, take off and landing distances and turn performance. As it was clear from those explanations that power loading and wing loading would be optimized together.

Average wing loadings for the airplanes attended to Design/Build/Fly Competition in 1998 and in 2001, and wing loadings for "Conqueror" were as follows:

1998-1999: averageW_{total}/S=3,691032 lb/ft^2 and average Wo/S=2,760613 lb/ft^2

2000-2001: average Wo/S= 1,68318648 lb/ft^2

Conqueror :

Loaded: 2,371612399 lb/ft^2

Unloaded: 1,483600509 lb/ft^2

Our wing loading seemed logical when compared to aircrafts of previous years' competitions. Certainly, it was also caused by the change at payload determined by contest rules.

5.1.3. Stall speed

The stall speed of an aircraft is directly determined by the wing loading and maximum lift coefficient. Stall speed is a major contributor to flight safety, with a substantial number of fatal accidents each year due to "failure to maintain flying speed." At the same time, which is the most important factor in landing distance and also contributes to post- touchdown accidents, is defined by the stall speed.

Maximum lift coefficient is usually admitted as 90% of maximum lift coefficient of airfoil. For our profile (Clark Y) at a very close Reynold's number to our actual flight conditions 1.2 was given in published data. In our calculation we used 1,1 as maximum lift coefficient.

$Vstall=\sqrt{(2xW/S/(Cl_{max}x\ \rho))}$

From the formula above we obtain the stall speed as: Vstall=14.77 m/s.

5.1.4. Take off distance

For the take off distance formula that was used contained the thrust to weight ratio.

V_{TO}=1.2 Vstall=17.72m/s.

$S= V_{TO}^2/2g*(T/W)$=53.34m.=175ft.

5.1.5. Climb Performance

$\sin\gamma=T/W-D/W$

$D/W= q.Cdo/(W/S)+K.(W/S)/q$

$K=1/(\pi.AR.e)=1/(\pi.7.0,84)=0,054$

Cdo was determined by program "Drag estimator". It was using the "Component Drag Method".

$Cdo=Cd_{fuselage}+Cd_{profile}+Cd_{induced}+Cd_{Htail}+Cd_{vtail}+Cd_{gear}$

Cdo=3,479xE-3+4,638xE-3+4,003xE-3+5,205E-3+5,193xE-3+0,4223=22,97xE-3

D/W=0,024+0,066=0,09

$\sin\gamma=0,21 \quad \gamma=12^{\circ}$

Rate of climb(R/C)=V x $\sin\gamma$ = 4,2 m/s.

5.2. Stability Analyzes

Stability analyzes were really challenging tasks for the team. Because none of the team members had taken the stability and control course, yet. Team members had to learn the analyze methods themselves. "The Principles of the Control and Stability of Aircraft", W.J.DUNCAN, Cambridge University Press, 1959 had great help to carry out this difficulty.

Besides using the Matlab codes a software called Pitch Stability Estimator was used to compare the calculations.

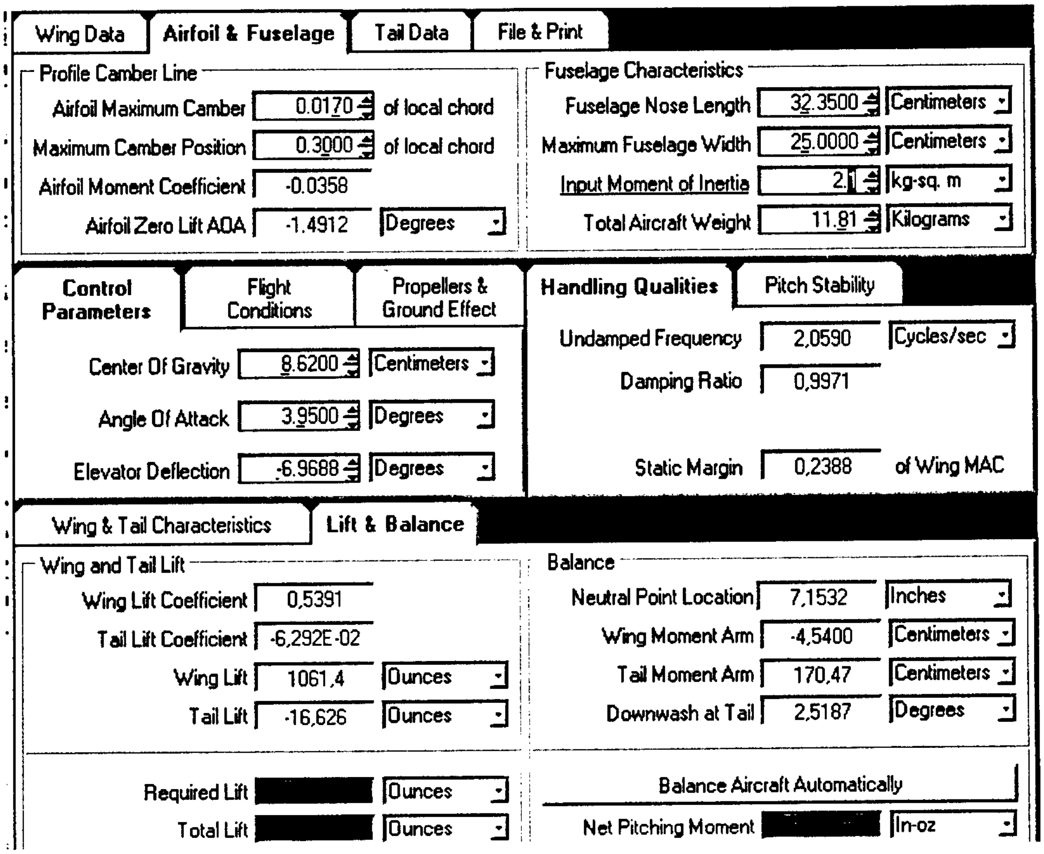
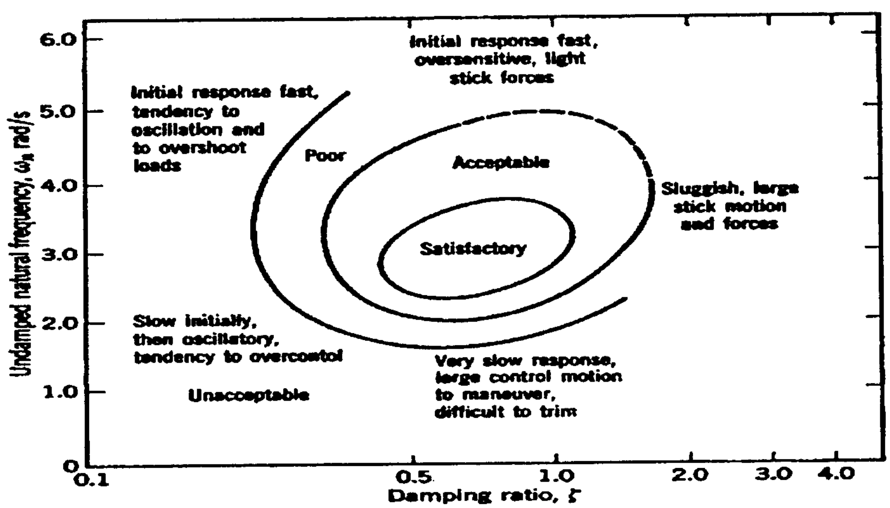
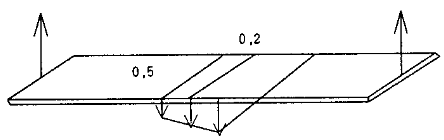


Figure 5.1.-5.2. After obtaining the values of undamped natural frequency and damping ratio from the software, we checked them with the help of the chart below. Conqueror had proper pitch stability characteristics.



5.3. Structural Analyzes

We made some basic calculations and modeled two critical conditions of "Conqueror": first of them was the wing tip test and the other calculation was to determine the forces on wheels during landing.

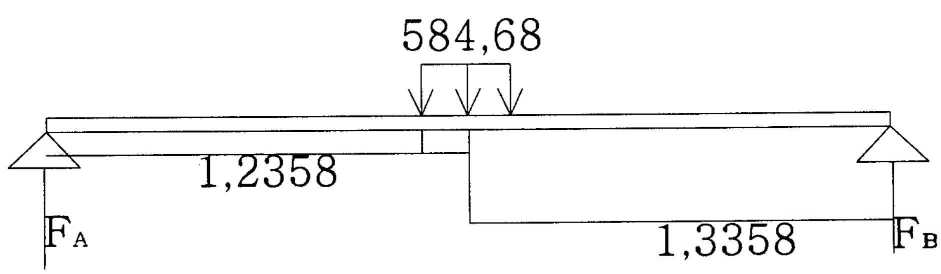


11,92 kg =1169352N

P = F/ A = 1149,13 Pa

Q = 1149,13 *0,5088

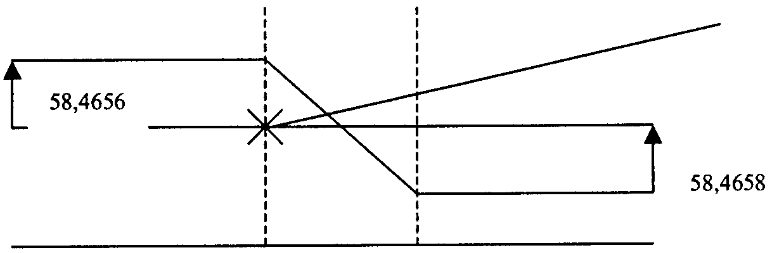
Q = 584,68 N/m

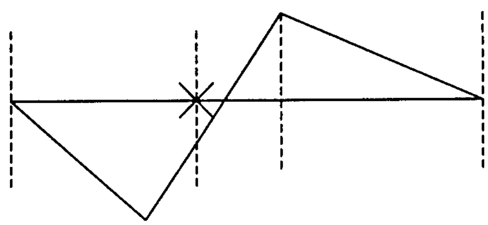


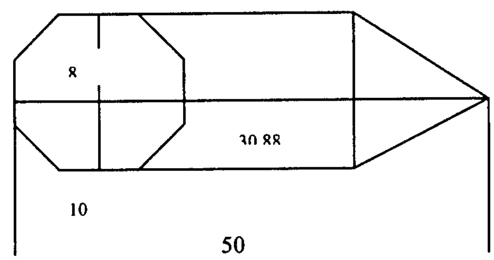
$\sum M_A = 0$ 584,68 * 0,2 *1,3358 = F_B * 2,6717

F_B = 58,4658

F_A = 58,4658







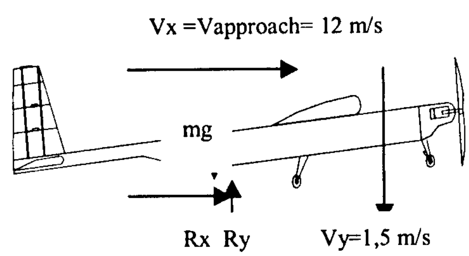
$M_{max} = F_B * 1{,}2358 = 58{,}4658 * 1{,}2358$ $\quad M_{max} = 72{,}252$
$Bh^3/12 = (0{,}08)^2 * 0{,}3088\ /12 = 0{,}000164693$
$\Pi R^3/12 = 3{,}14 * (0{,}1)^3 = 0{,}000261667$
$Bh^2/4 = 0{,}08 * (0{,}1)^2\ /4 = 0{,}00020$
$I = 000626493\ m^3$
$\Gamma_{max} = M_{max}\ /\ I = 72{,}252/0{,}000626493 = 115327{,}76\ Pa = 115{,}3\ kPa$

Stretch distance= dy =3 cm=0,03m

$Vy^2 = 2aydy$ $\quad ay = 25$ m/s

$\Sigma Fy = 0$ $\quad mg-Ry = m.a$ $\quad Ry-mg=m.ay$

$g=9{,}81$
$m=11{,}92$
$Ry = 414{,}93$ N



5.4. Propulsion System Design

5.4.1. Motor and cells

When we arranged our first meeting we found out in our model airplane club an electric motor had never been used before. In addition to that any shop that was selling equipment for small aeroplanes had any. That was really over our estimations. We had to order it from another country and the arrival time was not exact. We had not even completed our analyzes for propulsion system but we had to order one urgently in order to receive before the competition. Motocalc had great help to us. Certainly we could not totaly rely on it. When we analyzed the commercially available brushed electric motors from their online catalogs we decided on Astroflight 691. It was one of the strongest motors within the limits of the competition. Our analyzes could seem to be useless however it was not. We hope them to be useful for next year's team.

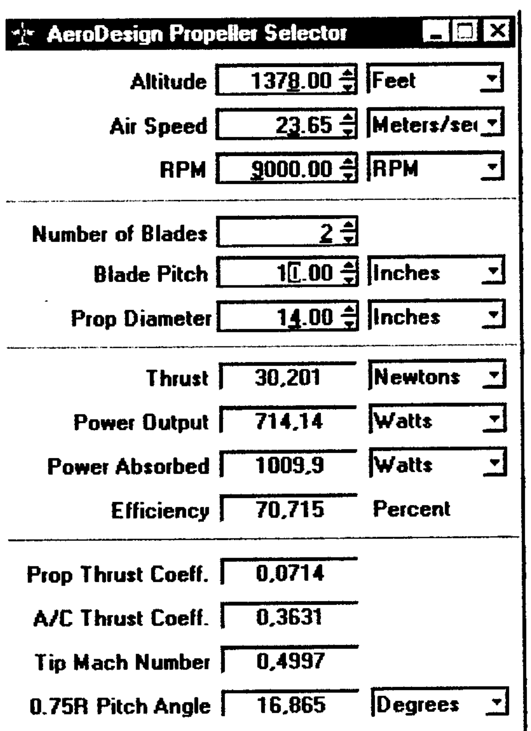
Erkan spent great effort on propulsion system and fortunately he was student at electronics engineering. Briefly decision was on Astroflight 691 and 36, 2000mAh cells. Catalog values for the motor was shortly:

Best propeller:	RPM:	Cells:	Control:	Charger:
14x10	9000	36	270	1120

Power in (watt)	Power out (watt)	Weight (oz)
1335	1100	28

5.4.2.Propeller selection

We used prop selector for first estimations and extended prop selector for determined the actual propellor. An 14 X 10 inch propeller was the optimal choice for the characteristics of "Conqueror."



6. Manufacturing Plan

After months of studies, analyzes, calculations result would be seen as an aircraft totally designed by us. In order to manufacture "Conqueror", as needed in all design stages a very careful planing had to be done.

6.1. Manufacturing processes investigated

Manufacturing department was consist of three model airplane club members. Sıtkı, Kürşad, Erkan had many experience of fabricating scale models and sailplanes. Their main material for the structure of their models was mostly balsa. The first alternative for manufacturing process was using balsa for most of the structure, while birch plywood and spruce would be used sparingly for high-stress areas especially for motor mounts, landing gear blocks, and wing attachment pieces. Such a structure would require an additional plastic or fabric covering material. This was a popular method of construction for aircraft of this size and manufacturing department got used use this method for some years.

The second alternative for manufacturing "Conqueror", a fiberglass based approach was considered. In this manufacturing process option, the main structure of the aircraft would be fabricated by forming fiberglass cloth in pre-formed molds for the various components of the aircraft. In such a construction balsa usage would be minimized and might be used only as formers or mounting hardpoints for motors, landing gear, or servos. Similarly, carbon fiber or kevlar cloth in molds was considered to be used at portions of the aircraft.

As it was mentioned before safety was a very important criteria in all design phases and fabricating most of the aircraft with aluminum was also considered. Great part of the airplanes were constructed by using that method since Second World War. However, it would cause great increase at structural weight.

Each component of the aircraft had different requirements and they would operate under very different loads. A combination of the methods was considered to satisfy the figures of merit for each structural component. That was the fourth main alternative for manufacturing process.

Instead of the attempt to generate the manufacturing process for whole airplane, each component could be analyzed individually. That approach would best suit to fourth alternative of manufacture. Before the presentation of figures of merit explaining the materials and making comments about would be suitable.

6.2. Material properties

Brief investigations of material properties and idiosyncracies are below:

Balsawood: Balsa is strong in tension and reasonably strong in compression provided the cross section is square, otherwise it is prone to buckling. Hard balsa twice in width to thickness such as 3/8" x 3/16" can be used as wing spar. However balsa webs should be used to prevent buckling.

Balsa is a strong material in shear across the wood grain. That is why spar weds should have vertical grain and be positioned between the ribs, so that the ribs have only spar flange cut outs. Balsa is weak in shear along the grain, but is easily bent across the grain. This mistake it ideal for wing and fuselage skins where the grain runs parallel to the wingspan or fuselage center line, and where it abosrds bending and shear loads to best advantage.

Spruce or Basswood: These two have much the same properties as balsa, but are stronger and heavier. They are ideal for spars on thin wings as used for gliders and, in sheet form, much stronger than plywood for wing center section reinforcement. There it would replace the balsa shear webs. Its grain would run horizontally, without being interrupted by the center section rib or ribs.

Plywood : Plywood is made from thin birch laminations, with their grains at $90^0$to each other . The outer laimination grains run in the same direction. It is stiff and rigid in all direction. It is ideal for such heavily stressed areas as firewalls, and landing gear mounts. Use of 1/16 $''$ ply for control surface ribs to which are bolted brass or plastic concealed horns is recommended. When epoxied into the control surface, there ribs will not fail as balsa would.

Aluminum : At one- third the weight of steel, is good for landing gear legs and to reinforce stress points. Various alloys are available, but 2024-T6alloy is recommended it doesn't have to be formed.

Covering Materials And Cement : Regarding cements heat shrinkable plastic films like monokote and regarding cements were used extensively but the dissolved cellulose types like Ambroid and Duca shrink on drying and can distort the structure. Since "Conqueror" would not be in touch with water cements like "Tite Bond" is also usable. Short drying time of cyanoacrylates is also considered however they are not as strong as others.

6.3. Figures of merit

During the manufacturing plan these figures of merit were considered:

- Previous experience
- Availability
- Cost
- Required skill levels
- Time of construction
- Safety
- Weight

Previous experience was generally indicating if the manufacturing department had any experience of using that material. Because such an experience would ease the construction significantly and certainly the problems that might be encountered would be minimized.

Materials to be used in the construction of the aircraft had to be available to the team. This figure of merit was limiting for the team. Certainly, balsa was always available but we were not sure about obtaining some materials. For example the foam that has the properties we want or ,if we mention about the propulsion system, when this report was being written down team was not able to receive the motor ordered two months ago.

Real cost of the aircraft was not a part of the rated aircraft cost. On the other hand as a student project this study had a very limited budget so careful material selection was required.

Skill of the manufacturing team was especially reliable on balsa and wood fabrications. Because of that usage of composite materials would cause time loss during manufacturing process.

Since the aircraft needed as much flight time as possible before flying at the competition, it needed to be built as quickly as possible. That necessity was one the main reasons for using time of construction as a figure of merit.

In all design phases safety was always a very important factor. For different components different safety criteria was required. This forced the team use some heavier materials especially at critical areas on which greater stress applied.

Weight was directly related with aircraft performance and rated aircraft cost. It was also taken into account during the final selection of manufacturing process and especially the need to find an optimal solution between weight and safety required a careful judgement.

6.4. Final selection of manufacturing processes

After considering these figures of merit as seen in Table 6.1. , the team decided to use balsa for wing and tail constructions, white foam for fuselage, for landing gear. Fuselage and wings would be covered with balsa for extra endurance against shear stress and in order to obtain give the aerodynamically efficient geometry. Landing gear materials were determined as carbon robs, aluminum and commercially available wheels.

Component		Experience	Availability	Cost	Required skill	Construction time	Safety	Weight	Total
Wings	All balsa	5	5	5	4	4	4	5	32
	All foam	3	5	4	5	5	4	3	29
	Foam&balsa sheet	4	5	4	4	3	5	2	27
	All aluminum	0	4	3	4	4	5	0	20
	Composite	1	4	3	4	4	5	4	25
Fuselage	All balsa	5	5	5	3	2	2	4	26
	All foam	3	5	4	4	5	2	3	26
	Foam&balsa sheet	4	5	4	4	4	4	3	28
	All fiberglass	0	3	2	4	5	4	4	22
	Carbon fiber composite	1	4	3	2	4	3	4	21
	All aluminum	0	3	2	4	5	5	0	19
Empennage	Balsa	5	5	5	2	3	2	5	27
	Aluminum	3	5	5	4	5	5	3	30
	Carbon sticks	2	5	4	5	5	4	4	29
Wing spar	Aluminum	2	5	4	5	5	5	1	27
	Thick balsa	5	5	5	5	5	3	3	31
Tail	All foam	3	5	5	5	5	4	4	31
	All balsa	5	5	5	4	4	4	5	32
	Foam&balsa sheet	1	5	4	3	3	5	3	24

Table 6.1. With the help of table presented above manufacturing processes for main components were determined. Landing gear was not listed above. Instead of buying, manufacturing team decided to make the landing gear themselves with aluminum and carbon fiber.

6.7. Wing construction

A two spared structure was realized. The reason for using such a structure provided the ability to butt joint the balsa sheet over the spars without having to resort to large widths. After that structure was full sheet covered for maximum strength.

In this process the built-up bulkheads improved strength and reduced material usage compared to cutting out of sheet balsa. The small weight increase at the corners where the balsa overlaps was admitted as negligible.

To satisfy the structural requirement of having the fully loaded aircraft picked up by its wingtips, 3/4"-wide unidirectional carbon fiber tape was applied as spar caps. The carbon fiber was applied symmetrically from top to bottom.

Here is a list of the material sizes used:

Wing

Wing skins	3/32" thk.
Ribs	1/8" thk
Main spar flanges	¼ " x ½
Main spar webs	2 x 1/8 "
Auxiliary spar flanges	¼ x 3/16 "
Auxiliary spar webs	3/32" thk
Leading edge spar	3/16" sq
Rib spacing (span wise)	4"

Control Surface

Skins	2/32"
Ribs	As above
Spars-Ailerons	¼"sht
Spar caps &L.E. Flaps	3/16" sq
Webs	2/32" thk

6.8. Tail surface construction

Tail construction was not very different from the construction of the wing. Only a different airfoil was used and certainly sizes of the materials were different and they were as follows:

Tail

Skins	1/32" thk
Ribs	1/16" thk
Main spar flanges	1/8" x ¼"
Main webs	1/16"
Rear spars	1/8"thk sheet
Rib spacing	3"

Elevator-Rudder

Skins	1/16
Ribs	1/16 thk
Main spar flanges	3/16" x ¼"
Main webs	1/16"
Rear spars	3/16"thk sheet
Rib spacing	Approx.3

6.9. Fuselage construction

Our material for fuselage was mainly white foam with epoxy laminated on each side as bread and the foam core as the ham. Epoxy formed the flanges and while the foam made up the web. In that method we obtained very stiff, homogeneous, strong structure that overcame all the metal problem. It eliminated the vast majority fasteners and joints necessary. Labor was significantly eased and the aerodynamic shape wanted was provided.

Before hot wiring the solid foam a front view of the fuselage including the space for payload and cells was drawn at 1/1 scale with the help of autocad. That drawing was used to make the mould.

6.10.Manufacturing timing

	Planned	
	Start	Finish
Wings	03.05.2002	03.16.2002
Mail Airfoil Sections	03.05.2002	03.09.2002
Build Wing Spar	03.05.2002	03.05.2002
Frame and Sheet Wings	03.12.2002	03.15.2002
Finish Work	03.16.2002	03.16.2002
Landing Gear	03.17.2002	03.19.2002
Assemble and Test Landing gear	03.17.2002	03.19.2002
Fulelage	03.14.2002	03.16.2002
Mould preparetion	03.14.2002	03.14.2002
Wiring the foam	03.15.2002	03.15.2002
Finish Work	03.16.2002	03.16.2002
Compenents and control Systems	03.17.2002	03.18.2002
Servo and Controller Installation	03.17.2002	03.17.2002
Mount Landing Gear, Motor,and ,Tailboom	03.18.2002	03.18.2002
Final Preparation	03.19.2002	03.25.2002
Test Mounts and control Linkages	03.19.2002	03.21.2002
Monokote all Components	03.21.2002	03.25.2002

8.General References :

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- Christopher A. Lyon, Andy P. Broeren, Philippe Giguere, Ashok Gopalarathnam, and Michael S. Selig "Summary of Low Speed Airfoil Data (1973)
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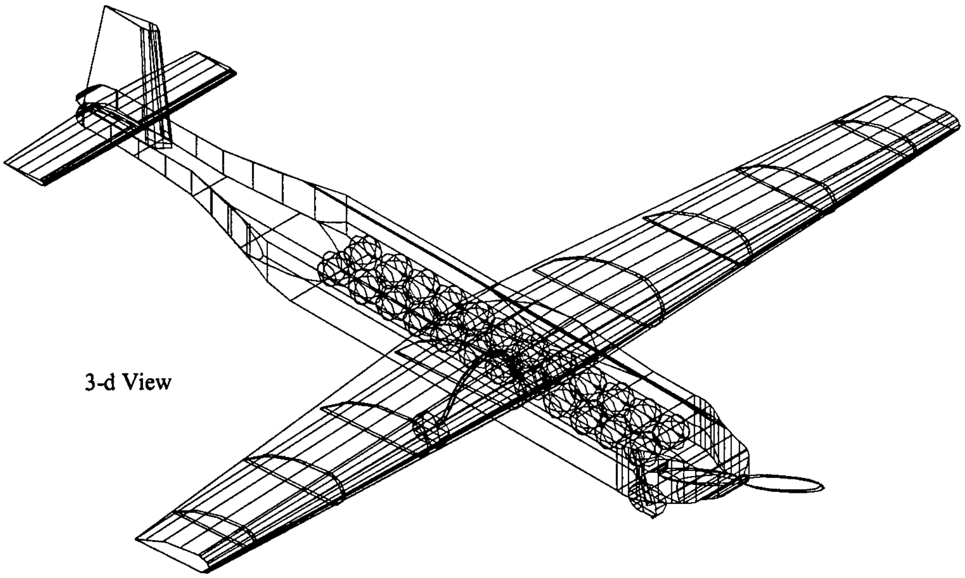
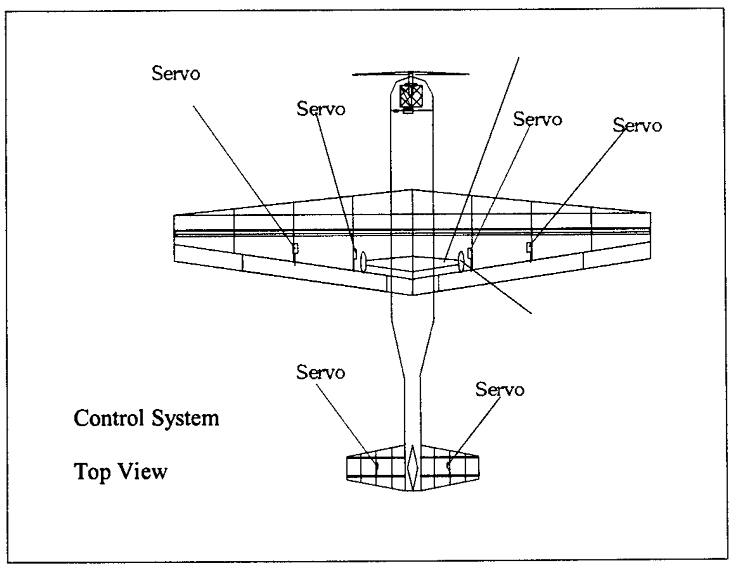
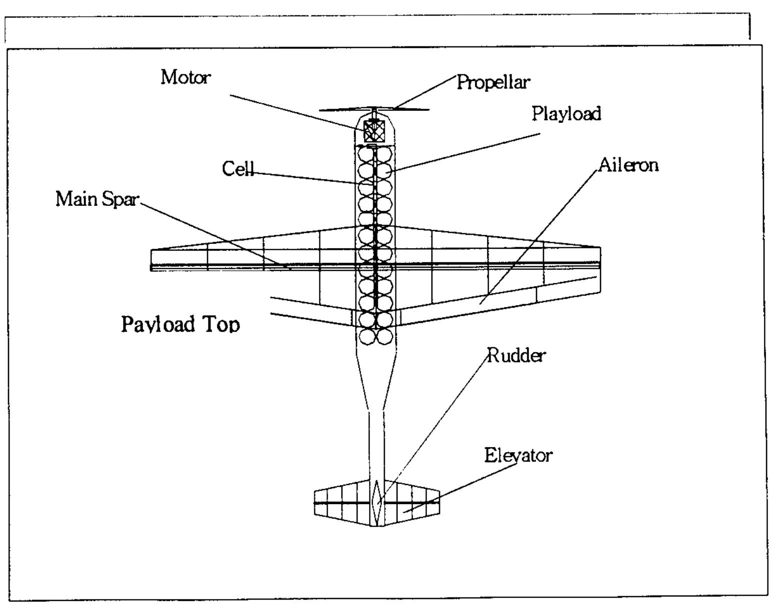
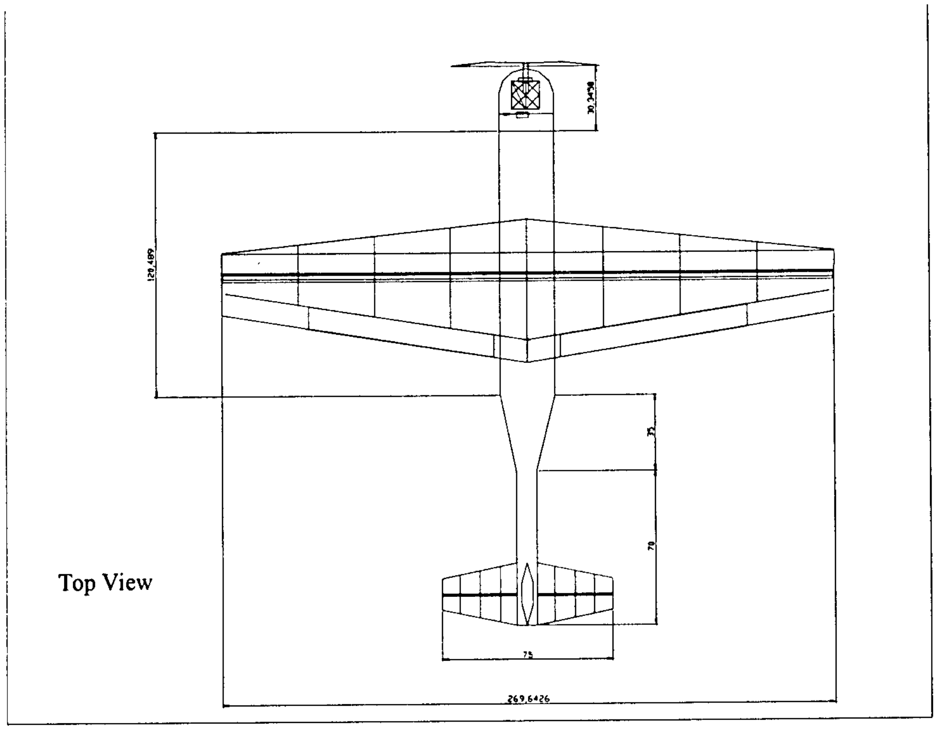
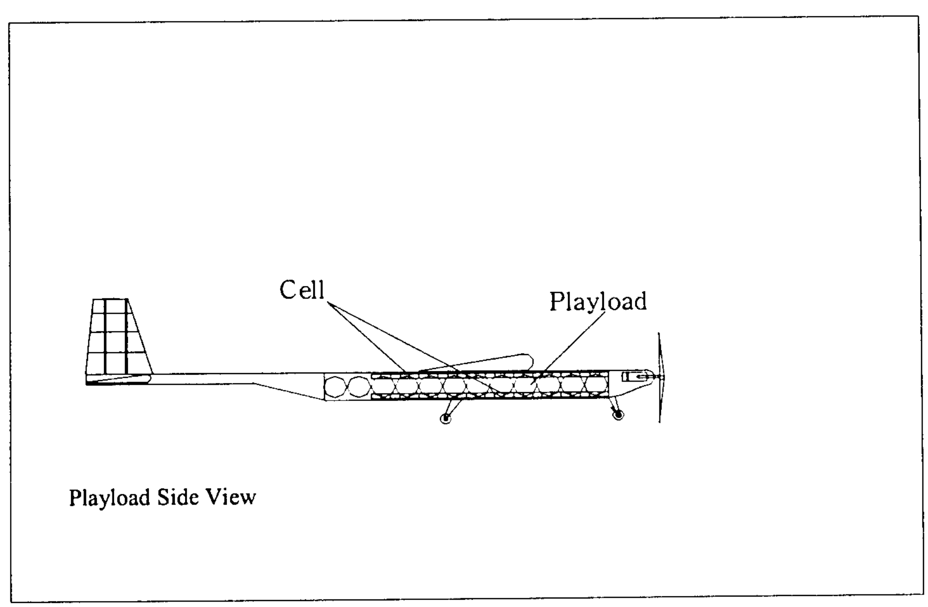

3-d View


Servo
Servo
Servo
Servo
Servo
Servo
Control System
Top View

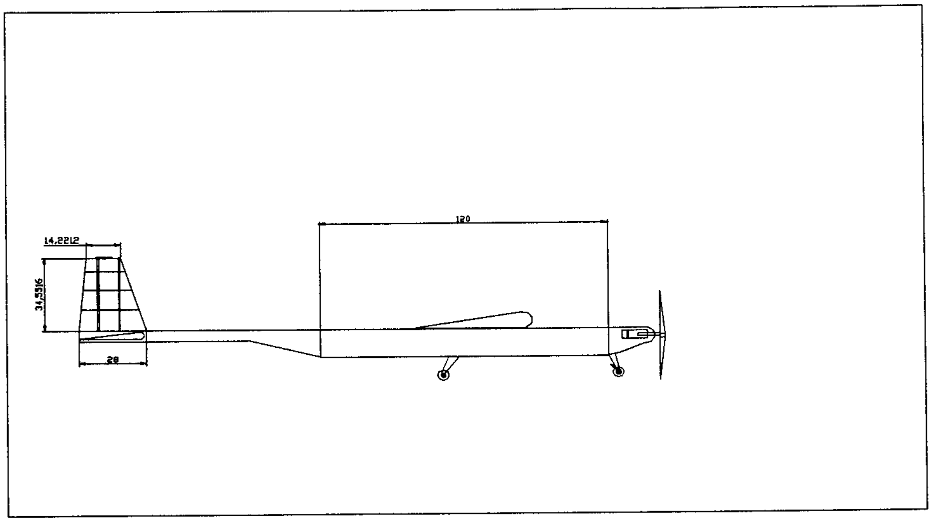

Motor
Propellar
Playload
Cell
Aileron
Main Spar
Payload Top
Rudder
Elevator

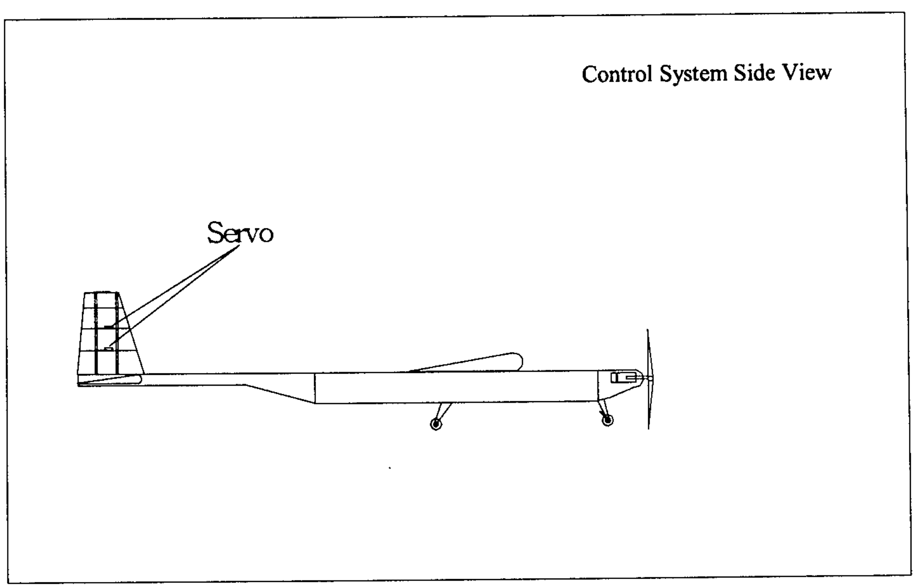


Top View



Playload Side View


14.2212
34.5316
28
120


Control System Side View
Servo

2001/2002 AIAA

Cessna/ONR Student Design/Build/Fly Competition

Design Report

"It's Close Enough"

Miami University
Oxford, Ohio

March 2002

Table of Contents

1. Executive Summary

1.1. Introduction

A team of students from Miami University (MU) has chosen to participate in the 2002 Cessna/ONR Student Design Build Fly Competition. The seniors are using this contest as part of Miami's required capstone experience. The design team is composed of five seniors and additional members from the MU Aeronautics department, with the help of four advisors. Two of the advisors are from the Engineering and Applied Sciences department and the two additional advisors from the Aeronautics department. The school is entering the competition for the fourth year.

Our senior design classes cover one full academic year and are split into two successive semesters. It is designed to give students control over all aspects of a project. These experience classes gives us the experience and provides specialized knowledge in the end of our baccalaureate work, as well as utilizes our knowledge in science, mathematics, and engineering to perform this major open-ended design project. The students involved apply to a variety of project selections based on their own interests and whom they work well with and vice versa. Students are then placed into groups by the advisors of the projects based on this input. After the selection process, our group was informed of our chosen project. The problem assigned to our team is to work on the designing, building, and flying of a remote control airplane to be entered into the 2002 Cessna/ONR Student Design Build Fly Competition. This competition gives specific requirements, rules, scoring, and tasks that our group must engineer a plane that is capable of meeting these expectations and surpassing other competitors.

This year, the requirements of the contest are to design and construct an electric powered, propeller driven, unmanned remote controlled (RC) airplane that is capable of carrying a payload consisting of American Softball Association approved softballs through a designated course. This course consists of one sortie flown without a payload, landing and taking on a payload, flying another sortie, landing and unloading, and flying a final sortie unloaded. The scoring of the competition is weighted on the best three flights completed during the plane's timed runs, the completeness of a written design report judged by AIAA officials, and the computed Rated Aircraft Cost of the designed plane. Cash prizes are awarded to the top three qualifying teams.

The team's objective is to design, build, and fly a remote controlled airplane that will give us the highest possible score. By balancing our strengths, we divided the team in order to better focus the overall abilities of every person. We also divided the overall objective of producing the best possible plane into three sections. These sections were the conceptual design, preliminary design, and detail design.

1.2. Conceptual Design

The conceptual design section covers the three major designs that the group had decided on investigating. These major designs were the cylindrical fuselage design, the flying wing design, and the airfoil fuselage with tail design. All these models were graded and given scores that were then weighed against one another using a criteria matrix and other factors of merit in an Excel spreadsheet. This

helped find where the weakest points on each design were located and possible ways to improve upon them. The differences between the designs also became much more apparent. From this fact, we were able to decipher the possibilities of the team in regards to how to possibly manufacture the designs, and from that, which designs seemed the most feasible. The criteria matrix also includes information on the component requirements and all their criteria to help in our initial decision making process. These components underwent preliminary investigations in order to narrow down the possibilities that would be used in our plane. Having completed this information on the components and the designs, we were then able to move forward in constructing a winning airplane.

1.3. Preliminary Design

The preliminary design entailed a more detailed description of the basic design that was further investigated in order to find the best possible ways to improve the design. Through a more thorough investigation, decisions could be made regarding specifics of individual components, such as size of the tail and the wingspan. Finalizing the component information and details also occurred during the period using various figures of merit.

1.4. Detail Design

Moving on to the detailed design, we focused only on the design that met our requirements and criteria. The entire design was first generated on a computer using AutoCAD. Component research and selection was finalized, along with their placements, making complete production of the plane possible. Using ECalc, statics, and structural stability techniques, we were able to investigate the stability and capabilities of our designed plane. The fuselage went under various changes in order to secure the best possible performance. Overall, the detail design was the finalizing of the efforts put into the conceptual and preliminary designs, thus yielding our final entry to the 2002 Cessna/ONR Competition. Our choice for the final design is a cylindrical fuselage capable of carrying 24 softballs and requiring only one motor. This design was the best choice in rated aircraft cost and figures of merit.

2. Management Summary

The Miami University team that undertook the Cessna/ONR Student Design/Build/Fly Competition consisted of the five seniors enrolled in their capstone experience classes, as well as six additional members. These members were of a wide range of majors and interests. Given this fact, the team had to divide up the workloads that would help the students compliment one another.

2.1. Architecture of the Design Team

The initial division of the team members was done with a very systematic approach. Our first step of the division was to meet and determine all areas and subjects that would be necessary to complete in order to manufacture an airplane. We then spoke freely with each other in order to find the raw talents, interests, and contributions that each student could give toward the overall objective. Next, we matched students along with others where similarities existed and could fuel a productive environment. The number of students assigned to a particular task was proportional to the perceived importance of that task. We also made clear that the assigned areas did not have definitive borders, and all students may help other team members in other areas.

Table 2.1 contains a listing of the Miami University team that entered the competition.

2001-2002 AIAA DBF Team Management

	Name	Year	Major	Major Contributions
1	Adam Moos *	SR	Engineering Management	Written Reports, Batteries, Wings
2	Nicholas Nemecek *	SR	Engineering Management	Servos, Braking System, Fuselage, Stress Tests
3	Brandon Tudor *	SR	Physics	Budget/Proposals, Written Reports
4	Zach Tweardy *	SR	Engineering Management	Materials, Technical Drawings, Empennage
5	Sarah Wyse *	SR	Physics	Motors, Pilotage, E-Calc Materials
6	Ryan Avery	SO	Undecided	Basic Construction
7	Lennie Marsh	JR	Marketing	Payload Carriage, MU Inventory
8	Raoul Rausch	SR	Engineering Physics	RAC Optimization
9	Chris Ruifrok	SR	Finance	Wings, Batteries
10	Kate Schafer	JR	Sociology	Payload Carriage, MU Inventory
11	Daniel Vanes	SO	Finance	Softball Acquisition, Aeronautical Knowledge

Dr. Osama Ettouney - Advisor (Engineering)	Dr. Richard Walker - Advisor (Aeronautics)
Dr. James Stenger - Advisor (Engineering)	Mr. Tom Schroder - Advisor (Aeronautics)

* denotes SR enrolled in capstone class

upper classmen =	7
under classmen =	4
Total	11

% under classmen = 36.4%

Table 2.1. Team members and organization with member contributions.

This list includes the major contributions or assigned work for each member of the team. During meetings, however, every member contributed some input and some effort into every category. Every member was included on the basic plane construction, which included the fuselage and wings. All aspects of the project were very open to all thoughts from all members and only moved on, based on the satisfaction of the team as a whole.

Each area of research was not assigned leaders. The group felt that the overall objective itself should keep the student on task and following the team's overall schedule. The basic concept was that the students would feed off their team member's contributions, enthusiasms, and support.

2.2. Schedule Control and Configuration

A schedule was created at the beginning of the school year. This schedule described in detail the deadlines and other project milestones that the team would have to meet. A timeline forecast was also created that represented the estimated times that the areas of research and work would begin and finish. This provided a set schedule that gave the students targets in the progress and requirements for certain tasks that may need certain information in order to complete its own requirements. A Gantt chart showing the schedule breakdown is presented in Figure 2.1.

As a whole the group held meetings with the advisors once a week in order to give an update on the progress of the report and the future goals and ambitions of the team. This meeting was also held as an investigative period for other team members and the aeronautics advisors to review the work of others and give their opinions or perceptions. This maximized the feedback of all members and helped the motivational progress for all members to do the best job possible on their respective tasks. The main designing took place during these times, given the amount of input and insights that could be contributed.

In addition to these weekly meetings, a separate weekly meeting was scheduled with engineering advisors. These meetings were attended by the five students that were enrolled in their capstone experience classes. During these meetings, the students communicated with their advisor the current progress of the project, progress of the report, kept track of minutes, discussed research, future plans, and the tracking of timecards. These meetings served as management meetings, becoming a reflective period on what the students had learned and discussions on directions to take.

ID		Task Name	Duration
1		Overall Airplane Analysis	52 days
2		Detailed Design	44.5 days
3		Purchase Parts	36 days
4		Assembly of Airplane	50 days
5		Rework and Redesign	68 days
6		1st Flight	1 day
7		Test Flights and Practice	29 days
8		AIAA/449 Report	45.5 days
9		449 Prelim Presentation	1 day
10		449 Prelim Report Due	1 day
11		5 Copies of AIAA Report Due	1 day
12		AIAA Competition	2 days
13		449 Final Presentation	3 days
14		449 Final Report/Essay/Evals D	1 day

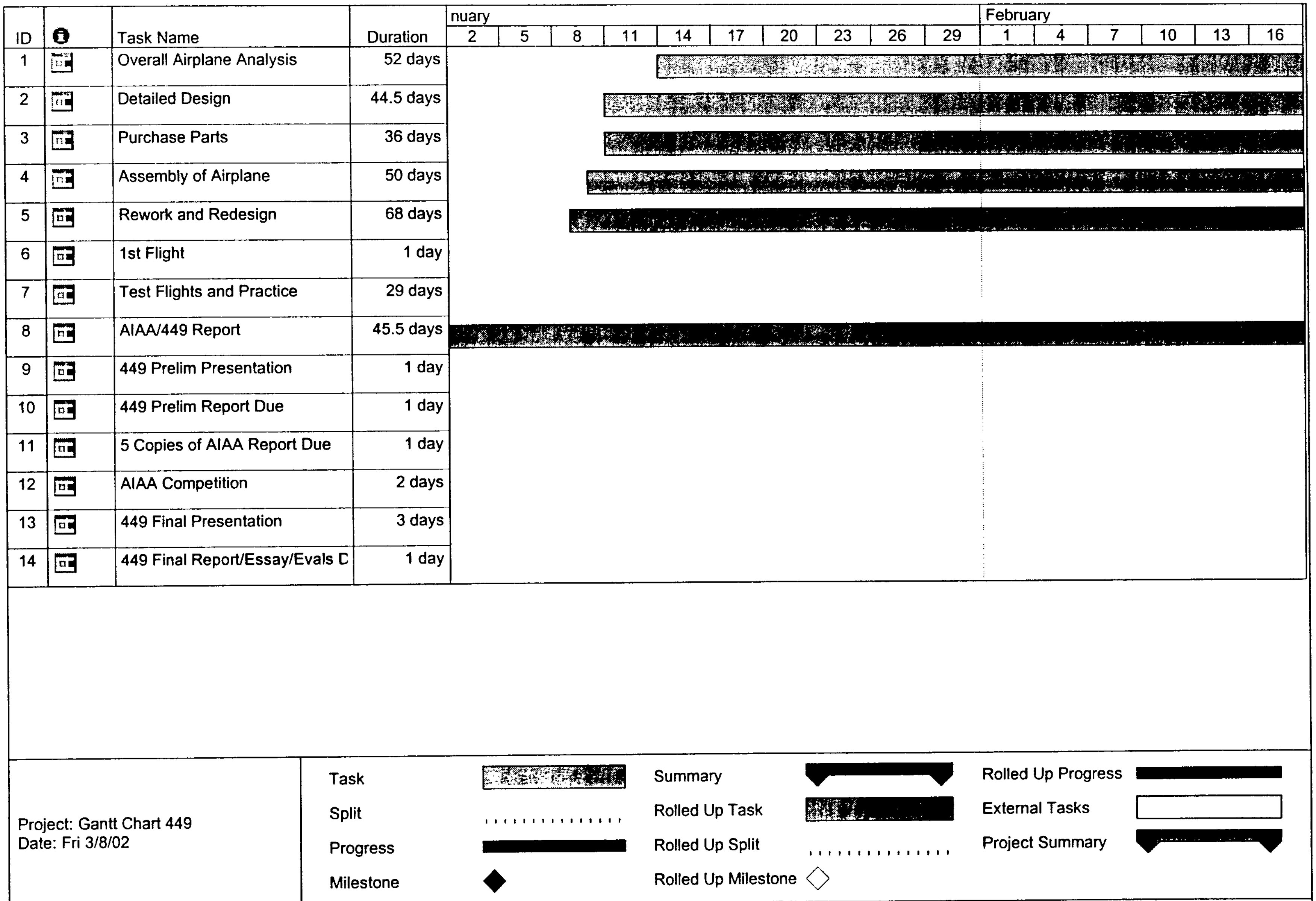


Figure 2.1. Gantt chart showing management plan used for team scheduling.

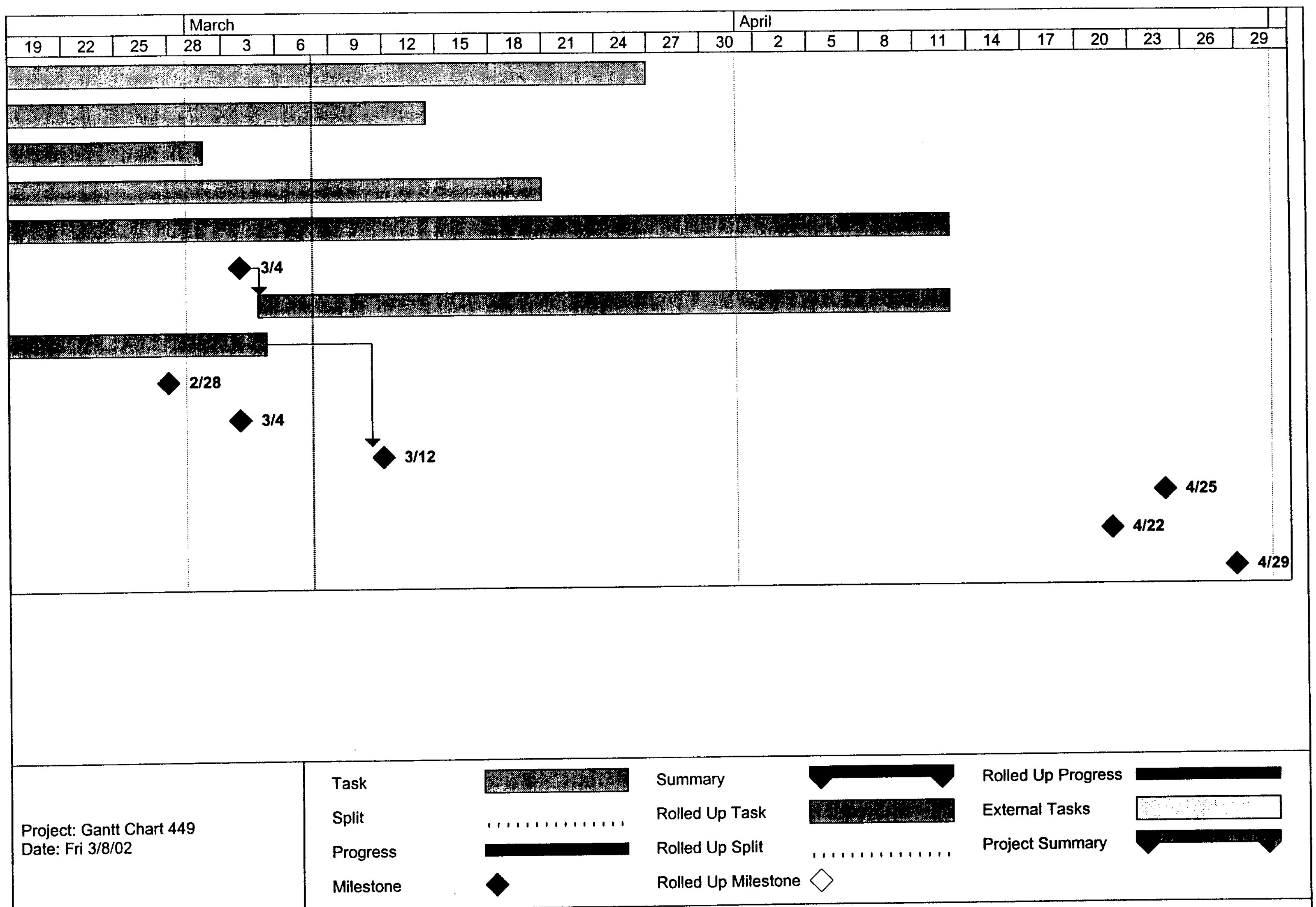


Figure 2.1. Gantt chart showing management plan used for team scheduling.

3. Conceptual Design

After having completed a permanent meeting schedule and the division of focused contributions, the team began the conceptual design phase. The basic premise of the overall approach to designing our plane was to adopt the steps to the design process and to use a problem solving methodology. A design process allows for a project to cover all areas thoroughly and to completely use all information available to complete a project with excellence.

The first step to the design process was to develop a clear problem definition. Our problem definition had to be as specific as possible, including objectives, goals, and constraints. Again, our objectives were to design, build, and fly a remote controlled airplane that would give us the highest possible score. Constraints became especially important due to the numerous rules and regulations that are mandated by the competition.

The next step in our design process was to do extensive research. At the beginning, we quickly found out that our personal experiences and knowledge were not very extensive in aeronautics and radio-controlled aircrafts. We utilized the most recent resources we could find on the specific subject of designing radio-controlled aircraft through books and magazines, while also learning basic concepts of aeronautics through similar means. We also found that the aeronautics professors were a tremendous resource and were able to teach us many of the basic concepts that we needed to be familiar with. After having familiarized ourselves with the project we were about to undertake, the team took all constraints, abilities, and advantages of our situation and began to brainstorm necessary attributes of the components and preliminary designs.

3.1. Design Parameters Investigated

In order to produce feasible conceptual designs, the team looked to break down the plane into many different areas of focus. This included the examination of certain components that would be necessary in order to complete the best final design possible. These components included the electric batteries, servos, motor and propeller, airfoils, and center of gravity analysis.

The group concentrated on having a general criterion for all parts that would help meet the overall objectives of the team. These attributes include having the lightest components possible in order to increase the speed of our aircraft. This determined weight must not hinder the performance or technical requirements needed for flight. Also included were to try to obtain materials at a small price. This again must not limit the performances of the components to the degree that they affect the flight of the plane. Cost was included in order for the team to recognize certain sacrifices that might have to be made. An example of this is to check the school's inventories in order to use all capable materials that are of no direct cost to our budget or income from certain applied grants. We had to recognize that the initial budget was limited. Ease of manufacture was also important to keep in mind. Manufacturability can influence the time of production, quality of production, and even the completeness of production. Using materials that were familiar to team members and that could be dealt with would help the overall success of our project.

3.1.1. Batteries

The airplane must have a battery power system that weighs no more than five pounds. It was important to the team to find a plane design that would affirm which size of battery would best fit into the fuselage. When researching, it was known that the main distributor of R/C batteries is SR Batteries. Through research on the SR Batteries' website, we were able to break down our options, and come up with a few solutions. The team had the option of using the battery the previous Miami University team used or purchasing a new one. The specifications and requirements on the batteries last year were the same as they are this year, allowing us this option. The team decided to also investigate other battery options to determine whether it would be worthwhile to spend some of the budget upgrading batteries.

The batteries used last year were the 2400 max series produced by SR Batteries. To use these batteries again the team would need to have them reconfigured to fit into the new design, for a small cost. The team also looked into new batteries such as the CP-3600CR, N-3000CR, and the N-2400SCR. These batteries were to be considered when trying to create the best possible configuration for the final design.

3.1.2. Servo Analysis

In order to control the plane during flight it would need four or five servos. These servos would control the ailerons, elevator, rudder, and throttle. Through the team's research, we found servos come in various sizes and technical performance. It was also found that they would not be an issue for the overall weight of our plane since they weigh no more than two ounces each. When researching the servos we focused on Futaba brand servos. A Futaba transmitter had been obtained from the inventory in the Miami University Aeronautics Department. This again saves us money but then limited us to use Futaba brand servos. Also in the inventory was a set of Futaba servos that was used the previous year for the Cessna/ONR Competition that was in great condition.

Our research on Futaba servos was primarily done at the Futaba web page. After researching the various servos and looking at the Futaba servos that are on last year's plane we decided it is best to reuse the ones that were used last year. The model number for these servos was the S3001 Precision BB. We found no great dissimilarities with them when compared to similar models when studying the component's weights, speed, torque, or dimensions. The team felt that the 42.0 oz.-in. would be enough to satisfy our needs as far as controlling the flight of our plane considering similar conditions as last year's plane. Since these servos performed very well and were still in excellent working condition, the team decided to use them for this project as well.

3.1.3. Powerplant

The primary function of the motor is to provide the power to turn the propeller. Together, they form the power plant. The propeller, or prop, mounted on the front of the motor, translates the rotating force of the motor into a forward-acting force called thrust that helps to move the airplane through air. In order to decide on the types of attributes our power plant must obtain, the group used a very important computer program to help with the deliberations.

In selecting a motor and prop, a general knowledge of some motor parameters is needed. *ElectriCalc*, or *ECalc*, is one of the most powerful software tools designed specifically for electric flight and can be very helpful in making a decision on the selection of airplane components. Its main screen provides all the information one would need to evaluate the performance of a given electric flight system. There are several motor parameters included on the main screen that help decipher flight variables.

Three of these variables are three motor constants: Kv, Rm, and Izero. Kv is expressed in RPM/Volt. It indicates how fast the motor would turn for a given voltage if there were no internal resistance. It is inversely proportional to Kt, the torque constant. The next motor constant, Rm, is expressed in million ohms. It characterizes the equivalent internal resistance of the motor. The last motor constant, Izero, is expressed in amps. It indicates the amount of current necessary to turn a motor without load, or how much current does not contribute to the output power.

The rules specify that all motors must be from the Graupner or Astro Flight families of brushed electric motors. More specifically, we compared the advantages and disadvantages of a cobalt 60 and a cobalt 90 motor, the cobalt 90 motor being more powerful. These two sizes are the only two brushed electric motors powerful enough to support an aircraft of the expected size and weight. The Aeronautics department at Miami University has a cobalt 90 motor, giving us our requirements. Therefore, the Astro Flight 90 was considered, as well as the Astro Flight 60.

3.1.4. Airfoil

The team felt that with RC Airplanes, it isn't necessary to perform laborious calculations for each potential airfoil. Instead, direct comparison of the curves and coefficients of candidate airfoils is more easily done.

Final selection of the airfoil depends on the design and how one wants the airfoil to perform, i.e., its "mission profile". For a sport model, an airfoil with a high maximum coefficient of lift, low drag, moderate pitching moment, and a gentle stall would be preferable. These, therefore, became our desired criteria due to our goals of speed, high lift qualities, and ease of handling.

3.1.5. Center of Gravity

One of the crucial aspects our team felt that we would have to deal with is the center of gravity (cg) of our airplane. The center of gravity is the average location of the weight of the airplane. The location of the cg is determined by establishing a datum line somewhere on the airplane. The weight of each component of the airplane is then multiplied by their distance from the datum to give a moment about that point. The moments are then summed and divided by the total weight of the airplane to give a distance from the datum to where the cg is located.

According to our advisor Dr. Walker, the wings of an airplane are generally attached to the fuselage such that the cg is at 25% of the airfoil. In other words, the cg is one quarter of the chord length back from the leading edge. The relative position of the cg and the center of lift of the airfoil have a large effect on the flight performance of the airplane.

This was important for our airplane design for many reasons. First, our design has to be capable of flying the course with and without a payload. Because of this, we decided to design the plane so that the cg is located in the same place when loaded and unloaded so that it provides for similar stability of flight. This is why the design evenly distributes the payload over the cg.

We then based the design of the remainder of the plane on the already established location of the cg at the quarter chord location. This is done by varying the distance the rest of the components were located from this cg. For instance, the length of the tail was determined by how much weight was needed to counteract that on the forward part of the plane. By proceeding in this manner we were assured that our cg was in the optimum location for flight performance.

3.2. Concepts Investigated

Concurrent with the ongoing research of the component requirements and acquisitions, the team approached the basic designing and analysis of a collection of preliminary brainstorming results. Because of doing these designs alongside the component research, we had a good idea as to what the designs were to need. Due to the constraint of housing softballs, which have a large volume, the team concentrated on having a larger fuselage that would be able to carry the maximum of 24 softballs. Carrying this many softballs would result in the highest possible score in that area.

The team's approach was to summarize what was successful in the past. To help enlighten students who had not participated in the competition in previous years, pictures that had been taken in the 2001 event were handed around to help the students visualize the possibilities. From these pictures and insightful ideas, three major preliminary designs were approached in full. These designs were a flying wing design, a basic cylindrical fuselage design, and an airfoil fuselage design with a tail.

3.2.1. Flying Wing

The flying wing design in Figure 3.1 focused on having the possibly of producing a triangular shaped fuselage that would be thick enough to hold all twenty-four softballs, yet be sleek enough that it will be aerodynamically efficient. This design was designed to have no empennage, with the possibility of retractable landing gear. Retractable landing gear would be expensive and could also possibly hamper the interior components of the fuselage. The team's main perspective however, was for this design to hopefully be the most aerodynamic, making up for a possible wide fuselage. It was viewed that it may also have small fins on the tips of its wings to assist in the handling of the vehicle. An initial thought of this design was that it would be difficult to manufacture and may possibly be too thick to keep consistent with our criteria of having a very fast plane and to eliminate all possible drag. The main design appealed to the team because of its unique and challenging design. The estimated rated aircraft cost for this design in given in Figure 3.2.

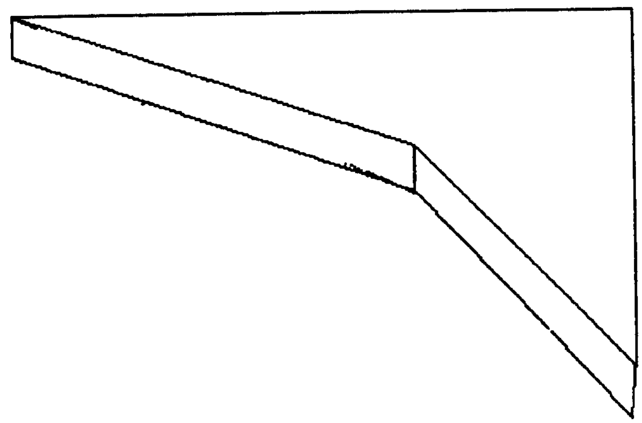


Figure 3.1. Flying wing

Rated Aircraft Cost

Flying Wing

Coef.	Description	Value
A	Manufacturers Empty Weight Multiplier	$100
B	Rated Engine Power Multiplier	$1,500
C	Manufacturing Cost Multiplier	$20
MEW	Manufacturers Empty Weight Multiplier (lb)	25
REP	Rated Engine Power	4.92

MFHR Manufacturing man hours

Wings:	
8 hr/ft wing span	40
8hr/ft max exp. Wing chord	16
3 hr/control surface	6
Total=	**62**
Fuselage:	
10 hr/ft	40
Total=	**40**
Empennage:	
10 hr/vertical w/control	10
10 hr/horizontal	10
Total=	**20**
Flight Systems:	
5 hr/servo or motor control	30
Total=	**30**
Propulsion Systems:	
5 hr/engine	10
5 hr/prop	10
Total=	**20**
Grand total=	***179***

RAC= 13.32

Figure 3.2. Estimated rated aircraft cost for flying wing concept.

3.2.2. Cylindrical Fuselage

Our basic cylindrical fuselage design as seen in Figure 3.3 was a very typical airplane design. Contained in our first drawings were the fuselage, empennage, wings, power plant, and landing gear to make up the structure of the plane. It was decided early on to have to have the landing gear fixed and not come up. Limiting the fuselage to contain as little volume as possible was a large concern for the team. This fuselage would be capable, however, of housing all the components and the maximum number of softballs. The softballs would run in a 2x12 formation down the fuselage, not interfering with any internal components. Another goal was to limit the wingspan as much as possible to cut down on the overall weight. In order to do this, the team came upon the ideas of supporting the base structure of the wing with some kind of carbon fiber. This fiber would run a specific length through the wing. So that this would not interfere with the internal components of the fuselage, the wing would be its own individual piece and be connected on the bottom of the fuselage. This would help with the overall stability of the plane as a whole. These were all possibilities that would need to be sorted and investigated over time. The estimated rated aircraft cost for this design is given in Figure 3.4.

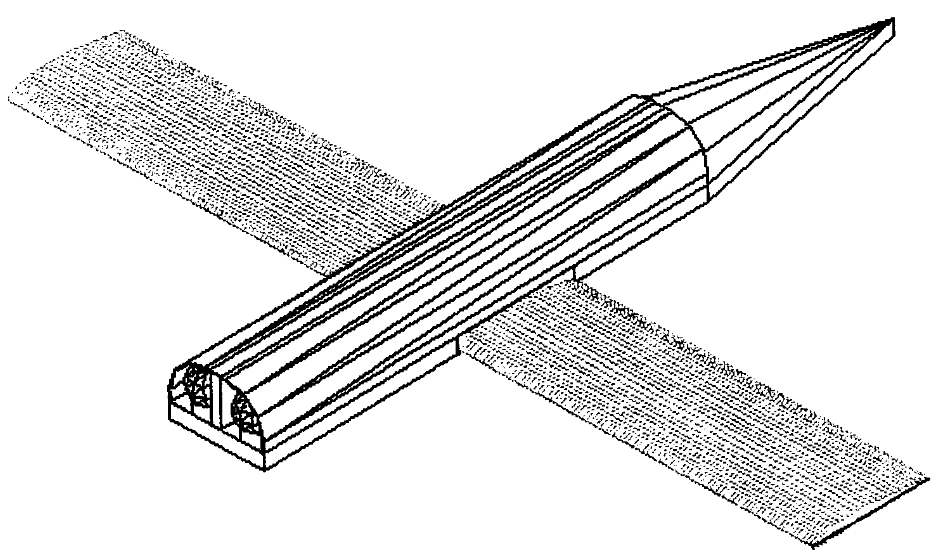


Figure 3.3. Cylindrical fuselage

Cylindrical Fuselage

Coef.	Description	Value
A	Manufacturers Empty Weight Multiplier	$100
B	Rated Engine Power Multiplier	$1,500
C	Manufacturing Cost Multiplier	$20
MEW	Manufacturers Empty Weight Multiplier (lb)	23.6875
REP	Rated Engine Power	4.92

MFHR Manufacturing man hours

Wings:		
8 hr/ft wing span		56
8hr/ft max exp. Wing chord		8
3 hr/control surface		6
	Total=	**70**
Fuselage:		
10 hr/ft		38.2
	Total=	**38.2**
Empennage:		
10 hr/vertical w/control		10
10 hr/horizontal		10
	Total=	**20**
Flight Systems:		
5 hr/servo or motor control		25
	Total=	**25**
Propulsion Systems:		
5 hr/engine		5
5 hr/prop		5
	Total=	**10**
Grand total=		***179***

RAC= **13.01275**

Figure 3.4. Estimated rated aircraft cost for cylindrical fuselage concept.

3.2.3. Airfoil Fuselage

The airfoil fuselage design with tail as shown in Figure 3.5 was our last initial plane design. This design would consist of a fuselage in the shape of an extruded airfoil, a pair of wings, an empennage, and a set of landing gear. It had not been decided whether this landing gear would be retractable of fixed. Some preliminary ideas revolved around the choices of airfoils for the fuselage and the wings and how they would relate and interact with one another. Initially, it was settled to concentrate on a wing built to gain maximum speeds for the wings, and for the fuselage, to adopt a stable airfoil capable of great lift. A sailplane airfoil was investigated for this component. The storage of the softballs became an initial concern as to how they should be arranged. Some of the possibilities ranged from 4x6, 8x3, 2x12, or even two sets of 4x3. These setups were viewed as important due to their involvement in the actual size and shape that would make up this fuselage. It is directly related to how the thick or thin the fuselage width would be, which then immediately impacts the length. These ratios would be determined by the selected airfoil. Overall, this was looked at in very high regard in the first meetings of our team, but it was eliminated due to the difficulty of production and some questions of stability and payload capacity. Figure 3.6 shows the estimated rated aircraft cost for this design.

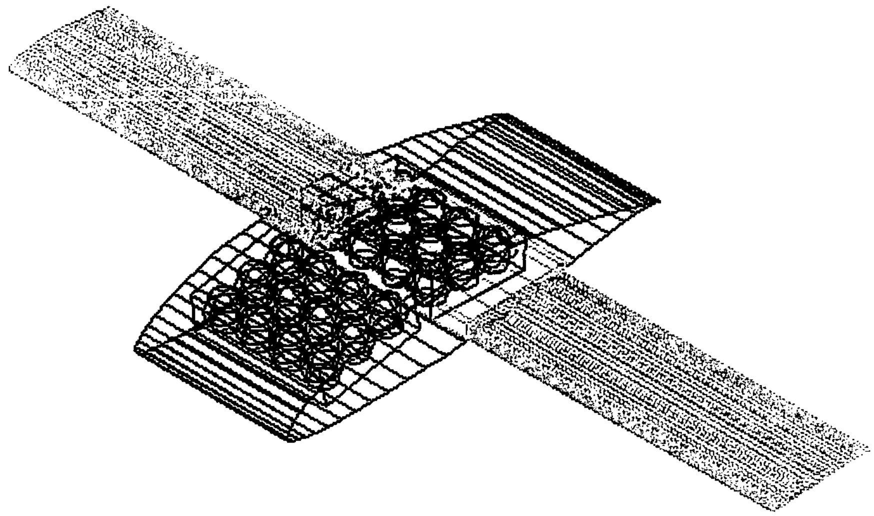


Figure 3.5. Airfoil fuselage

Rated Aircraft Cost

Airfoil Fuselage

Coef.	Description	Value
A	Manufacturers Empty Weight Multiplier	$100
B	Rated Engine Power Multiplier	$1,500
C	Manufacturing Cost Multiplier	$20
MEW	Manufacturers Empty Weight Multiplier (lb)	25
REP	Rated Engine Power	4.92

MFHR Manufacturing man hours

Wings:		
8 hr/ft wing span		56
8hr/ft max exp. Wing chord		12
3 hr/control surface		6
	Total=	**74**

Fuselage:		
10 hr/ft		35
	Total=	**35**
Empennage:		
10 hr/vertical w/control		10
10 hr/horizontal		10
	Total=	**20**
Flight Systems:		
5 hr/servo or motor control		30
	Total=	**30**
Propulsion Systems:		
5 hr/engine		10
5 hr/prop		10
	Total=	**20**
Grand total=		***179***

RAC= **13.46**

Figure 3.6. Estimated rated aircraft cost for airfoil fuselage concept.

3.3 Figures of Merit

In determining the final design to progress with, the team created a figure of merit worksheet for all three designs. The figures that each design is rated on are manufacturability, carrying capability, speed of loading, aerodynamics, maneuverability, and stability. A value was given to each design for these criteria as shown in Figure 3.7. These values were assigned by referring back to our original objectives of wanting to construct the fastest plane possible while still taking into account our other constraints such as manufacturing capabilities. The team decided to proceed with the cylindrical fuselage design due to many reasons; one of those having a great effect on the decision was the manufacturability of this design.

Figures of Merit Worksheet

Preliminary Designs	Rank Multiplier (10 points per category)	Cylindrical Fuselage	Airfoil Fuselage with Tail	Flying Wing
Manufacturability	4	9	6	4
Carrying Capability	4	10	10	10
Speed of Loading	3	8	7	6
Aerodynamics	4	6	7	8
Maneuverability	2	7	8	7
Stability	3	7	7	5
Total	200	159	150	135

Table 3.7. Figures of merit worksheet used to select preliminary design with selection for future progress highlighted.

4. Preliminary Design

Once the conceptual design phase was complete and the team decided to proceed with the cylindrical fuselage design, many other issues arose. These include decisions on specifications for the main components as well as choosing materials for constructing these components.

4.1. Component Material Selection and Figures of Merit

Before researching different materials, we first decided on four major criteria that the material must possess. These criteria include (1) cost, (2) durability, (3) weight, and (4) manufacturability. The following are explanations of each of these criteria.

Cost: This parameter is not only critical in the overall scoring of our airplane; it is also critical because we are limited in the amount of money we can spend on this project. We are looking for a relatively inexpensive material.

Durability: This parameter is critical to the performance of the airplane. The material of the airplane must be able to withstand the forces that will be applied to it in flight.

Weight: The weight of the airplane is directly related to the speed of the airplane. The lighter the plane is, the faster it will be.

Manufacturability: This parameter refers to the ease to which we can work with the material. Obviously the harder it is to work with, the longer it will take to build. Having a material that is easy to work with allows us to concentrate on flying and testing the plane, rather than building it.

We focused on three general materials that were comparable based on these four criteria. These materials were balsa wood, polystyrene, and aluminum. We then developed a criteria matrix so we could determine the benefits of each material. These four criteria were weighted since some were more important than others. We rated durability the highest because if the airplane is destroyed in the air, then there is no chance of finishing the competition. After weighing each of our selected materials based on these four criteria, we determined that in some cases combinations of materials would be the most beneficial way of constructing components of the plane. Below in Table 4.1 is the material criteria matrix used for selection of materials to be used in the fuselage, wings, empennage, and landing gear.

Figures of Merit Worksheet

Components

		Manufacturability	Durability	Weight	Cost	Total
Fuselage	Rank Multiplier (10 points per category)	3	5	3	4	**150**
	Balsa/Foam Composite	8	8	7	8	**117**
	White Styrofoam	8	0	9	10	**91**
	Blue Styrofoam	8	0	8	10	**88**
	Balsa	7	6	6	7	**97**
	Foam with titanium sheeting	5	3	8	10	**94**
	Aluminum	9	10	4	5	**109**
Wings						
	Foam/Carbon reinforcements/balsa sheeting	7	10	5	7	**114**
	Foam with carbon reinforcements	8	5	8	9	**109**
	Blue Styrofoam	9	0	8	10	**91**
	White Styrofoam	9	0	9	10	**94**
	Balsa/Foam Composite	8	4	6	8	**94**
	Foam with balsa reinforcements	8	3	7	9	**96**
Empennage						
	Balsa	9	7	8	8	**118**
	Balsa with carbon reinforcements	7	9	7	7	**115**
	Balsa/Foam Composite	5	6	7	8	**98**
	Foam with carbon reinforcements	6	6	8	9	**108**
Landing Gear						
	Aluminum	9	10	5	10	**132**
	Carbon Tubing	7	8	8	6	**109**

Table 4.1. Figures of merit worksheet used in determining the best possible materials and processes for manufacturing the main components. Selections are highlighted.

4.2. Design Parameters

In order to focus our efforts on the design, the criteria and parameters needed to be retooled. Each area was to be reevaluated and broken down so that the team would be able to set new standards and expectations for each section.

4.2.1. Landing Gear

There were several different types of landing gear to consider during the design of our plane. Our first consideration was to use either fixed or retractable landing gear. Retractable landing gear adds no drag while the plane is in flight however it would add weight, increase the difficulty of manufacture, and add a servo to our rated aircraft cost. It also adds to the space needed in the fuselage. For these reasons, we decided on fixed landing gear.

Next we needed to consider whether to use a tricycle type landing gear or a tail dragger landing gear configuration. While a tricycle landing gear works very well and gives increased control on the ground, it adds a great amount of drag and weight. The tail dragger design adds less drag and weight than the tricycle while still giving us our desired level of control on the ground. This is due to the difference in wheel size for the tail dragger. This setup also allows for a great amount of prop clearance, while keeping a minimum of the external material required for the landing gear. Also, since one of our overall goals is the desire for speed at a low cost, the team decided that the retractable landing gear would be too expensive for our plane. In addition, retractable landing gear would add extra requirements regarding power and space. For these reasons we have decided to use a tail dragger landing gear configuration. After some research, we found that the ideal material for us to use for our landing gear is aluminum due to its lightweight, strength, durability, and flexibility.

4.2.2. Wing

The teams looked into having a single piece wing that would attach to the underbelly of the plane. This wing would be at its minimum length and as decided in the conceptual design phase, would have some sort of support reinforcement. This support would be a strip of carbon fiber running the full 7 feet of the wingspan. The support would be implanted by cutting the wing into two slices along the chord. Cutting along the chord would maximize the impact of this fiber. A thin groove may also have to be developed in the wing in order to fit the fiber into place, keeping the wing firm when it is put back together. It was decided that whatever is used to squeeze and hold the two wing pieces together will have to be strong. These actions completed the designs of the wing, until actual testing began on them.

4.2.3. Fuselage

In the MU team's eyes, the fuselage may have been the most important aspect of our plane's design. The fuselage was responsible for carrying all twenty-four softballs safely and securely. This simple requirement was accompanied by the requirements to hold all necessary internal components, while keeping a stable center of gravity when carrying the payload and when not. Finally, when dealing with the payload, the fuselage would need to be designed for very easy to access the softballs , while

making no great changes to the structure and staying very aerodynamic. For this reason, the fuselage design may have logged in the most time from our group members.

The designed fuselage was first drawn in a T-form with a length of 6.5 feet. This design had the bottom part of the T as a wall separating two rows of twelve balls so that they do not bump into one another. This wall had holes cut into it, which minimized the weight of the whole fuselage. Small indentations were cut out in order to hold the balls in place during the flight. The balls would also be held in place with the assistance of the fuselage hatch, which will fit snugly along the top of the fuselage.

This housing, made of aluminum, was to be used to cover the fuselage and be the hatch that will allow quick entry to the payload placement in the plane. The aluminum is easily bended around the fuselage, allowing each entry to the payload compartment. Also it is a very smooth, making it very aerodynamic. Fastening the housing to the base of the fuselage had not been thought out yet, but there were definitely ideas. Some of these ideas were to use some kind of strong tape, Velcro, metal twist fasteners, or a screw related sealer. The only questions that needed answering for this decision were to wonder, "How fast is it?" and "How many times can it be reused?" Following through on meeting these quandaries would help the project complete a fast transfer of payloads during the competition.

The final key to creating a fuselage was to complete one that would completely balance the components so that the plane ran smoothly with or without the payload. This could be accomplished by simply placing the weights of the components in such a way that the weights that would be fixed in the internal fuselage would cancel each other out by balancing over the center of gravity. Having this accomplished would eliminate any possible hindrances or bias that may occur while flying the plane. This would be the final key to complete when approaching the plan of how to build a great and efficient fuselage.

4.2.4. Empennage

The elements of our tail that we needed to analyze and make decisions about were location, material, shape, and size. We decided to create our tail out of single sheets of ¼ inch balsa wood. The light weight as well as the ease of manufacturability of balsa played a large role in this decision.

We then analyzed whether or not to build the plane with the tail at the top or bottom of the fuselage. With the tail at the bottom of the fuselage it may be placed in the turbulent air coming off the wing and this would cause for poor flight characteristics. Therefore, we decided to place the tail at the top of our fuselage in order to clear it from the wake of the wing.

When deciding the size of the horizontal and vertical tail surface areas we looked at many different equations. We found that traditionally the horizontal surface area is 25% of the wings surface area. This fact was given to us by one of our aeronautic advisors. He also stated that the vertical tail surface area is traditionally half of the horizontal tail surface area. The tails placement behind the fuselage is a matter of analyzing the moments created on the plane and balancing them all out to where the plane balances on its center of gravity.

5. Detail Design

During the detail design phase, our team looked through the status of all of our components and parts to our plane in order to finalize all existing decisions to complete our final aircraft design. Along with any existing decisions that had to be made, our team combined to do certain tests and use design tools in order to evaluate our own decisions. This would help in our understanding of the exact capabilities of our plane, as far as the stability and other variables.

5.1. Component Selection

As was previously mentioned, the team had divided the research of components into sections to study in more detail. It was the team's goal to maximize the performance of the airplane design by making an informed decision on the components while still staying within the boundaries set by the competition and by the team.

5.1.1. Airfoil

As stated earlier, the team felt that with RC Airplanes it is not necessary to perform laborious calculations for each potential airfoil. Instead, direct comparison of the curves and coefficients of candidate airfoils is more easily done. Before these curves and coefficients are explained, one should know a few key definitions. Angle of attack (AoA) is the angle at which the wing strikes the air (in flight) measured from the chord line. Chord is a straight line that joins the leading and trailing edges. Reynolds number (Rn) comes from the aircraft's speed in miles per hour multiplied by its chord in inches multiplied by a constant K that changes with altitude.

Final selection of the airfoil depends on the design and how one wants the airfoil to perform, i.e., its "mission profile". For a sport model, an airfoil with a high maximum coefficient of lift, low drag, moderate pitching moment, and a gentle stall would be preferable. After taking this into consideration and consulting with our aeronautics advisor, we selected the SD7032 airfoil, which gives us the following lift characteristics in Figure 5.1.

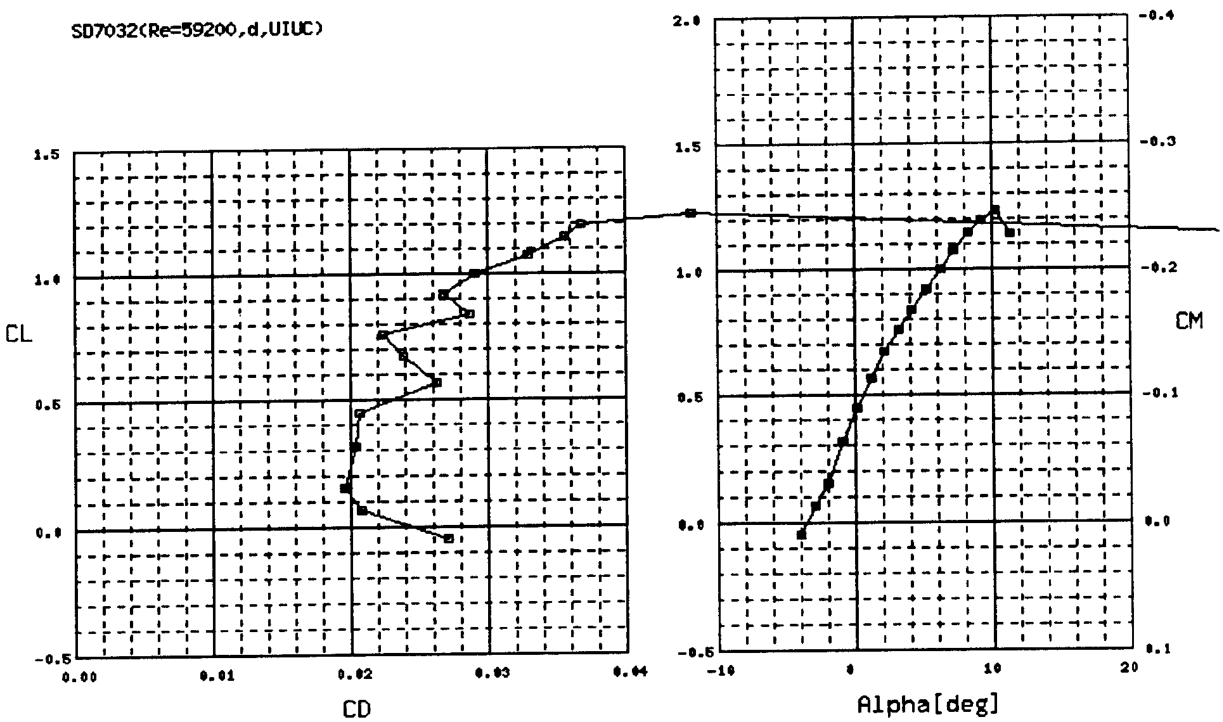


Figure 5.1. Lift and polar curves for SD7032 airfoil.

Data in lift curves and polar curves are given in terms of coefficients. C_L is for lift, C_{Do} is for profile drag, and C_M is for the pitching moment around the ¼-chord point. The lift curve on the right shows that the airfoil, for which this particular data was collected, generates positive lift at an AoA of minus 4 degrees. The peak of this curve represents the maximum coefficient of lift. The polar curve on the left shows very little increase in profile drag despite increasing lift.

5.1.2. Powerplant

To make decisions concerning what battery, motor, and prop combination to use, a computer program called *ElectriCalc* was used. Its main screen is divided into four panels: power, motor, drive, and flight parameters. When data on this screen is changed, the other parameters are automatically updated making for a very convenient and simple means of performing "what if?" scenarios.

Ultimately, we decided to go with the Astro Flight Cobalt 90 motor for financial reasons. Table 5.2 shows a figures of merit worksheet used in making this determination. We already had this motor in our stock from last year's competition. The question was whether we should leave the gearbox on the motor or go with direct drive. One of the rules of thumb as listed on Astro Flight's web page is that the propeller pitch speed should be a minimum of 2.5 times the stall speed of the model. Keeping the power parameters fixed, i.e. battery cells, and varying the prop size, it was impossible to find a prop size that would produce such results for the motor with a gearbox. For that reason, we decided to send the motor we had in to have the gearbox removed. This decision would also result in other advantages. For

example, there would be a reduction in weight due to the smaller prop used with the direct drive motor and the absence of a gearbox.

The Sanyo CP-3600CR fast charge battery cell was chosen to be the power supply for the airplane. The team decided that it would be beneficial to sacrifice some of the budget in this area, therefore batteries with higher capacities that the one already available to the team were investigated and this final selection was determined as shown in Figure 5.2. Each cell weighs 3.10 ounces and has a diameter of 1.02 inches and a length of 1.97 inches. Twenty-five cells result in a package weight of 78.7 ounces, remaining just below the 80-ounce weight limit. These flight cells have the highest capacity for the lowest weight and give the longest flight times. The CP-3600CR has a capacity of 3450 MAH (milliamp hours).

A 16.5 X 15 wooden prop will be used in combination with aforementioned battery pack and direct drive motor. Although a carbon fiber prop would offer more rigidity and less weight, only wooden props could be found in the size needed. As stated before, this size was determined by using ECalc to find the best possible outcome.

Figures of Merit Worksheet

		Output	Durability	Weight	Cost	Total
	Rank Multiplier (10 points per category)	5	2	4	4	150
Powerplant						
	Motor					
	1. Astroflight 60 Direct	6	10	9	3	98
	2. Astroflight 60 Geared	8	10	8	2	100
	3. Astroflight 90 Direct	**8**	**10**	**7**	**9**	**124**
	4. Astroflight 90 Geared	0	10	6	10	84
	Battery					
	1. CP-3600CR	9	10	10	6	129
	2. N-3000CR	6	10	10	8	122
	3. N-2400SCR	4	10	10	9	116

Table 5.2. Figure of merit worksheet used to select motor and battery highlighting selections made.

5.2. Final Configuration

Although the final selection had been made on the overall design, some investigation still needed to be done to determine the final configuration of the airplane. This included testing of the various components that were manufactured to confirm that they would perform in the way in which the team envisioned. Additionally, analysis was completed in determining the final placement of components of the design. The final estimated rated aircraft cost for this design can be seen in Figure 5.3

Rated Aircraft Cost

Cylindrical Fuselage

Coef.	Description	Value
A	Manufacturers Empty Weight Multiplier	$100
B	Rated Engine Power Multiplier	$1,500
C	Manufacturing Cost Multiplier	$20
MEW	Manufacturers Empty Weight Multiplier (lb)	23.6875
REP	Rated Engine Power	4.92

MFHR	Manufacturing man hours	
	Wings:	
	8 hr/ft wing span	56
	8hr/ft max exp. Wing chord	8
	3 hr/control surface	6
	Total=	**70**

Fuselage:	
10 hr/ft	38.2
Total=	**38.2**
Empennage:	
10 hr/vertical w/control	10
10 hr/horizontal	10
Total=	**20**
Flight Systems:	
5 hr/servo or motor control	25
Total=	**25**
Propulsion Systems:	
5 hr/engine	5
5 hr/prop	5
Total=	**10**
Grand total=	***179***

RAC= 13.01275

Figure 5.3. Final estimated rated aircraft cost for design.

5.2.1. Bending Moment Analysis on Wings

In order to see if our desired material was able to sustain the forces acting on the plane, we performed a bending moment analysis to find the maximum tensile and compressive stresses in order to compare these values to the yield strength of the Styrofoam.

The maximum load that the plane will experience will be during the tip-to-tip test prior to flight. In this test the plane will be supported by only the tips of the wings at the quarter chord to see if the plane will break.

In order to solve for the maximum stresses, we found the moment of inertia of the airfoil (I), the distance from the neutral axis to the outermost edge of the airfoil (y), and the moment (M) acting on the wing, due to the forces. The moment of inertia and the distance from the neutral axis to the outermost edge was found using AutoCAD.

From Figure 5.4, one is able to see that the moments on the wing cause the top of the wing to be in compression while the bottom of the wing is in tension. Due to the location of the neutral axis being farther from the top of the wing than the bottom of the wing, the compressive stress on the top of the wing will be greater than the tensile stress on the bottom of the wing. Using the variables M, y, and I, we found the maximum stress on the wing by using the formula:

$$\sigma_{max} = (M^*y) / I$$

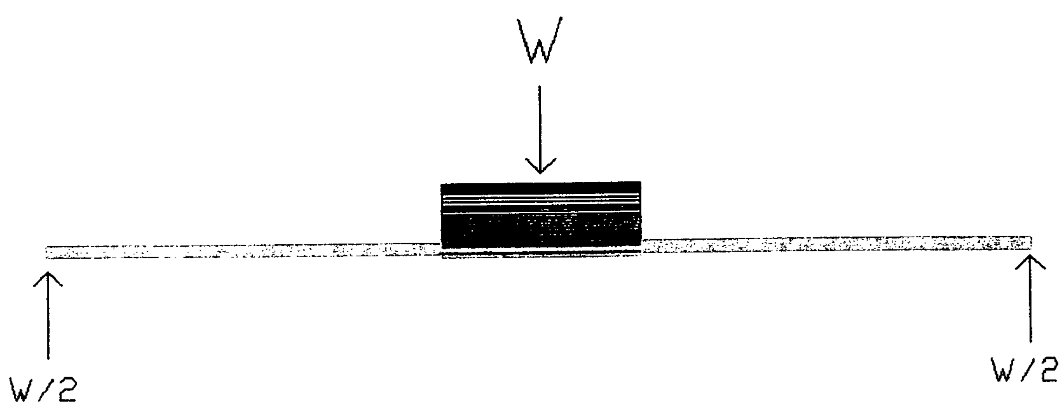


Figure 5.4. Free body diagram of tip-to-tip test.

Figure 5.5. Wing section loading.

Figure 5.5 depicts the actual loading on the wing. The maximum stress applied to the wing is 480.72 psi. This is much higher than the yield strength of Styrofoam, which is 72 psi. This led us to the conclusion that reinforcements are needed in the form of either a horizontal or vertical spar through the length of the wing. For reinforcement, we used a carbon fiber strip 0.03 inches thick and 1.0 inch in height to place across the wingspan. After performing the same calculations on the carbon fiber strip, we found a maximum stress of 70,766 psi. This is much less than the yield strength of the carbon, which is 300,000 psi.

5.2.2. Bending Moment Analysis on Fuselage

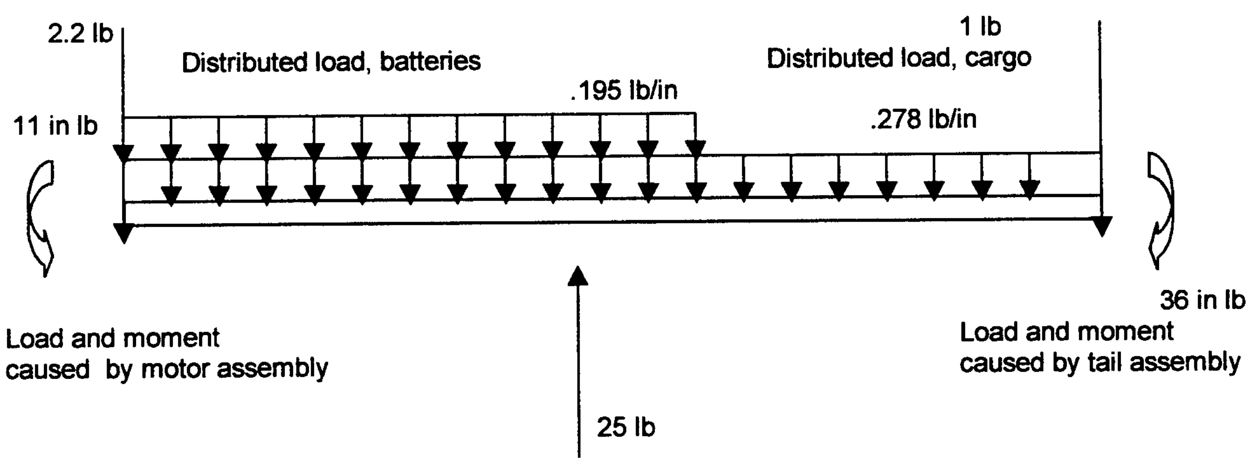
The key aspect that was considered in the fuselage is the bending moment caused by the distributed load of the various components of the plane including the payload. Our leading idea is to use a foam and balsa composite to create a strong and lightweight fuselage.

In order to determine the maximum stress in the member it is necessary to treat the fuselage as a composite beam. The ratio between the Modulus of Elasticity between the foam and the balsa was found to be n = 0.109. This allowed us to transform the composite components into a single material to perform the analysis.

First, the neutral axis of the cross-section of the fuselage was determined. When this was completed, the team then calculated the bending moment about the neutral axis. The team then analyzed the distributed loads to find where the maximum bending moment occurred and what the

magnitude is. At this point it was possible to calculate the maximum compressive stress at the furthest point from the neutral axis and compare it to the yield strength of the wood to find that the fuselage would indeed be strong enough while weighing only 2 pounds. The loading is illustrated in Figure 5.6, and on the following two pages showing the AutoCAD analysis. Thus, we have decided to construct our fuselage using a foam and balsa composite.

Distributed Loadings on Fuselage for Bending Analysis



Maximum bending moment = $-11-2x-.473/2x^2 = 170$ in lb

Figure 5.6. Bending analysis for fuselage.

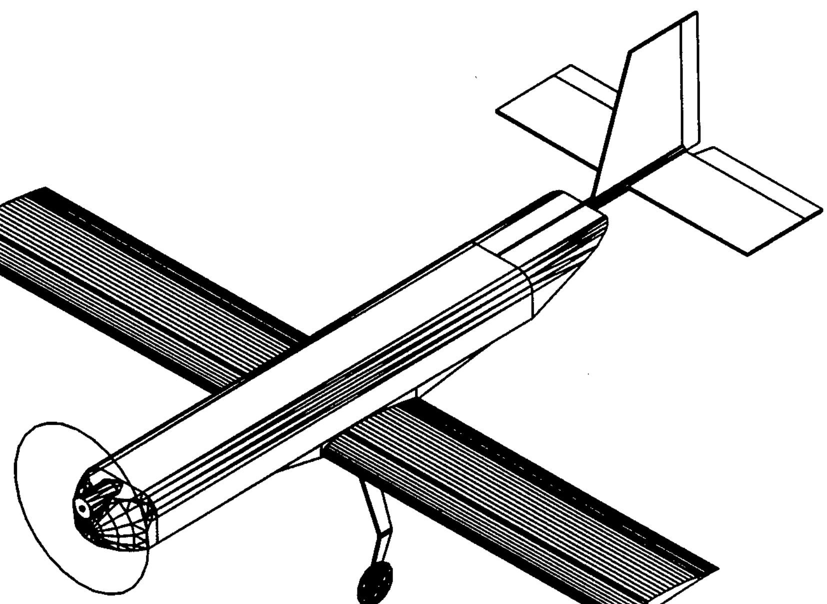
5.2.3. Weight and Balance

The final configuration of components was also determined by their weight and location. The design selected has conventional landing gear, therefore the center of gravity must be slightly behind the front landing gear in order to ensure that the plane will rest on all three wheels and not tip forward. It is also beneficial to have the center of gravity located at the quarter chord point on the wing as stated earlier for stability in flight. To accomplish this, the team weighed each individual component in order to find the exact needed placement of them along the fuselage. These weights created moments that had to be counteracted on the opposite sides. The fuselage was then designed around these factors so that it could be manufactured correctly. This balance would help offset all unwanted moments and support the structure as a whole. Table 5.4 was used in calculating the overall weight of the design and the placement of the components.

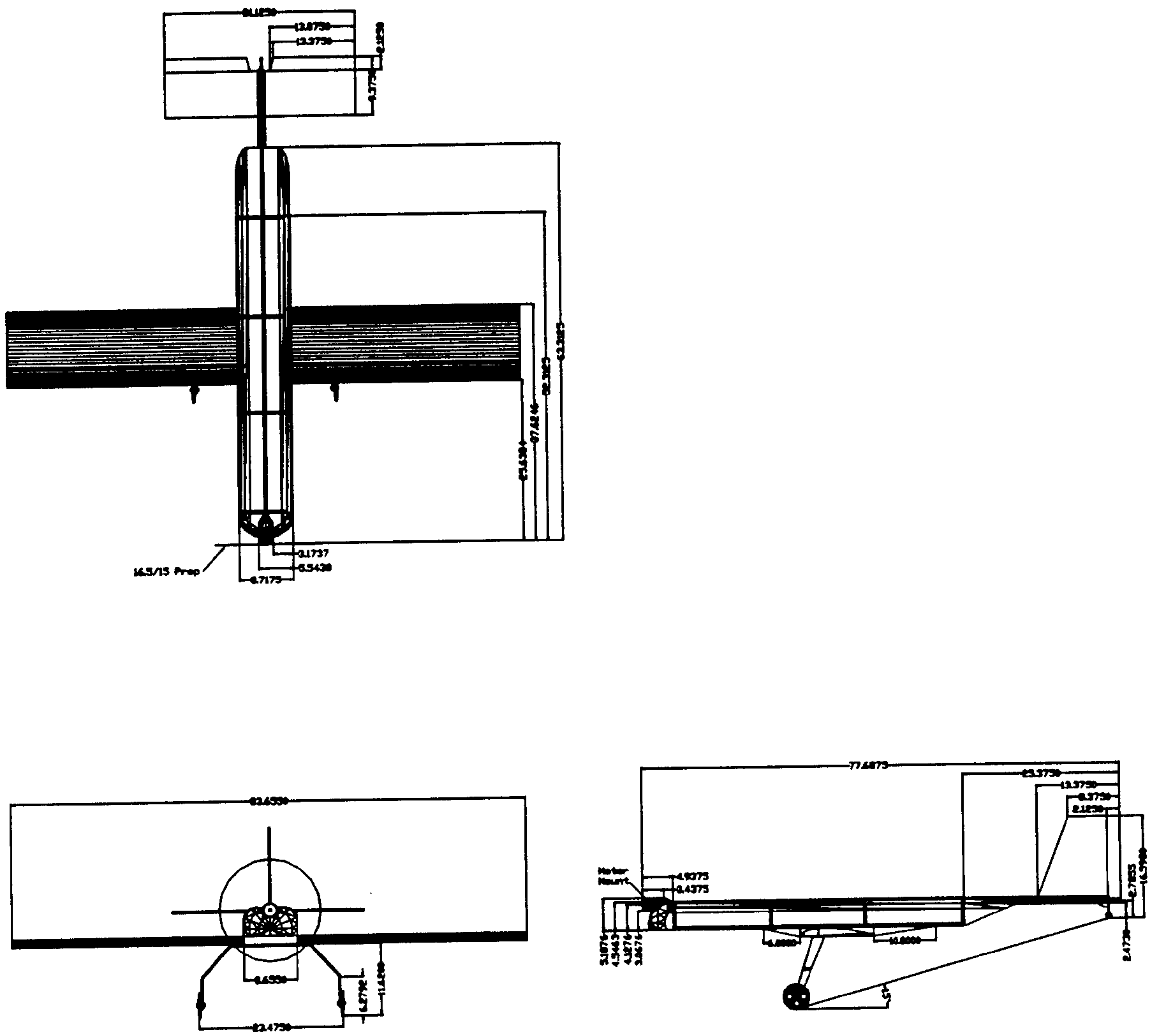
Estimated Weight and Balance of Final Design

Component	Estimated Weight		Distance from CG (in)	Moment (in*lb)
Fuselage	2.0625	lb	0	0
Wing	1.6250	lb	0	0
Tail Section	1.0000	lb	-36	-36
Landing Gear	1.0000	lb	0	0
Motor/Prop	2.0000	lb	27	54
Batteries	5.0000	lb	-1.3	-6.5
Radio	0.5000	lb	-23	-11.5
Servos and Wiring	0.5000	lb	0	0
Softballs	10.0000	lb	0	0
Total	**23.6875**	**lb**		**0**

Table 5.4. Estimated weights and balancing of final design.

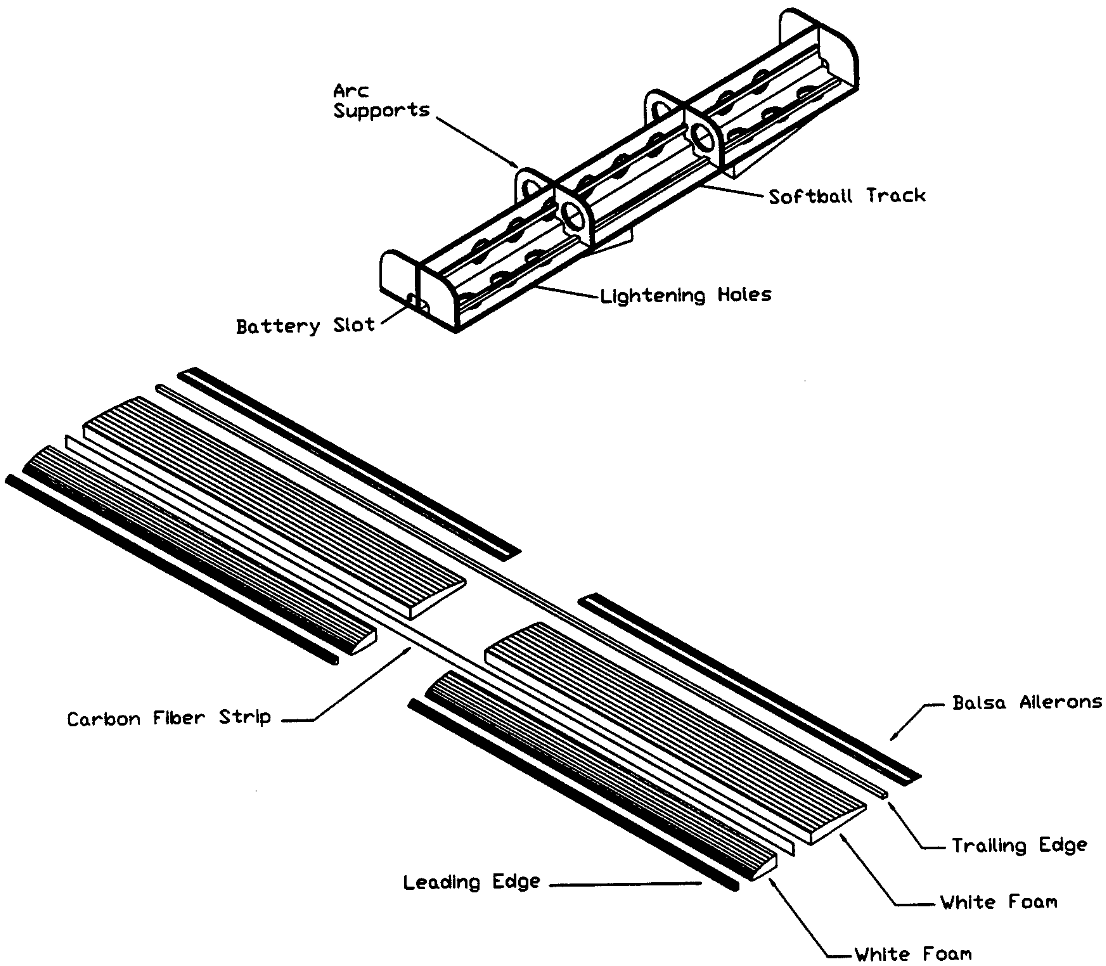


Miami University		
Final Plane Design	Team: AIAA Competition	
	Date: 3-3-2002	
	Scale: NTS	
	SH._1_ OF_5_	



Miami University

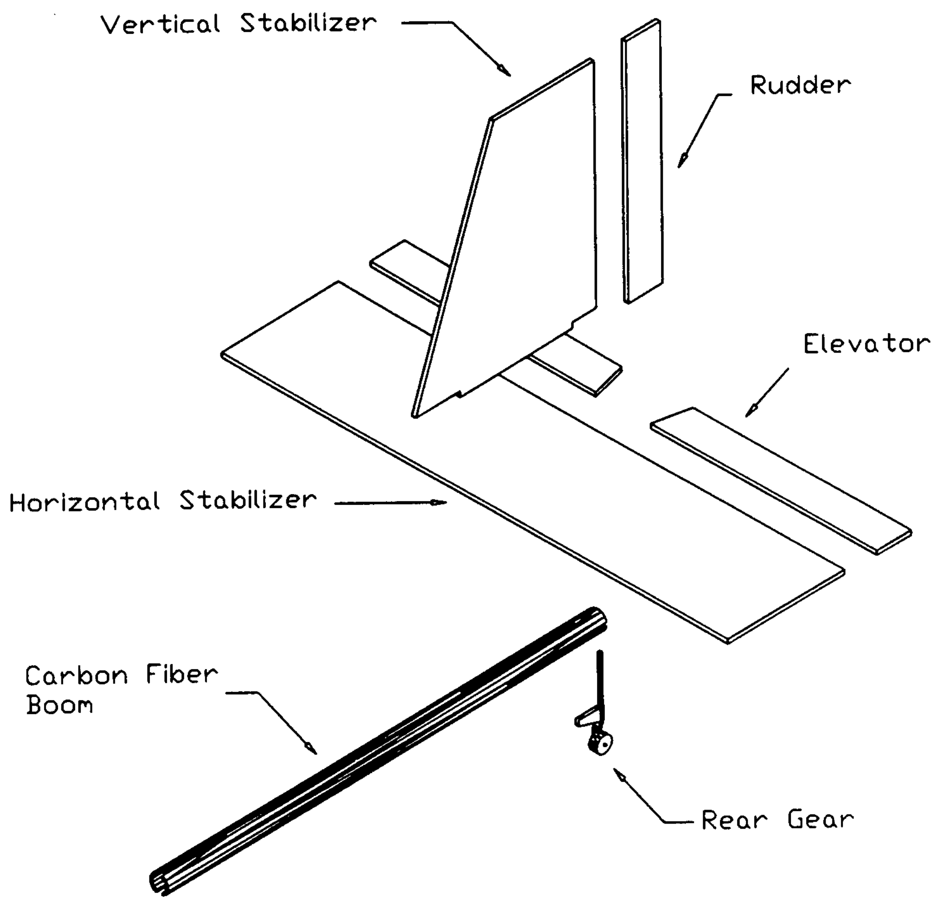
3-View Drawing of Airplane	Team: AIAA Competition
	Date: 3-3-2002
	Scale: 1:30
	SH. 2 OF 5



Note:
1. The fuselage T-form base and arc supports are composed of two sheets of 1/16" birch wood and 1/4" blue foam.
2. The ball tracks are 1/2" balsa wood.

Note:
1. Leading edge and trailing edge are made of 1/2" strips of balsa wood.
2. The entire wing is covered in 1/32" balsa sheeting.

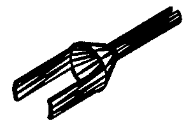
Miami University	
Fuselage and Wing Components	Team: AIAA Competition
	Date: 3-3-2002
	Scale: NTS
	SH. 3 OF 5

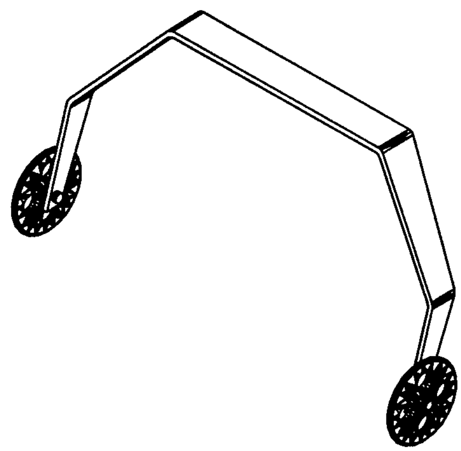


Note:
1. Vertical and horizontal stabilizers, rudder, and elevator are made of 1/4" balsa wood.

Miami University

Tail Components	Team: AIAA Competition	
	Date: 3-3-2002	
	Scale: NTS	
	SH. 4 OF 5	





Note:
1. Motor mount was bored out of one stock of 6061 aluminum.

Note:
1. Landing gear is made out of 6061 aluminum.
2. Wheels are made out of plastic with rubber walls.

Miami University	
Landing Gear and Motor Mount Components	Team: AIAA Competition
	Date: 3-3-2002
	Scale: NTS
	SH. 5 OF 5

5.3. Performance Analysis

To ensure ourselves that the plane will fly at our expected expectations, the team needed to do analysis that involved in flight situations. These situations will test the stability of our plane and make sure that it reacts the way we have expected it to. During the course of flight, the plane will undergo many different pressures and forces. To evaluate our plane we chose these forces in order to do wing and power loading tests.

Figure 5.6 shows a system efficiency graph. It simply shows where our power is going. If the motor had a gearbox, there would be an additional slice for this loss. The graph in Figure 5.7 shows how thrust and drag vary with speed. This graph was produced by *ElectriCalc*. Thrust decreases with speed. It drops off dramatically above pitch speed, 68 MPH. The point at which thrust and drag are equal determines top speed of 60 MPH. The right half of the drag curve is due primarily to parasitic drag, which increases as the square of the speed. The two dashed lines are reference curves. They represent props with P=D and P=D/2, where P stands for pitch and D stands for diameter, with D chosen to draw the same current as our chosen prop. The prop with the highest apparent static thrust has the poorest performance at normal flying speeds.

According to *ElectriCalc*, at 100% throttle, the system efficiency is 63% and the motor efficiency is 75%. The aircraft will be able to produce 101 oz of thrust and 20 oz of drag. This requires 33.6 Amps of current draw. This model has a maximum climb rate of 525 ft/min at a pitch angle of 10°, a maximum speed of 60 mph, and a stall speed of 28 mph. This information can be seen in Figure 5.8, which shows *ElectriCalc's* main screen with our aircraft data entered.

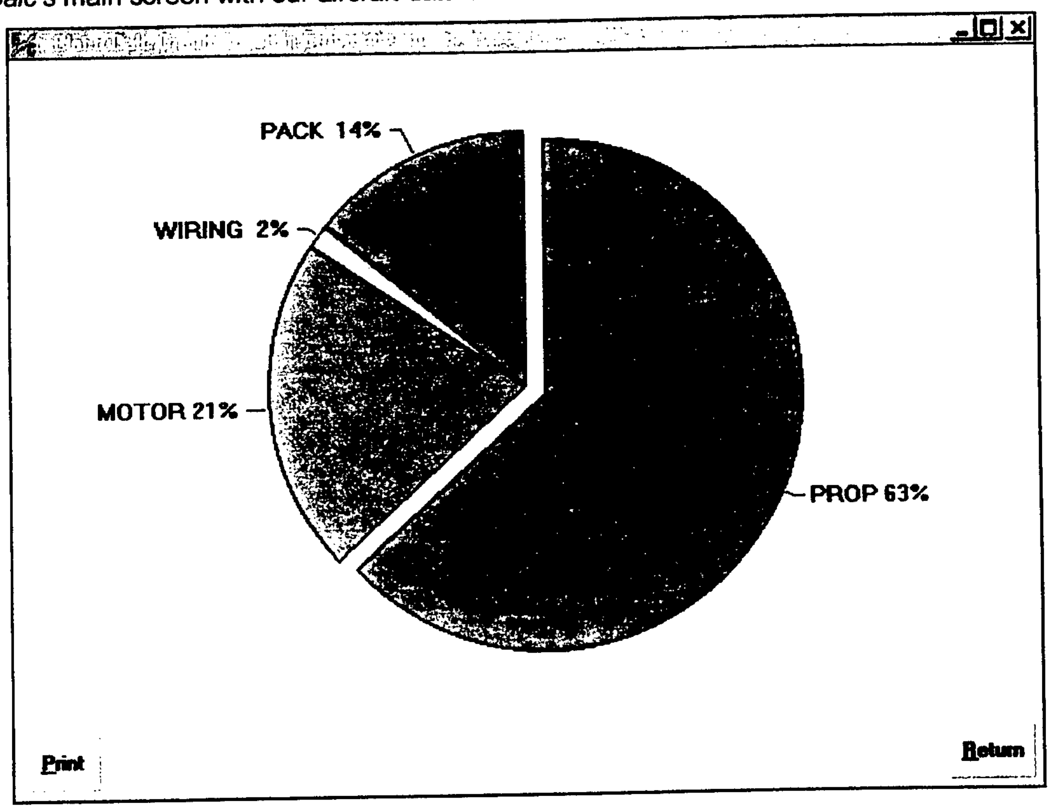


Figure 5.6. System efficiency

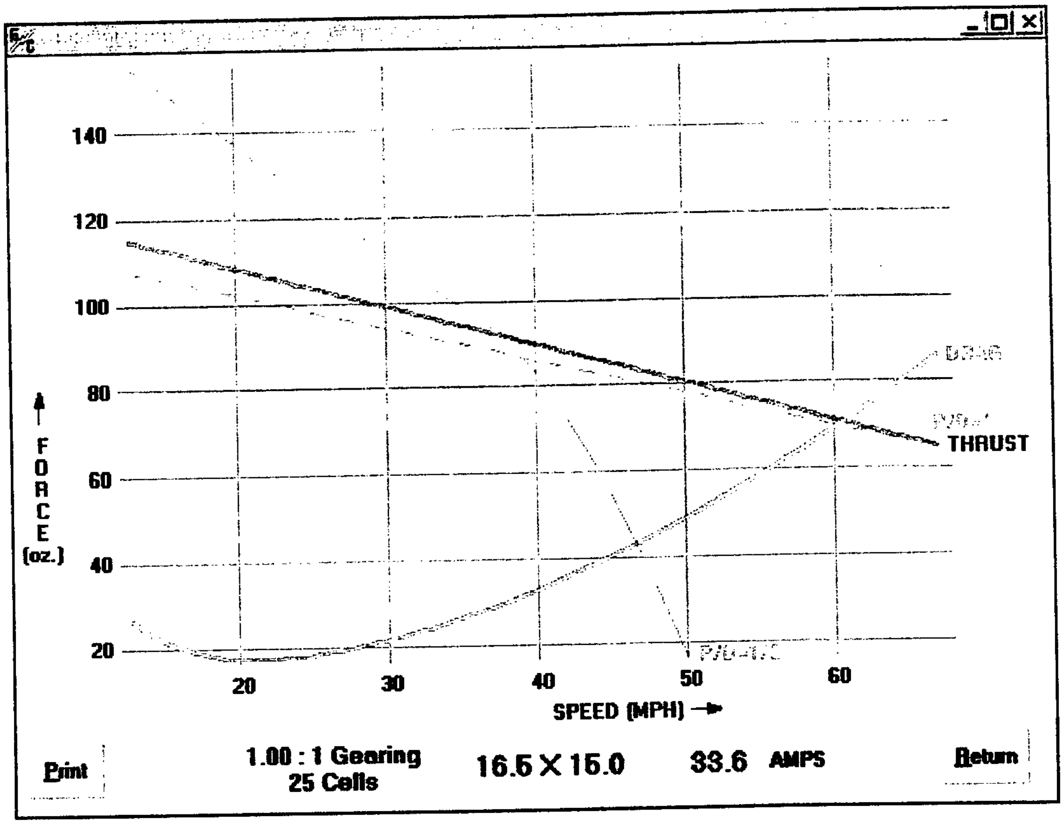


Figure 5.7. Thrust and drag vs. speed

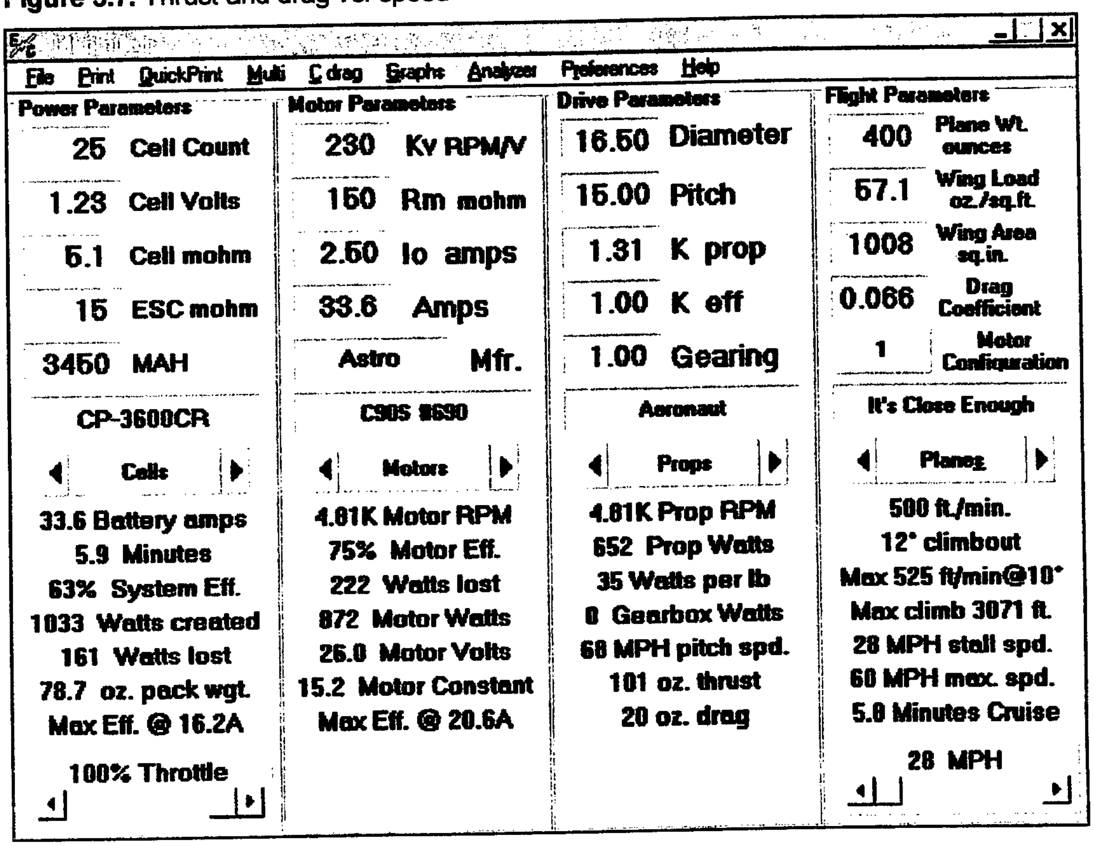


Figure 5.8. ElectriCalc main screen used in component selection and performance predictions.

6. Manufacturing Plan

To maximize our team's effectiveness, a manufacturing plan and schedule was created. This not only aided the team in selection of manufacturing processes, but also kept the team on schedule by comparing actual progress with planned progress. The manufacturing schedule can be seen in Figure 6.1.

6.1. Figures of Merit

While a separate calculation of figures of merit was not created for analysis of manufacturing processes, there were criteria taken into account that directly affect the manufacturing process. We did however use the same material matrix to help decide what material to use in what parts. This helped make decisions involving certain supports. The team felt that taking the following factors into account throughout the manufacturing process would result in the desired outcomes.

6.1.1. Manufacturability

The manufacturability of the components and final design is something that has been considered since the beginning of the team's progress in this competition. In all figures of merit calculations manufacturability was always taken into account. If the team is unable to construct a design, or if a manufacturing method will take too much time, then it was not pursued. The team did not want to take the chance of producing something unreliable or even dangerous; therefore, the final methods used were ones that the team felt comfortable with.

6.1.2. Cost

A large factor in the selections on designs and manufacturing was cost. The team was faced with a tight budget and therefore assigned a team member to create a budget, track funds, and solicit funding from the university. Some rationale due to budget constraints is explained here.

The budget right at the moment is a work in progress. The member in charge of the team's finances has taken some figures from past DBF projects and tried to project an expected cost for this year. It is our hope to use many parts from past planes to help us lower this year's cost. The current estimate for cost is approximately $2,800.

To help reduce the overall cost, which is split between the Aeronautics and Engineering Departments, the team has applied for a grant through the university. The first grant the team applied for was the Undergraduate Research Grant, which is awarded by the Undergraduate Research Committee of the University Faculty Senate. The application for this grant has already been submitted, and we were fortunate enough to receive this grant for $400. Another grant that the team has yet to apply for is the Student Initiative Fund, which is given by the Associated Student Government. If the team receives this amount, the total money brought into the project would be $900.00. By adding this income to our budget, it is our hope that the total cost to be divided between the departments will be approximately $1,900. On the sample design budget in Figure 6.2, one can see how important and massive an impact using the parts from previous planes can mean for our budget.

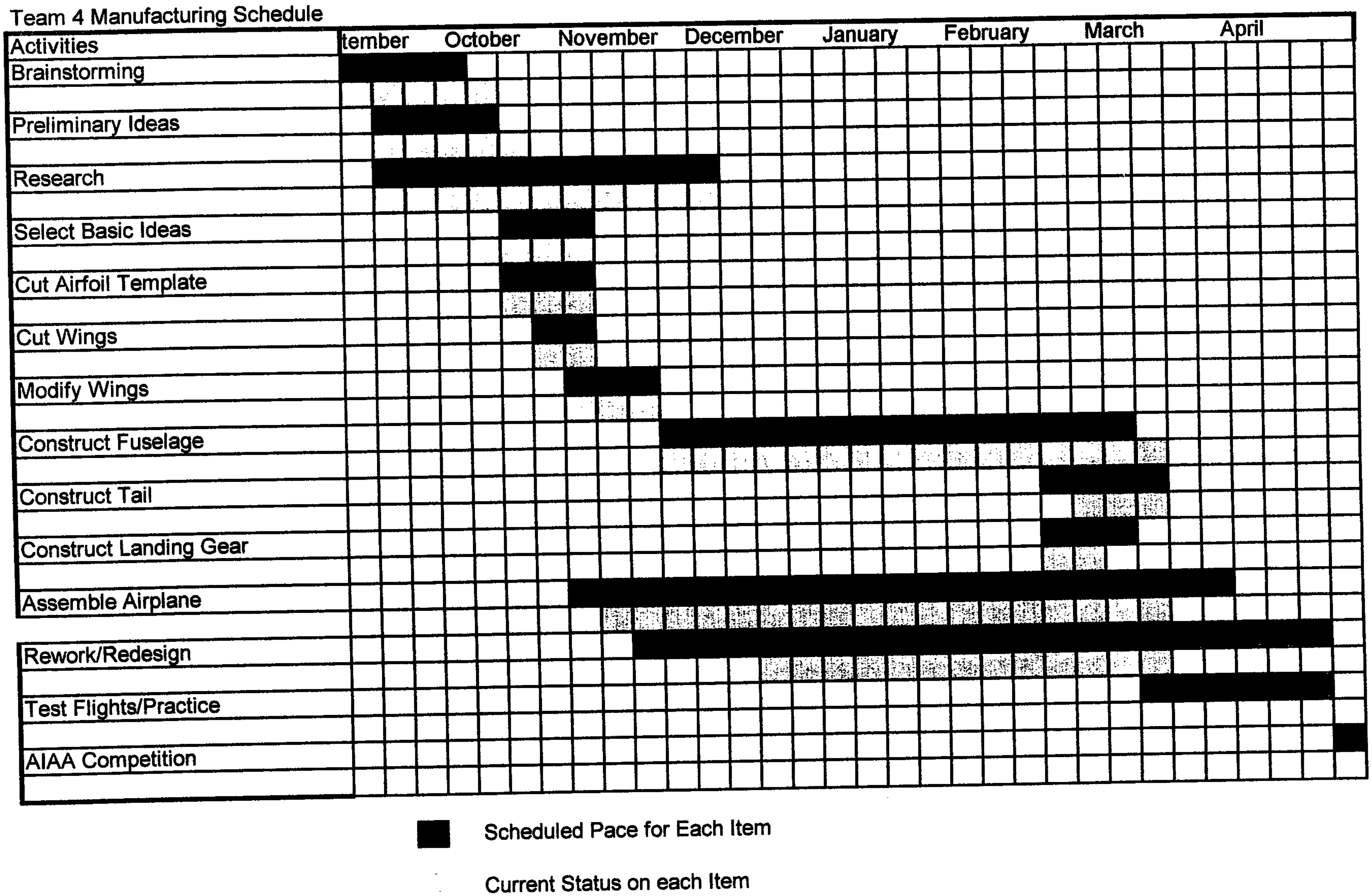


Figure 6.1. Manufacturing plan showing the team's goals for completion of main sections related to the manufacturing process.

AIAA Airplane Design Budget - (2001-2002)

Fixed Costs:			**Potential Income:**		
Competition Costs:			Student Initiative Fund	$500.00	
Transportation	$740.00				
Hotel	$484.00		**Actual Income:**		
Meals	$242.00				
			Research Grant	$400.00	#
Powerplant:					
Controller	$0.00	*	**Total Income:**	$900.00	
Motor	$0.00	*			
Battery Pack	$35.00	*			
Propeller	$150.00				
Fuselage:					
Styrofoam (white)	$0.00				
Balsa/Plywood	$165.46	#			
Miscellaneous	$37.37	#			
Wings:					
Carbon fiber strips and tubes	$180.00				
Styrofoam (blue)	$24.00	#			
Empennage:					
Balsa wood and glue	$110.00				
Landing Gear:	$0.00	*			
Wheels (rubber,bearings)	$0.00	*			
Other:					
Last year's misc.	$463.64				
Softballs	$130.00	#			
Servos	$0.00	*			
Monokote	$40.00				
Wiring	$0.00	*			
Hose clamps	$10.00				
Total Expense:	$2,811.47				
Potential Income:	$900.00				
			* denotes used items		
Total Cost to Departments:	$1,911.47		# denotes final number		

Figure 6.2. Budget used throughout design and manufacturing process.

6.2. Manufacturing Processes

Many processes were investigated when determining how to manufacture each component of the airplane. While many dimensions and other exact information is located in the detail design section, the final manufacturing process used for each component is given in the following sections.

6.2.1. General

In all of our manufacturing, various methods were looked into, but in the end similar methods were used in the manufacturing of all major components. Due to the fragile nature of balsa, birch, and foam, larger saws could not be used because this would cause the material to tear. It was because of this that all foam and balsa was cut using an exacto knife and straight edge.

All general components will then be connected using bolts and/or epoxy. Once construction is complete, the entire plane will then be covered with monokote.

6.2.2. Fuselage

The fuselage was constructed out of birch wood and blue foam. First, using a knife and straight edge, the wood and foam was cut to exact specifications. The center ¼" x ½" birch wood was then glued to a base sheet of 1/16" birch wood using 12 minute epoxy. Next, base sheets of ¼" x 4.14" blue foam and birch wood were glued together to form a composite with the base sheet of ¼" birch wood. Then, the vertical composite of the fuselage was glued together and later this piece was glued around the ¼" x ½" birch wood. At this point, the basic T shape of our fuselage was complete.

Another part added to the fuselage was formers located in the inner compartment. Gluing an 18" x 18" blue foam and birch wood composite together created these. Templates were made for the formers and cut out of cardboard. Next, eight boxes were cut from the 18" x 18" composite using a band saw. On these boxes, the shape of the formers was traced and then created using a band saw and a sander. The former composites were then glued to the T-frame in their specified locations. Lastly, lightning holes were cut in the T-frame with a 2 1/8" hole saw to lessen the overall weight of the fuselage.

6.2.3. Wing

Constructing the wing involves many different steps. The materials used include polystyrene, a carbon fiber strip, aluminum, and balsa wood. The wings were produced by cutting into sheets of aluminum and producing a smaller piece that the airfoil could be cut out of. Using a drill press, two smaller sheets that had been cut were drilled into. This produced three holes that would allow us to connect the two sheets with screws and nuts. Next, we were able to go back to the band saw with an example of the airfoil taped to our sheets of aluminum. We then cut the airfoil our, following the paper as a guide. The two sheets, when separated, produced two replica airfoils. We then used the sander to clear any burrs of the templates so that the cutting of foam would not be interfered with by them.

Next, these templates were placed on opposite ends of a block of foam. The templates had lines written around the outside, measuring one inch apart from the next. Each line was given a number. We then used a hot wire to cut around the template. This wire was electrified, to produce the heat and was held by two people on opposite sides. The two people communicated where their side of the wire's

position was by the lines on the templates. This kept the wire stable and stopped either person from rushing ahead. This imbalance would jeopardize the quality of the cut. As this wire was guided along the airfoil template it melted away the unneeded material. In the end, another piece of polystyrene was then glued between the two wings to allow for a place to connect the fuselage and landing gear.

Next, the entire wing was sectioned vertically ½" back from the leading edge (at the ¼" cord) and 2 ½" forward from the trailing edge using a table saw. A ½" piece of balsa wood was glued on to create the leading edge. At the trailing edge, ½" of balsa wood was glued on and at quarter cord, a carbon fiber strip was sandwiched between the foam to create the wing spar. We used balsa wood because in our material matrix, it proved to be much stronger in cases of density and volume. Finally, the entire wing was sheeted with 1/32" balsa wood and the leading and trailing edges were sanded to the shape of the airfoil using a sanding bar.

6.2.4. Empennage

The tail section was constructed solely of balsa wood and a carbon fiber tube. The carbon tube would grant us the stability needed in an empennage. Grooves were cut into a 1 1/8" carbon fiber tube using a milling machine to allow for the horizontal and vertical stabilizers. Horizontal and vertical stabilizers were cut from ¼" balsa wood using a knife and straight edge. These were then glued into the grooves cut into the carbon tubing using epoxy.

6.2.5. Controls

All control surfaces were cut out of ¼" balsa wood using a knife and straight edge. These will then be hinged to the back of the wings and tail surfaces. Holes will then be cut to allow for the servos and wiring will be installed last.

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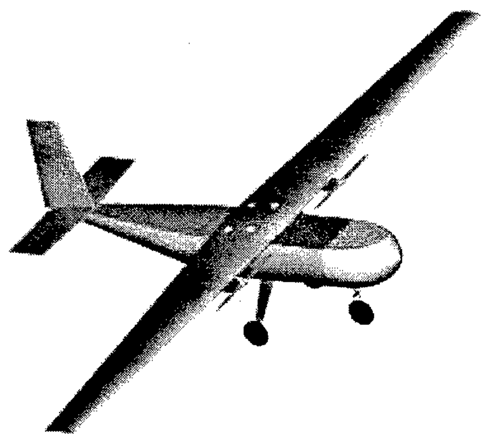
2001/2002 AIAA Foundation Electric Airplane

"Butter Duck"



Design Report

University of Central Florida

March 12, 2002

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Figure 6.3 **Executive Summary**

For this years' Cessna/ONR Student Design Build Fly Competition, a team of students at the University of Central Florida will be entering their number one aircraft "Butter Duck". The competition involves designing, building, and flying a remote controlled electric power, propeller driven, unmanned aircraft. The designed aircraft must be able to complete its mission within a period of ten minutes or less and must be able to complete at least three of these mission tasks. The mission involves three different sorties in which the aircraft must carry a payload of 10-24 softballs during the second sortie. The performance of the aircraft for this competition is basically a factor of speed and volume since every aircraft entering the competition will be judged only on how fast it can complete each of the three missions and what amount of softballs it can carry. The competition itself will be judged by the performance of the aircraft, a written report resulting from the design process, and the rated aircraft cost. The report itself due to involvement with senior design has been broken down into a conceptual phase, preliminary design phase, and detail design phase.

Figure 6.3 **Conceptual Design**

The major areas for the aircraft are the wing, fuselage, empennage, landing gear, and power system. The design was broken down into three sections. Each section consists of a manager in order to provide guidance to the team members. The purpose of each team member is to study different configurations within their assignments and to come up with the best configurations for our needs by applying guided iteration methodology to the engineering conceptual design phase. Once several configurations are decided, they will be combined into various design configurations for the whole aircraft in order to come up with the best design suitable for the competition. The method for evaluating the different designs was dependent on the total score formula. According to past years designs and strategies, we decide to rearrange the total score formula to make it a function of volume and speed. The results from the conceptual stage of the project yielded 20 different configurations. From these 20 configurations, we eliminated ten due to manufacturing difficulty due to the inexperience of the team in building. The final decision on which configurations to use came down to budget constraints. One of the final configuration consisted of a single low-winged aircraft. The wing consisted of NACA 4412 airfoil with a wingspan of 10 feet. The fuselage would carry a total 10 softballs with 5 in each of two columns. The landing gear configuration was a tricycle landing gear. A single boom was considered to make the empennage. Conceptually the aircraft was that of a single motor configuration using an Astroflight 661S with a 3.1:1 gear ratio using a large 22x12 propeller with up to 36 Sanyo 2400mah cells. Three other configurations were also analyzed in further detail to aid in optimum performance for the aircraft.

1.1.1 Design Alternatives

For the design alternatives we looked at each of the individual components of the aircraft and their possible configurations. The alternatives for the several components are: beginning with the wing, the type of airfoil, wing planform, wing configuration, number of wings, and placement of the wing(s). For the fuselage, its volume and shape which depends on the amount of softballs, and internal structure. Several different configurations for the empennage were also considered such as whether to use a single boom or double boom configuration and number of vertical stabilizers. We considered different types of landing gear configurations and materials. Design alternatives for the motor selection consisted of all the blushless cobalt geared sport motors from Astroflight along with individual specification data. Propeller selection consisted of several companies with readily assessable size and supply.

1.1.2 Design Tools

The design tools used for the project consisted of several Excel spreadsheets to determine the best possible Rated Aircraft Score and Flight Score with each aircraft configuration. Pugh's Method and the Dominic Method we used to decide the best aircraft components. Also, a commercially available Internet program called MotoCalc aided with analysis for the power plant by inputting variables for each motor, battery, speed controller, propeller, and gearbox to output data that would initially be used for motor selection. Usage of MotoCalc was limited to assisting in basic analysis of the power plant. Mathcad was utilized later to facilitate in power/flight parameters and aircraft performance data.

1.2 Preliminary Design

In the preliminary design phase the final three design configurations were studied further in detail in order to come up with one final configuration. Further details in payload-loading methods and power system we were studied and scrutinized. Wing area requirements were investigated in order to come up with the minimum area required for our design. Materials were evaluated for construction of the wing, fuselage, and empennage, and landing gear. Our results yielded a smaller wingspan, which would decrease our RAC and increase flight speed. We would carry a total of a dozen softballs, of which ten would be carried in the fuselage and two would be placed in the tail. The softballs would be loaded thru the nose cone by hinging the nose to the top of the fuselage. The empennage would consist of a single tail boom with one vertical tail surface and one horizontal tail surface. The preliminary design results for the power plant resulted in the reduction of motor size, an additional motor, the location of mounting, number and size of battery cells. The new motor selection was reduced to, two Astroflight 640G motors with a 1.63:1 gear ratio using smaller 13x8 propellers with a range of 18-21 cells per motor.

1.2.1 Design Alternatives

The alternatives for each component of our aircraft derived from our three configurations resulting from the conceptual design phase were many. The materials for the wing construction ranged from using white foam with at least one spar for structural support to using blue foam with two spars for support of the twin motors, which would be placed on the wing. Alternative methods of unloading payload were investigated. In one of these designs, the tail would be hinged to allow access to the payload from the aft section. Another option would be to load the softballs from the top into an egg crate type of holding device. This would require a low wing configuration. A third alternative would be to hinge the nose and load the softballs from the front. One of the tail section alternatives consisted of a single tail boom straight tapered from the fuselage with a large vertical stabilizer due to the heavy wind conditions that would be encountered in Wichita. Another alternative consisted of using a double vertical stabilizer but his would increase our RAC. New configurations in the aircraft body design presented new alternatives to motor placement as mentioned above. Alternative battery configurations were analyzed for the weight limitation of five pounds. The basic design problem was the higher the capacitance the heavier the cell, meaning fewer cells for a higher amperage.

1.2.2 Design Tools

Pro/Engineer was used to make accurate virtual preliminary models of the aircraft with the different configurations in order to see if the different design configurations could be integrated. Mass properties were also explored by using Pro/Engineer. Further use of excel sheets were used to investigate final configuration RAC and flight characteristics. Results gained for the new power plant configuration were based on Astroflight data and MotoCalc analysis.

1.3 Detail Design

During the detail design phase the aircraft was refined and optimized to allow for the best results in structural support, endurance, and stability and control according to the engineering requirements. Aerodynamic surfaces such as the wing, elevator and rudder were optimized with respect to placement. The fuselage was strengthened in key locations such as where the wing and landing gear are attached to the fuselage. Structural analysis was done on the landing gear to ensure rigidity usefulness. The power plant component placement was also optimized. The detail design phase resulted in an aircraft consisting of a wing area of 4.571 squared feet and a span of 8 feet constructed out of blue foam with two spars for motor attachment and with carbon fiber coating. The fuselage is 1.593 feet in length and able to carry ten softballs. The empennage consists of a 1.75 feet long single boom.

1.3.1 Design Alternatives

Alternatives for wing attachment and landing gear reinforcement were analyzed for added safety and structural integrity. Placements of internal components were reviewed to ensure cg and control requirements. Angles and heights were assessed to ensure clearance of propellers and servo motors. Power plant component placement was also analyzed in detail. The combination of motor, propeller, and battery configuration was evaluated to finalize the design was sufficient to assure the aircraft would be able to sustain a running/flight time under ten minutes.

1.3.2 Design tools

Detail drafting designs were performed on Pro/Engineer, as well as final mass and cg calculations. IDEAS was used to perform compile structural analysis information on the deflection, load and safety factor of the landing gear. MotoCalc was utilized once again in motor-propeller-battery calculation data.

2.0 Management Summary

The assembly of the team members was carefully reviewed t ensure the success of the project overall. The team was broken down into three main sections, aerodynamic surfaces, fuselage and landing gear and power system. Each section consists of a manager, which is responsible in guiding the rest of the team members and dividing the work within their assignments. The section managers, who are the driving force behind the design phase of the project, are made up of the senior class members. The design team consists of three seniors and eight underclassmen of which one is a computer-engineering student. The rest of the team consists of aerospace-engineering students.

2.1 Architecture of the Design Team

The organization of the team was arranged according to the team member strengths. For the entire design process and the manufacturing process, the team was broken down into the sections shown in figure 2.1. Each group was responsible for brainstorming, analyzing, and manufacturing of their particular assignment in the project. Figure 2.2 shows the amount of participation by each of the team members and more specific personnel assignments.

2.2 Project Schedule

The team met with the faculty advisors for the project, Dr. J. McBrayer and Dr. R. Johnson, on every Tuesday at 10:30 AM to discuss the findings and the progress within their assignments during the design phase and to discuss any problems that the team may have encountered. The team would also meet without the faculty advisors present on Tuesdays and Thursdays from 10 AM – 12:20 PM and on

Saturdays from 11 AM until 4 PM. During the manufacturing phase the meetings with the team advisor were held on Thursdays from 10:30 AM until 11 AM. The separate groups from the team then concentrated the manufacturing efforts on Tuesdays and Thursdays from 9 AM until 12:20 PM and from 6:30 PM until 10 PM during the same days. The team as a whole then met on Saturdays from 1 PM until 6 PM to discuss and make sure the correct assembly of parts and the aircraft as a whole.

In order to meet the objectives and the major and minor milestones with in the required time, the team set up a Gantt chart, which is shown in figure 2.3. The Gantt chart shows the planned and actual timing of major elements of the design process in the conceptual design phase, preliminary design phase, the detailed design phase, including the report preparation periods as well as the progress of the manufacturing process and testing of the aircraft.

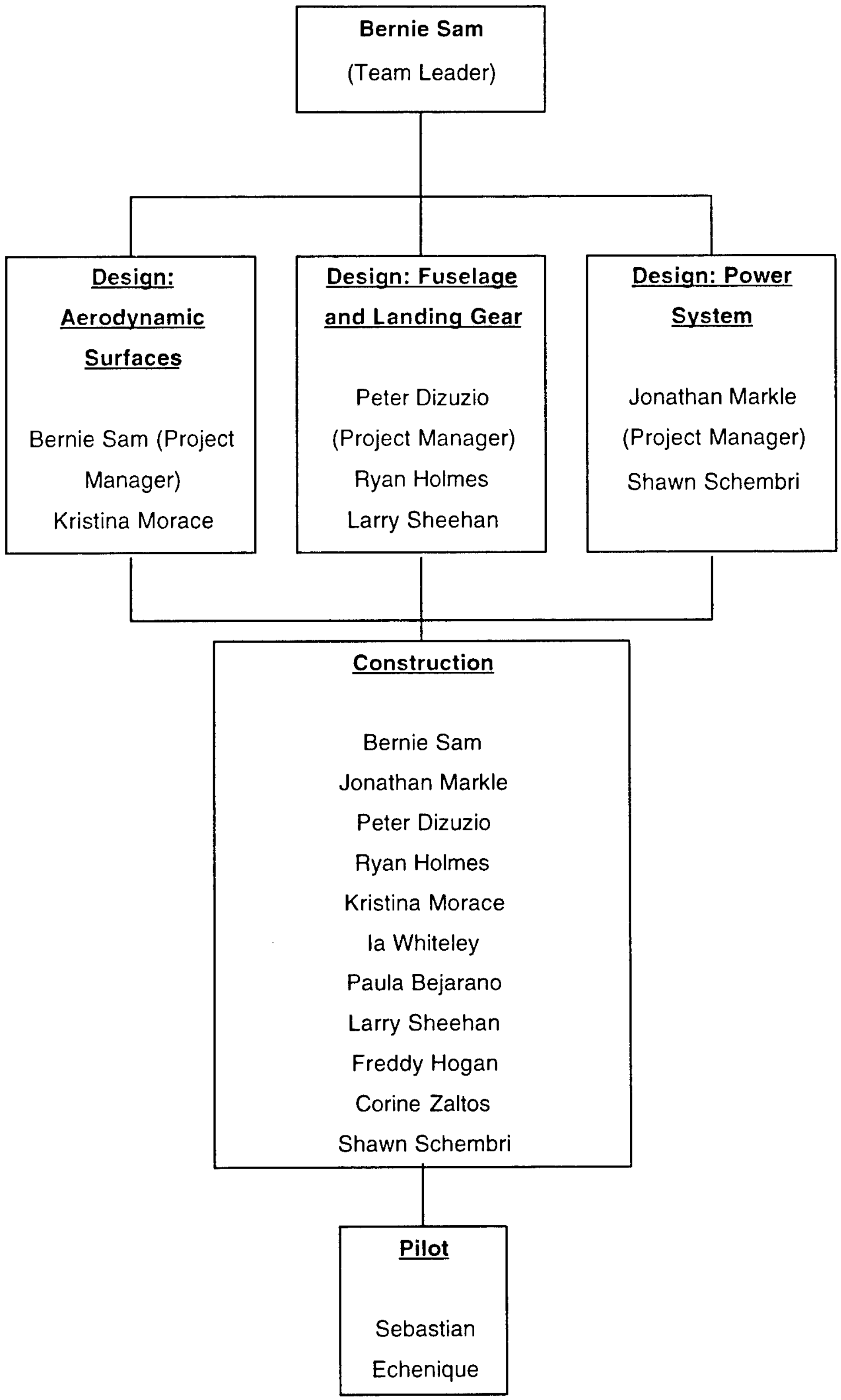


Figure 2.1 Task Breakdown

Design Task	Team Member(s)
Design Requirements	
Design Requirements Written Report	1,2,3,4,5
Design Requirements Oral Report	1,2,3
Conceptual Design Phase	
Component Analysis	1,2,3
Analytical Analysis	1,2,3
Conceptual Design Written Report	1,2,3,4,5
Conceptual Design Oral Report	1,2,3
Preliminary Design Phase	
Detailed Component Analysis	1,2,3
Final Component Analysis	1,2,3
Final Configuration Selection	1,2,3
Flight Performance Calculations	2
Preliminary Final Configuration	1,2,3
Preliminary Design Oral Report	1,2,3
Preliminary Design Written Report	1,2,3,4,5
Detail Design Phase	
Stability and Control Refinement	1,2
Detailed Drawings and Dimensions	1
Power Plant Refinement	3
FEM Analysis of Individual Components	2,3
Detailed Design Oral Report	1,2,3
Detailed Design Written Report	1,2,3,4,5
Manufacturing	
Manufacturing Process Selection	1,2,3,4,5,6,7,8,9,10,11
Detail Manufacturing Plan	1,2,3,4,5,6,7,8,9,10,11
Construction	1,2,3,4,5,6,7,8,9,10,11
Testing	
Flight Testing and Evaluation	1,2,3,4,5,6,7,8,9,10,11
Final Adjustments	1,2,3,4,5,6,7,8,9,10,11
Written Report for Competition	1,2,3,4,5

Personnel Assignment Breakdown:

1	Bernie Sam
2	Peter Dizuzio
3	Jonathan Markle
4	Kristina Morace
5	Ia Whiteley
6	Ryan Holmes
7	Larry Sheehan
8	Shawn Schembri
9	Paula Bejarano
10	Freddy Hogan
11	Corine Zaltos

Figure 2.2 Participation Breakdown

Task Name	Start	Finish
Design Requirements	Wed 9/12/01	Fri 9/21/01
Design Requirements Written Report	Wed 9/12/01	Fri 9/21/01
Design Requirements Written Report (Actual)	Wed 9/12/01	Fri 9/21/01
Design Requirements Oral Report	Wed 9/12/01	Tue 9/18/01
Design Requirements Oral Report (Actual)	Fri 9/14/01	Tue 9/18/01
Conceptual Design Phase	Fri 9/21/01	Thu 10/25/01
Component Analysis	Fri 9/21/01	Fri 10/5/01
Component analysis (Actual)	Fri 9/21/01	Wed 10/10/01
Analitycal Analysis	Mon 10/8/01	Thu 10/18/01
Analitycal Analysis (Actual)	Fri 10/12/01	Sat 10/20/01
Conceptual Design Oral Report	Fri 10/19/01	Tue 10/23/01
Conceptual Design Oral Report (Actual)	Sat 10/20/01	Tue 10/23/01
Conceptual Design Written Report	Mon 9/24/01	Fri 10/26/01
Conceptual Design Written Report (Actual)	Tue 9/25/01	Fri 10/26/01
Preliminary Design Phase	Sat 10/27/01	Fri 11/23/01
Submission of Competition Entry Form	Tue 10/30/01	Tue 10/30/01
Detailed Component Analysis	Sat 10/27/01	Sat 11/3/01
Detailed Component Analysis (Actual)	Sat 10/27/01	Wed 11/7/01
Final Component Selection	Mon 11/5/01	Mon 11/12/01
Final Component Selection (Actual)	Thu 11/8/01	Mon 11/12/01
Final Configuration Selection	Mon 11/12/01	Mon 11/19/01
Final Configuration Selection (Actual)	Tue 11/13/01	Tue 11/20/01
Flight Performance Calculations	Mon 11/19/01	Wed 11/21/01
Flight Performance Calculations (Actual)	Mon 10/22/01	Thu 10/25/01
Preliminary Final Configuration	Sat 10/27/01	Fri 11/23/01
Preliminary Fianl configuration (Actual)	Sat 10/27/01	Mon 11/26/01
Preliminary Design Oral Report	Sat 10/27/01	Thu 11/29/01
Preliminary Design Oral Report (Actual)	Thu 11/29/01	Thu 11/29/01
Preliminary Design Written Report	Sat 10/27/01	Fri 11/30/01
Preliminary Design Written Report (Actual)	Sat 10/27/01	Fri 11/30/01

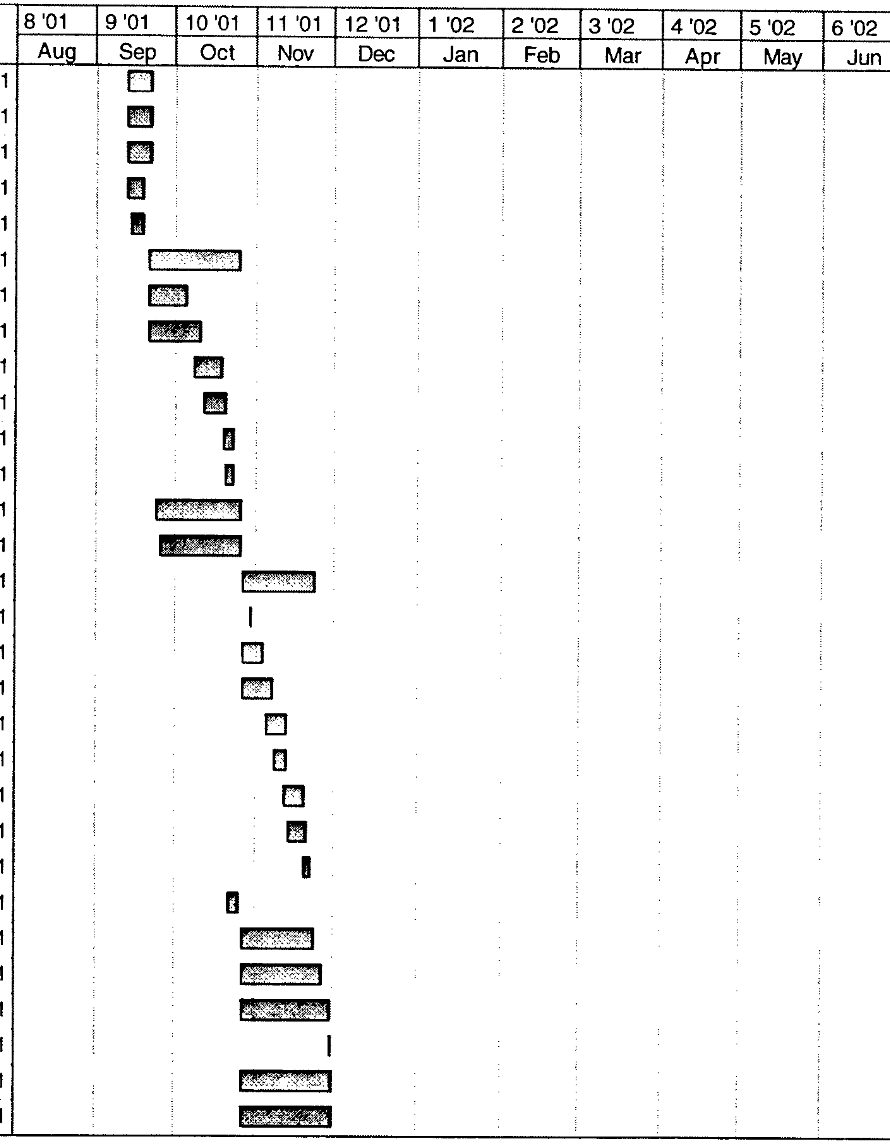


Figure 2.3 Gantt Chart

Task Name	Start	Finish
Detail Design Phase	Mon 11/26/01	Fri 3/1/02
Stability and Control Refinement	Mon 11/26/01	Mon 12/10/01
Stability and Control Refinement (Actual)	Fri 11/30/01	Sat 12/15/01
Detailed Drawings and Dimensions	Mon 11/26/01	Tue 2/26/02
Detailed Drawings and Dimensions (Actual)	Mon 1/7/02	Tue 2/26/02
FEM Analysis of Individual Components	Wed 2/20/02	Wed 2/27/02
FEM Analysis of Individual Components (Actual)	Wed 2/20/02	Wed 2/27/02
Datailed Design Oral Report	Mon 11/26/01	Thu 2/28/02
Detail Design Oral Report (Actual)	Wed 1/16/02	Thu 2/28/02
Detailed Design Written Report	Mon 11/26/01	Fri 3/1/02
Detailed Design Written Report (Actual)	Fri 12/28/01	Fri 3/1/02
Manufacturing	Mon 1/14/02	Fri 3/1/02
Manufacturing (Actual)	Thu 1/17/02	Sat 3/2/02
Flight Testing and Evaluation	Fri 3/1/02	Tue 3/5/02
Flight Testing and Evaluation (Actual)	Fri 3/1/02	Sun 3/3/02
Final Adjustments	Wed 3/6/02	Thu 3/28/02
Final Adjustments (Actual)	Wed 3/6/02	Fri 3/8/02
Written Reports for Competition	Thu 9/20/01	Fri 3/8/02
Logbook	Mon 9/10/01	Fri 4/26/02
Ship Airplane to Pilot	Fri 3/29/02	Fri 3/29/02
Competition	Fri 4/26/02	Sun 4/28/02

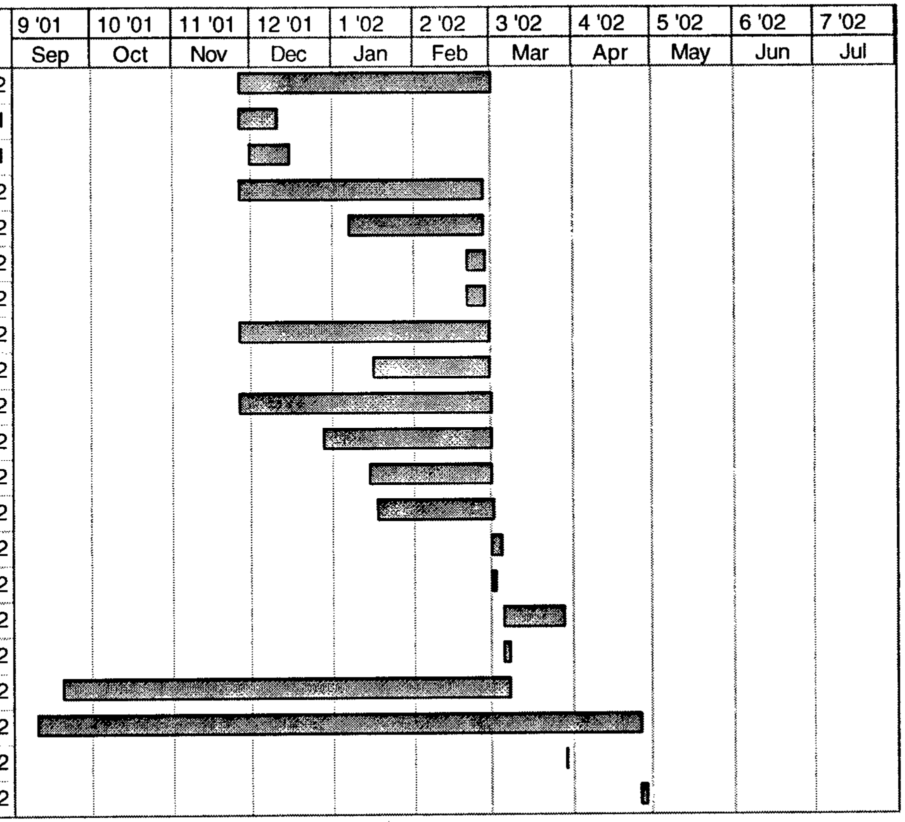


Figure 2.3 Cont. Gantt Chart

3.0 Conceptual Design

For the conceptual design phase many different configurations were evaluated against the given parameters, which would first have to be investigated to achieve a working design that meets all of our design objectives. The main objective is to achieve a lightweight simple design aircraft that is efficient, durable, and fast. While a perfect aircraft is always desired, compromises must be made since every part of the design may not have the best configuration. This is due to the fact that every part of the aircraft affects all other parts of the aircraft.

3.1 Aircraft Component Configurations

Several concepts were studied for each component in order to come up with several final configurations. Each component was separately evaluated with its own figure of merits to achieve an optimum design. The components were also evaluated according to its estimated detail design parameters. The components of the aircraft were broken up as follows: wing, fuselage, empennage, landing gear, and power plants. All these components were evaluated separately to reduce complexity.

3.1.1 Wing Configuration

The airfoil is the heart of the airplane since if affects the cruise speed, takeoff and landing distances, stall speed, handling qualities, and overall aerodynamic efficiency during all phases of flight. Since the two main factors, speed and stability are affected by the airfoil, we have to pay special attention to these factors and determine the best airfoil for the job.

There are a variety of airfoils that may be able to do the job. We looked at airfoils which had data already developed for it. Some of the airfoils families that were researched were the Selig family of airfoils and the NACA 4 digit family of airfoils. The afore mentioned family of airfoils were investigated since past teams had used these types of airfoils in the past for these kinds of competitions. We also looked at the option of purchasing a model airplane kit from which we would use the already fabricated wing. This would eliminate the need to fabricate the wing ourselves, which can prove to be a difficult task. It would also eliminate the imperfections that arise from fabrication when the wing is handcrafted. Some specific airfoils that were considered were the Selig 1210, Selig 1221, Selig 1223, Selig 2027, NACA 2412, NACA 4412, and NACA 4415 because of the amount of data available for these airfoils and because the geometry of most of these airfoils is fairly easy to fabricate.

The geometry of the wing was looked at in the following manner. One begins with the "reference" wing which is the basic wing geometry used to begin the layout. The reference wing is fictitious, and extends through the fuselage to the aircraft centerline. It includes the part of the reference wing that sticks into the fuselage. For the reference wing, the root airfoil is the airfoil of the trapezoidal reference wing at the

centerline of the aircraft, not where the actual wing connects to the fuselage. There are two key sweep angles. The leading-edge sweep and the sweep of the quarter-chord line that is the sweep most related to subsonic flight. The shape of the reference wing is determined by its aspect ratio, wing sweep, taper ratio, twist, wing incidence, dihedral, vertical location, wingspan, and wing area.

The aspect ratio is defined as the span of the wing divided by the chord. For a tapered wing, the aspect ratio is defined as the span squared divided by the area. A long skinny wing has less drag for a given lift than a short fat wing. This is due to the 3-D effects. When a wing is generating lift, it has a reduced pressure on the upper surface and an increased pressure on the lower surface. The air would like to "escape" from the bottom of the wing, moving to the top. Air escaping around the wing tip lowers the pressure difference between the upper and lower surfaces. This reduces lift near the tip. Also, the air flowing around the tip flows in a circular path when seen from the front, and in effect pushes down on the wing. Strongest near the tip, this reduces the effective angle of attack of the wing airfoils. This circular, or "Vortex," flow pattern continues downstream behind the wing. A wing with high aspect ratio has tips farther apart than an equal area wing with a low aspect ratio. Therefore the amount of the wing affected by the tip vortex is less for a high-aspect –ratio wing than for a low-aspect-ratio wing, and the strength of the tip vortex is reduced. Thus, the high-aspect-ratio wing does not experience as much loss of lift and increase of drag due to tip effects as a low-aspect-ratio wing of equal area. The trade off of having a high aspect ratio is the increased weight of the wing. We came up with an aspect ratio of between 7.4 and 7.8 (Aircraft Design: A conceptual Approach, Daniel P. Raymer, page 59).

Wing Sweep is the Angle by which leading edge or quarter-line is rotated with respect to plane normal to flight direction. It is primarily used to reduce the adverse effects of transonic and supersonic flow. Therefore this option was not considered.

The taper ratio, λ, is the ratio between tip chord and centerline root chord. Taper affects the distribution of loft along the span of the wing. Minimum drag due to lift, or "induced" drag, occurs when the lift is distributed in an elliptical fashion as shown below.

For an untwisted and unswept wing, this occurs when the wing planform is shaped as an ellipse. However, an elliptical wing planform is very difficult to build. The option is to taper the wing. When a rectangular wing is tapered, the tip chords become shorter, alleviating the undesired effects of the constant-chord rectangular wing. Our taper ratio was chosen to be around 0.4, which almost completely eliminates the unwanted effects for an unswept wing, and produces a lift distribution very close to the elliptical ideal.

The twist is the angle, by which tip section is rotated with respect to root section. It is used to prevent tip stall and to revise the lift distribution to approximate an ellipse. Values typically range fom zero to five degrees. Due to the high difficulty of manufacturing a wing with a geometric twist, we chose not to consider this option.

The wing incidence angle is the pitch angle of the wing with respect to the fuselage. Wing incidence angle is chosen to minimize drag at some operating condition, usually cruise. The incidence angle is

chosen such that when the wing is at the correct angle of attack for the selected design condition, the fuselage is at the angle of attack for minimum total drag. For a typical, circular straight fuselage, this is often a few degrees nose up, allowing the fuselage to contribute to lift. From wind tunnel test data, the general wing incidence angle is usually about 2 degrees for an untwisted wing.

Wing dihedral is the angle of the wing with respect to the horizontal when seen from the front (see figure below). Positive (tips higher) dihedral tends to roll the aircraft level whenever it is banked. This is caused by a rolling moment generated by a sideslip due to the bank angle. The aircraft "slides" toward the lowered wing, which increases its angle of attack and therefore its lift. Therefore, providing stability for the aircraft. In addition, the position of the wing on the fuselage has an influence on the effective dihedral, with the greatest effect provided by the high wing configuration where the fuselage in sideslip pushes the air over and under itself. If the wing is high-mounted, the air being pushed over the top of the fuselage pushes up on the forward wing, providing an increased dihedral effect. The reverse is true for a low-mounted wing.

Figure 3.1 The chart below is used for dihedral guidelines.

	Wing position		
	Low	Mid	High
Unswept	5 to 7	2 to 4	0 to 2
Subsonic swept wing	3 to 7	-2 to 2	-5 to –2
Supersonic swept wing	0 to 5	-5 to 0	-5 to 0

The wing vertical location with respect to the fuselage is generally set by the environment in which the aircraft will operate. One major benefit of a high wing is that it will prevent the wing tips to hit the ground during take off. For an aircraft designed with a short takeoff and landing requirement, a high wing offers several advantages. The high position allows room for very large flaps needed for a high lift coefficient during takeoff. A disadvantage to having a high wing configuration is that the fuselage has to be

strengthened for the landing gear attachment, which increases the weight of the aircraft. The mid wing position of the wing will not be considered since it is very difficult to manufacture and would not allow for our payload to be fitted very easily. The low wing configuration offers an advantage when it comes to attaching the landing gear. The fuselage would not have to be strengthened with this configuration.

The wing area, which is the area of the wing projected on the ground plane, and the wingspan, which is the distance from tip to tip, will be determined by the wing loading (Weight/Wing Area) and thrust-to-weight ratio. These are the two most important parameters affecting aircraft performance. The approximate weight of the airplane is estimated to be 15 pounds. These calculations will be shown in our preliminary design.

3.1.2 Fuselage Configuration

The fuselage is one of the most important parts of the aircraft. It must be able to structurally support the assembly of the wing, tail, landing gear, and payload. It will also contain the batteries, servos and other components. The fuselage must also be aerodynamically efficient, easily manufactured and above all lightweight. Although lightweight, the fuselage has to be large enough to accommodate the 10 to 24 softballs and allow for easy loading and unloading of the payload. Three basic fuselage shapes considered were: a lifting body fuselage, a circular cross sectional fuselage, and a rectangular cross sectional fuselage.

The first shape considered was a lifting body fuselage, which is shaped like an airfoil. An airfoil is the most aerodynamically efficient shape available and would even produce some lift. The only disadvantage would be the cost and added complexity of the design. The circular cross section would the most volume efficient design since the payload is circular. Finally, the rectangular cross section would relatively easy to fabricate and would allow larger planar area for the landing gear to attach to. The only disadvantage would be a higher drag than the other sections considered. A hybrid between the circular and rectangular cross section fuselage was chosen for its simplicity and overall performance.

The total length and width of fuselage will be determined on the number of softballs that can be carried and how they will be configured into the fuselage. The amount of payload carried will ultimately decide the most efficient fuselage shape that should be used. Another important factor concerning the fuselage dimensions are that there are connected with rated aircraft cost and an inefficient design could ultimately lower the final overall score.

3.1.3 Empennage Configuration

The tail is an integral part of the aircraft. Tails provide trim, stability, and control. Trim is the counter-acting lift force that balances out the moments caused by the wing and other components of the aircraft. The tail also adds stability to the aircraft by restoring any changes in pitch or yaw. Although it is possible

to design a stable aircraft without a tail, in this simple design it would not be feasible. A flying wing is very unstable and hard to control. Finally, the other major function of the tail is control. The horizontal and vertical tails allow the aircraft to maneuver. The control power depends upon the size and type of movable surface as well as the overall size of the tail itself.

There are many aft tail variations that can be used in designing the tail. The three types that are most beneficial for our design needs are the conventional, T-tail, and V-tail arrangements. The other tail arrangements were looked at and were immediately discarded due to their complexity and redundancy.

The conventional tail is the most widely used arrangement, in that 70% of all aircraft have such a tail arrangement. A conventional tail consists of a horizontal and vertical stabilizer located aft of the center of gravity, with the vertical stabilizer located above the horizontal stabilizer. This tail configuration provides adequate stability and control and is the lightest weight.

The T-tail is also widely used. This arrangement lifts the horizontal tail from the wing wake and prop wash, which makes it smaller and more efficient. Also the T-tail reduces the size of the vertical tail. The disadvantage of this arrangement is that the vertical tail must be strengthened to support the horizontal tail, which makes tail inherently heavier.

The V-Tail configuration uses two surfaces aligned in a V-shape to provide stability in pitch and yaw. A V-Tail offers reduced interference drag but a complex control-actuation process since the rudder and elevator controls must be blended together. The conventional tail was chosen for further analysis because of its adequate stability and control with a simple and low cost design.

3.1.4 Landing Gear Configuration

Landing gear affects the performance of an airplane on the ground and in the air. An efficient landing gear design will have low drag characteristics and lightweight, without compromising strength and handling. There are many landing gear configurations that could be used in designing the landing gear. Three types that are most beneficial for our design are tail dragger, bicycle, and tricycle landing gears. The other landing gears were discarded due to complexity and cost

The tail dragger landing gear has two main wheels forward and a smaller wheel aft under the tail. This creates an inclination in the plane, which is beneficial if the second payload option with the trap doors is used. Also another advantage of this gear is that it allows for more propeller clearance, which in turn one could use a bigger propeller, which would provide greater thrust. Also another advantage it has a low weight and drag which increases lift in the aircraft. The only disadvantage is that this landing gear is inherently unstable and very hard to land.

The bicycle landing gear has two main wheels one forward and one aft with two outrigger wheels under the wings. The advantage of the bicycle gear is its stability. The only disadvantages is that the aircraft must have a narrow fuselage and wings must have a high aspect ratio, both to allow the wheels to fit under the wings.

Finally the tricycle landing gear has three main gears with one in the front and two aft on the aircraft. It is considered the most common landing gear for small aircraft today. The main advantage of the tricycle landing gear is its stability and ease of landing.

Another consideration is if the landing gear should be fixed or retractable. Traditional solid spring landing gear has been tested to be the most reliable landing gear available. It can be manufactured to be strong, durable and lightweight. However, the drag produced while in flight is considerable. Meanwhile retractable landing gear has no drag while in flight. The problems with retractable landing gear are increased weight, complexity, and reliability. Use of retractable gear would also add another servo, increasing the rated aircraft cost. The solid spring tricycle configuration was chosen for further analysis.

3.1.5 Power Plant Configuration

What the power plant consists of is the motor, propeller, and batteries. The initial power plant's purpose is supply the aircraft with sufficient thrust to takeoff within the allotted length, 200 feet, and fly three sorties, two unloaded and one loaded with a softball cargo. Secondly, supply enough power to fly the set course within the ten-minute flight period.

Battery selection was a principal factor for this competition. The amount of battery cells cannot exceed the five pound weight limit and still provide ample power to provide adequate thrust to fly the given course and sorties. According to previous competition team's findings, the Ni-Cad batteries posed a serious overheating threat. Based on new data, the previous team's batteries overheated due to a lack of completely discharging the batteries after usage. This instigated an imbalanced voltage across the battery packs, which concluded in overheating while in flight. Consistent with the previous team's findings, the aircraft will receive the peak performance through the usage of all five pounds of batteries. Several number of different sized capacitance batteries were all considered in choosing the proper cell to complete the competition within ten minutes.

Conceptually, all of Astroflight cobalt geared sport motors were considered using Astroflight's data for each motor. A single motor configuration was initially thought of as a reasonable configuration to complete the mission and keep down the rated aircraft cost. In addition to the motor and batteries selection, the propeller(s) was accessible by commercial dealers with different arrangements for each motor-propeller-battery selection.

3.2 Analysis of Aircraft Configurations

Several configurations were analyzed for each of the major components in order to optimize each system.

3.2.1 Wing Analysis

In order to decide which airfoil to use for our wing, we put our airfoils through "Visual Foil", which is a program that calculates different parameters for the airfoil. These parameters include lift, drag, and moments. We also gathered data that can be found on the Internet for all of these airfoils. The parameters lift, Drag, and Manufacturing, were then put into a Pugh Diagram to help us choose which airfoils best meets our need. These parameters were considered to be the most significant parameters in choosing our airfoil. For the model kit wings we would have to rely on information from the manufacturers of these kits. Currently, we are still researching this option as well as all the other options.

3.2.2 Fuselage Analysis

Efficient use of a fuselage volume will result in the best design. If the hybrid circular cross section is used it will increase the number of softballs that can be used and limiting the frontal area so to reduce drag. The rectangular top and bottom of the fuselage will allow for placement of the wing and landing gear to a flat area.

The main focus of the initial fuselage design was the optimum number of softballs that could be carried and in which manner they would be arranged in the fuselage. The number of softballs carried is a very important factor since it directly affects the overall score and indirectly affects the rated aircraft cost by changing the volume and mass of the aircraft.

The optimum number of softballs carried calculated was 12, placed two wide and six deep. This configuration resulted in less drag and weight, by decreasing the frontal area of the aircraft, which would increase our flight speed. Also another factor is that only one of the three missions involves flying the aircraft loaded. It seems highly inefficient to have such a large structure occupied only one third of the time. It was determined that carrying a smaller payload would attain the highest overall score, since the plane would be smaller and faster.

3.2.3 Empennage Analysis

The main focus of the empennage design was to get a general geometric sizing of the boom and control surfaces. From the sizing of these components an estimated weight could be calculated. For the initial sizing of the tail boom, the tail moment arm was determined to be 60-65% of the fuselage length. This is normally a good rule of thumb for R/C airplanes. The tail moment arm is needed to calculate the tail size. Also the tail size is related to the wing, since the primary purpose of the tail is to counter the moments produced by the wing.

The effectiveness of the tail generating a moment about the center of gravity is proportional to the lift produced by the tail and to the tail arm. This force due to the tail lift is proportional to the tail area. Thus, the tail effectiveness is proportional to the tail area times the tail moment arm. This is the tail volume coefficient method for initial sizing of the tail. Dividing this coefficient by the wingspan the vertical tail

volume coefficient can be found since the wing yawing moments must be countered. Dividing by the mean chord of the wing leads to the horizontal tail coefficient, since the pitching moments must be countered.

Once these volume coefficients are determined along with the moment arm the surface areas of the vertical and horizontal tails can be determined. These surface areas of the horizontal and vertical tail stabilize the aircraft. The control surfaces on the horizontal and vertical tail must be determined so that aircraft has adequate control. The elevator for the horizontal was determined to be approximately 25-50% of the horizontal of the tail chord. The rudder was determined to be approximately 40-50% of the vertical tail chord. These final sizing of these parameter will ultimately lead to a stable aircraft.

3.2.4 Landing Gear Analysis

The main parameter governing the landing gear design is structural integrity of the landing gear. Also important is the actual dimensions of the landing gear. It must be sized appropriately so that there is enough propeller clearance and wide enough so that the aircraft doesn't overturn when taxied around a sharp corner. Once the weight of the aircraft is determined the loads on the gear can be found and the best material can be chosen.

3.2.5 Power Plant Analysis

To optimize the performance of the battery configuration, many tests were performed using an Internet program called MotoCalc to obtain a grasp of how each of the selected motors would work together with packs of the chosen batteries. Multiple tests were run to gather information on performance. After performance data was gathered the weight of the batteries multiplied by the calculated number of cells were applied to the weight restriction of five pounds and used to enhance the operation of the motor-battery-propeller combinations. The top-rated motor-battery-propeller combinations were then applied to the design in all perspectives. Weight of the motor and batteries played a major role in the decision making process. A single motor configuration was first selected. An Astroflight 661S with a 3.1:1 gear ratio and 36 cell with Sanyo 2400mah batteries was one of the first analyzed propulsive proposals. This proposal would put the battery weight at 2.2 ounces per battery and a total 4.95 lbs for 36 cells. This was a satisfactory battery weight without taking in account for solder and shrink-wrapping weight. Other configurations were analyzed to optimize the performance.

3.3 Figures of Merit

A figure of merit is a way to quantify and compare the benefits and drawbacks of a given design concept. The figures of merit are different for each component of the aircraft as you can see below. Although there are common ones: weight, cost and manufacturing.

Pugh Method for Airfoil								
		Alternatives						
Evaluation Criteria		NACA 2412	NACA 4412	NACA 4415	Selig 1210	Selig 1221	Selig 1223	Selig 2027
Lift		-	s	+	+	+	+	+
Drag		+	s	-	-	-	-	-
Manufacturing		s	s	s	-	-	-	-
	+	0	0	1	1	1	1	1
RESULTS	S	1	3	1	0	0	0	0
	-	2	0	1	3	2	2	2

Pugh Method for Wing Geometry							
		Alternatives					
Evaluation Criteria		Rectangular	Tapered Wing	Swept	Elliptical	Dihedral	Rectangular/Tapered
Lift Generation		s	+	s	+	s	+
Stability		s	s	-	+	+	+
Weight		s	+	s	+	s	+
Manufacturability		s	-	-	-	-	-
Rate of Cost		s	+	s	+	-	+
Drag		s	+	-	+	s	+
	+	-	4	2	5	1	5
RESULTS	S	s	1	1	0	3	0
	-	-	1	3	1	2	1

Pugh Method for Fuselage				
		Alternatives		
Evaluation Criteria		Lifting Body	Circular	Rectangular
Performance		+	s	-
Weight		S	s	-
Cost		S	s	s
Manufacturability		-	s	+
	+	1	-	1
RESULTS	**S**	2	s	1
	-	1	-	2

Pugh Method for Tail				
		Alternatives		
Evaluation Criteria		Conventional	T-tail	Boom mounted
Stability		s	-	+
Weight		s	-	-
Size		s	s	s
Control		s	s	+
Efficiency		s	+	s
Manufacturing		s	-	-
	+	0	1	2
RESULTS	**s**	6	2	2
	-	0	3	2

Pugh Method for Landing Gear				
		Alternatives		
Evaluation Criteria		Tricycle	Bicycle	Tail dragger
Takeoff		s	-	+
Landing		s	+	-
Drag		s	-	+
Stability		s	+	-
	+	0	2	2
RESULTS	**s**	4	0	0
	-	0	2	2

Pugh Method for Motor Selection				
		Alternatives		
Evaluation Criteria		Cobalt 40	Cobalt 60	Cobalt 90
Cost		+	-	-
Weight		+	-	-
Size		s	s	s
Control		s	s	s
Efficiency		s	s	s
Manufacturing		+	-	-
	+	3	0	0
RESULTS	**s**	3	3	3
	-	0	3	3

Pugh Method for Battery Selection				
		Alternatives		
Evaluation Criteria		1400mah	1700mah	2400mah
Cost		+	-	-
Weight		S	-	-
Size		S	s	-
Capacity		-	+	+
	+	1	1	1
RESULTS	s	2	1	0
	-	1	2	3

Figure 3.2 Pugh Diagrams comparing benefits and drawbacks for selected components.

4.0 Preliminary Design

During the conceptual design phase many different configurations were evaluated with certain key design parameters. In the preliminary design phase these configurations were tested thoroughly and evaluated against refined parameters to obtain an optimum design that would meet all the mission requirements.

4.1 Design Parameters

In the preliminary design phase certain parameters were refined. The wing area, fuselage dimensions, and tail dimensions. Material options for every single structural component of the aircraft were evaluated. Power plant specs were refined and electrical connections optimized.

4.1.1 Wing Design Parameters

In the wing preliminary design, we chose to reduce the options of wing parameters. We eliminated certain parameters due to two causes. One cause was the fact the inexperience of the team in manufacturing prohibited us from using some options such as aerodynamic twists and dihedral angle. The other reason why some options were not chosen is the fact that these options simply did not apply to the engineering requirements. The parameters for the preliminary design concerning the wing did not change much. Just about all the parameters can be picked or eliminated from the beginning of the design

project. The parameters that were kept were aspect ratio, taper ratio, wingspan, wing incidence angle, wing area, wing location, and the NACA family of airfoils since it was decided that these would provide adequate lift. The reason behind the decision to only use NACA airfoils is that the Selig family of airfoils have a high drag profile associated with them. The mentioned parameters will be investigated and optimized further in the detail design phase.

4.1.2 Fuselage Design Parameters

The main parameters that influenced the design of fuselage thru the conceptual design were: the payload size, drag, and cross sectional area. These parameters were refined to include weight, structural integrity, ease of loading and unloading, and material selection, and the addition all the other components housed in the fuselage.

The payload size of 12 softballs overall (10 in the fuselage) was unchanged. The overall sizing of the fuselage was based from the dimensions of the softballs and the batteries solely. Any excess area was reduced to lower overall weight of the fuselage. In key structural points like the mounting point of the wing and landing gear, added structural stiffness was added by increasing the width of the material at these points. This will allow the fuselage the fully support these components.

Once the sizing of the fuselage was refined and optimized, the material selection could be evaluated. The materials chosen must be lightweight but yet rigid enough to withstand all the applied stresses on the fuselage. Materials considered were balsa wood, foam, and plywood.

Balsa wood is used extensively in model aircraft construction and is easily obtained. It has a very low density and a reasonable modulus of elasticity giving it an excellent stiffness to weight ratio. Blue foam has a very low density but a low modulus as well, so it must be reinforced by balsa to make it rigid. The disadvantage with the blue foam is that a comparable wood structure can be made with a lower mass making the blue foam not efficient. The material evaluated was microlite ply, which is stronger and heavier than balsa. Both balsa and microlite were chosen as good materials with balsa comprising most the fuselage with microlite used in high stress areas like the front bulkhead and wing mounting points.

With the fuselage sizing and material selection optimized, the payload loading and unloading was optimized. First the nose cone of the airplane was optimized to reduce drag and weight by reducing the frontal area. Since the motors will not be placed in the nose cone housing, the stresses are greatly reduced. The nose cone is there for aerodynamic purposes and for loading the payload. Materials considered for the nose cone design are balsa and blue foam covered with carbon fiber. Balsa is lightweight, strong and easily available. The only disadvantage is the complexity of the nose cone design and the added weight would be great for a non load-bearing component. Blue foam is very light and pretty simple to shape with a basic knowledge foam cutting using a hot wire. A carbon fiber shell will be placed over the foam to strengthen it so it can be carved hollow top reduce weight. The final result will be a lightweight rigid nose cone. A balsa spar can be placed along the front and bottom to allow for hinge

and latch points to be attached. With the hinging of the nose cone downward, the payload will be readily available and can quickly be loaded without removing any major components. This reduces the time of loading which in the end increases the overall score.

4.1.3 Empennage Design Parameters

In the preliminary design the main parameters governing the design of the empennage were refined and different materials were evaluated. The parameters that governed the conceptual design were sizing, stability, and control. Now in the preliminary design different configurations of booms and control surfaces will be evaluated as well. In addition tail has been modified to hinge up so that the payload can be unloaded from behind. This allows for easy access of the payload from the front and from the back.
Once the size of the boom was determined it was minimized to reduce weight and drag. The advantage of the single boom design is that part of the payload can be stored in the tail and then hinged to unload it. The added payload space greatly optimized the tail area. With the boom sized and optimized different horizontal and vertical stabilizers configurations could be evaluated. A single horizontal and vertical stabilizer was implemented over double stabilizers to reduce cost. Double stabilizers would need two servos, which would increase the RAC and lower our overall score.
Three different methods were evaluated in the manufacturing of the boom and control surfaces. Balsa rib and spar construction with balsa sheeting is the most common. Foam composites were considered in the control surface construction, but were quickly discarded due the complexity of the design. The final configuration consists of balsa ribs with balsa spars and no sheeting to reduce weight. The skin of the tail will be covered with monocote to make it aerodynamic.

4.1.4 Landing Gear Design Parameters

In the conceptual design the solid spring tricycle gear was chosen. It is the most stable and reliable gear. Different materials were analyzed to determine the best design. Aluminum, carbon fiber, and Kevlar were chosen. Aluminum was an ideal material because of its flexibility. Deflection was limited to prevent a prop strike. A stock landing gear made out of aluminum was found meeting all of our parameters in prop clearance, load bearing, and stable.

4.1.5 Power Plant Design Parameters

Modifications to fuselage and wing were created, which presented new alternative designs for the power system. A much faster aircraft was the deciding factor for this aircraft and competition. Since the Astroflight 661S required 36 cells for a total of nearly five pounds just for batteries per motor, therefore a new motor had to be chosen. One of the analyzed power systems that were evaluated was the

Astroflight 640G motor with a 1.63:1 gear ratio using smaller 13x8 propeller with a range of 18-21 cells for best performance. This configuration's overall weight was much lighter than that of the 661S. Since the battery amount was two times smaller an additional motor could be added to obtain the 36-battery total. With the addition of a second motor it would provide a higher wing and power loadings, along with better weight distribution and more efficient thrust which would allow for a lower power system to give the equivalent performance of a single motor aircraft. The addition would also create a higher structural weight, and to allow for a safety margin in the event of an engine out.

The next assessment would be to analyze whether to use the 640G or 640 in other words a geared or direct-drive motor. The difference between the two types of motors is that the geared motors use a gearbox, which is especially useful when speed is the major player for the aircraft. The gear-driven motors allow one to spin big propellers at lower RPMs, which loosely correspond too a much higher thrust at lower flight speeds. This is particularly useful for slower-flying planes, such as when the aircraft is loaded. The direct-drive motors spin small props at very high RPMs, mainly used for aerobatic aircraft. Since our aircraft will be used to create higher thrust the gearbox would be the best choice for the aircraft and the assessment was correct.

Following the evaluation of the decision on the new motor, the number and type of cells used was to be assessed. The first consideration into gaining more power for the plane was a higher capacitance battery, preferably by Sanyo, and then finding the specs for each motor to be considered and or used. Astroflight's variety of motors presented many alternatives to battery selection. After documentation was gathered for each motor, then the battery selection came into play. Sanyo's 1400mah, 1700mah, and 2400mah batteries were all relevant choices, but the cell weight posted the biggest threat for using any high power motor required larger, heavier, high capacitance batteries, which cut in on the five pound weight limit. So many problems needed to be solved to find an ideal battery for the Astroflight 40G motor. The first information that was gathered was from Astroflight itself. What batteries and how many did it recommend for its motor. According to data assembled, an excellent battery to use with the Astroflight 40G motor is the Sanyo 1700SCRCmah. This motor is nominally rated for 18-cell operation but is often run on 21 cells for more power. With higher power the faster the aircraft will fly with the proper propeller. Since the weight of the 1700mah is 54 grams per battery, or a total of 42 batteries for five pounds of batteries to meet the contest rules. So to meet the needs of two motors the battery number would double. For the maximum efficiency for the motor, the number of batteries would either be 20 or 21 batteries, or 40 to 42 batteries to meet the 5-pound maximum limit on weight not counting the soldering and shrink-wrap.

The background on the Sanyo 1700 SCRC consists of being one of the most popular Ni-Cad cells uses in electric models today. This cell has excellent capacity, has a very low five milliohm internal resistance, and can be fast charged thousands of times. The cell weighs 1.88 ounces. The voltage average of the Ni-Cad during a one-hour discharge is 1.25 volts. The capacity of these cells is nominally 1.75 amp

hours. So the electrical energy available is 1.25 x 1.75 = 46 Watt-hours. The equivalent potential energy is 121,925 foot-pounds.

The next assessment was to wire the batteries in series or parallel. A parallel configuration will cause the motor run will be short and the voltage depressed because of high battery internal losses. The motor's RPMs will be lower and the battery will get very hot. In the series, the configuration will yield longer motor runs and higher RPMs (less battery internal losses due to heat). The series configuration was selected based on the stated facts.

The selection of the propeller was one based on predictions by MotoCalc and Astroflight's information on the Astro Cobalt 40G motor. Such selections had to be made to gain the maximum motor efficiency out of the motors and with the large amount of batteries. Astroflight stated that the best propeller for the Astro 40G is the 13x8 prop for the highest efficiency. But to help verify if the 13x8 is accurately the best prop for the job other propellers were tested to insure its validity.

5.0 Detail Design

Once the final configuration resulting from the preliminary design phase was decided, the aircraft had to be optimized as much as possible. To accomplish this task, exact dimensions for areas, lengths, mounting angles, weights, velocities, and many other parameters have to be known. A detailed design must be gained using several tools such as VisualFoil, Finite element analysis in IDEAS, Pro/Engineer for solid modeling, Excel spreadsheets, and Mathcad worksheets. Once the detail design is conceived, the manufacturing process can begin with confidence that the designed aircraft has been optimized.

5.1 Component Detailed Analysis

The component analysis describes the manner in which the team arrived at specific values for the parameters mentioned above. The specific values for dimensions of the wing assembly, fuselage assembly, empennage assembly, landing gear assembly, and power system configuration and placements are discussed in the following paragraphs.

5.1.1 Wing

In the wing design, one simply must remember that the wing has to be able to generate enough lift in order for the aircraft to get off the ground. The selection of the airfoil, wing geometry, wing area and span was decided during the detailed design phase. In order to maximize the lift, optimization of angle of attack, flap size and location, aileron size and location must also be determined during the detailed design phase. Research indicated the selection of the airfoil should remain the NACA 4412. This airfoil has done very well in past competitions of the same sort. The wing geometry as mentioned before would

be that of a tapered rectangular wing with aspect ratio of 14 and a taper ratio of .45 and a wingspan of 8 feet to approximate an ellipse. The root chord and tip chord of the wing are .788 feet and .355 feet respectively. After careful review of Excel spreadsheets and calculations in VisualFoil, the angle of attack for the wing mounting should be at 0 degrees with respect to the ground in order to provide enough lift. Flaperons will be used to minimize the RAC and weight of the aircraft as well as decreasing the speed of the aircraft on landings as found necessary. The Flaperons will be 15% of the wing chord and 1.25 feet in length and begin at a distance of three inches from the fuselage. This was determined from past research done by previous teams in which research shows that the flaperons should extend 50-90% of the wingspan, which makes the 1.25 feet 63% of the wingspan. Also, according to past teams' research, the structure of the wing will be able to handle the static and dynamic loads that will be experienced. This is in effect of the two spars that will be place in the wing. The placement of two spars was prompted by the need to add to the structural stability of the wing structure as well as provide support the motors, which would be mounted on the wing. While blue foam, which was used instead of white foam, and carbon fiber

5.1.2 Fuselage

The fuselage, although it proved to be the most complex in structural integrity, it undergoes the most rigors for standard operation. As stated before, the fuselage will be carrying most of the payload, which includes the ten softballs, 4.35 lbs of batteries, and most of the electrical components. So the fuselage must be molded around the cargo, be lightweight, and maintain aerodynamic efficiency. After all these considerations were thought through construction could be accomplished.
Essentially, after generating Pro-E models of the balsa ribs for the fuselage, the ribs were cutout, placed together using stringers, and wrapped in lightweight balsa skin.

5.1.3 Empennage

The empennage as one might expect is closely associated with the design of the fuselage and the aircraft as a whole. In doing the stability and control calculations in Mathcad and Pro/Engineer, the dimensions for the tail boom and the vertical and horizontal surfaces were defined. The moment arm or length of the tail boom was determined to be 1.75 feet in length. The tail boom would taper down to an area of approximately 1" X 1" in square dimensions. The horizontal surface of the tail, which has an area of 0.914 feet squared, would then be attached to the top of the tail boom itself. The vertical tail, which has an area of 0.928 feet squared, would then sit on top of the horizontal tail. The taper ratios for both surfaces is 0.4.

5.1.4 Landing Gear

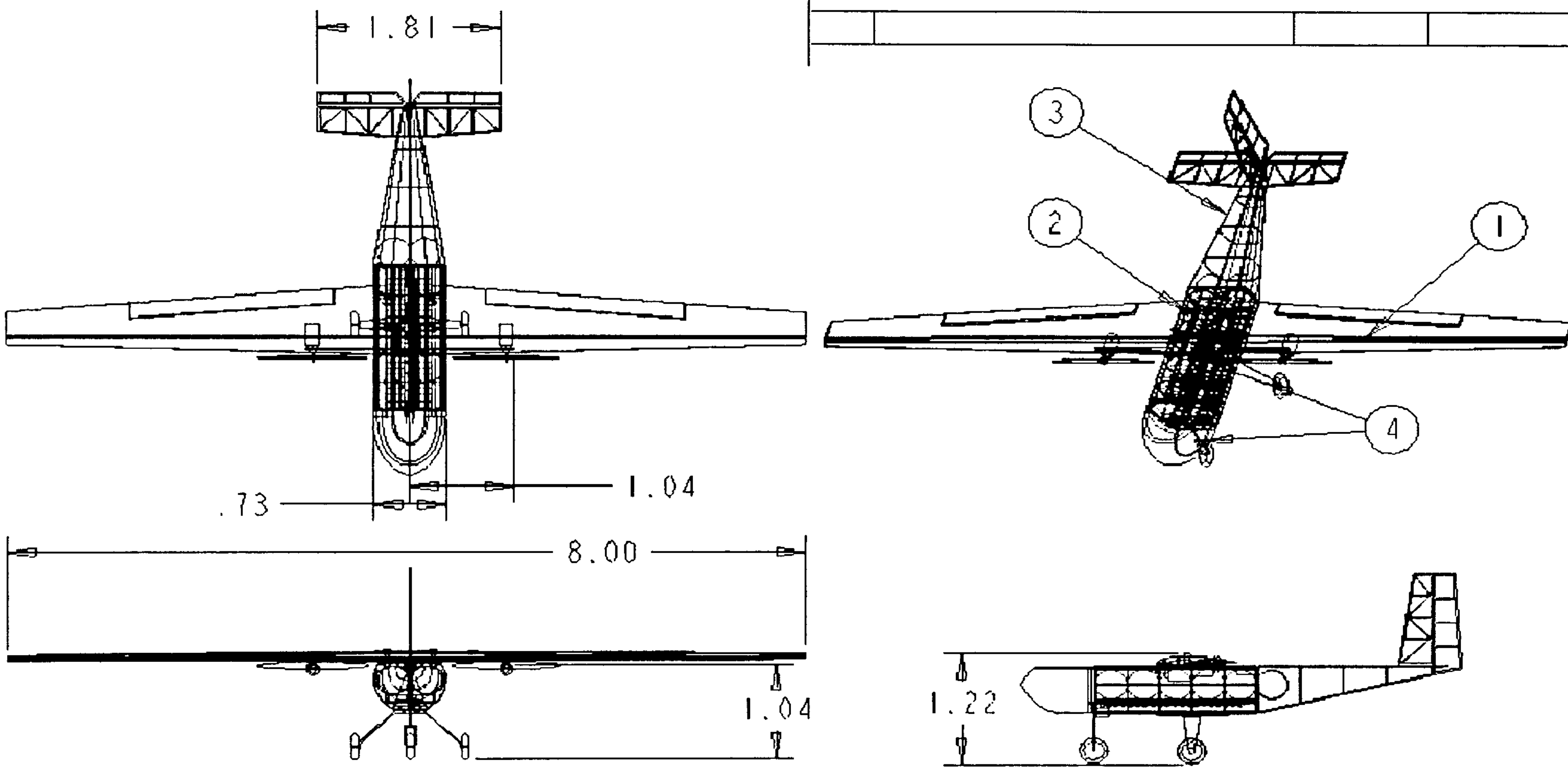
After careful consideration of loads that would be encountered in the competition using IDEAS and Pro/Engineer software, the landing gear would be composed of hand manufactured T-6(6061) aluminum. The dimensions for the landing gear, which were done to minimize the weight as mentioned before are as follows: The thickness of the landing gear itself is 3/16 in. with a maximum width between the tires of 12.25 in.. The height of the landing gear is 5 in. in order to provide enough clearance for the propellers, which are attached to the wing. The maximum contact length with the fuselage, which can be seen on the front edge of the fuselage is 3.5 in.. The landing gear deflection that could be seen from a forty pound load applied to it which a 2g load was only 4.057E-3 in..

5.1.5 Power System

For final optimization of the power plant, the fuselage and wing had to match the preliminary motor-battery-propeller configuration. The size of the propeller was known along with the motor, the question was where to mount the motors and how it would be done. The first consideration was on the diameter of the propeller. Knowing the diameter was thirteen inches; the clearance for the landing gear and fuselage must be at least eight inches to obtain a half-inch clearance. In order to achieve this clearance, the wing must be able to withstand the weight applied by the motor and propeller, and still be aerodynamically correct with this addition. The motors would be mounted on a reinforced wing spar using a commercially manufactured motor mount, then have foam molded around each motor to ensure aerodynamic qualities. Other issues of concern were that of the batteries and overheating. This problem has plagued past teams and will not be a setback for this team. In the event that an overheat does occur, a scoop for ventilation will be added to the fuselage to permit a constant stream of air to flow over the batteries allowing heat transfer.

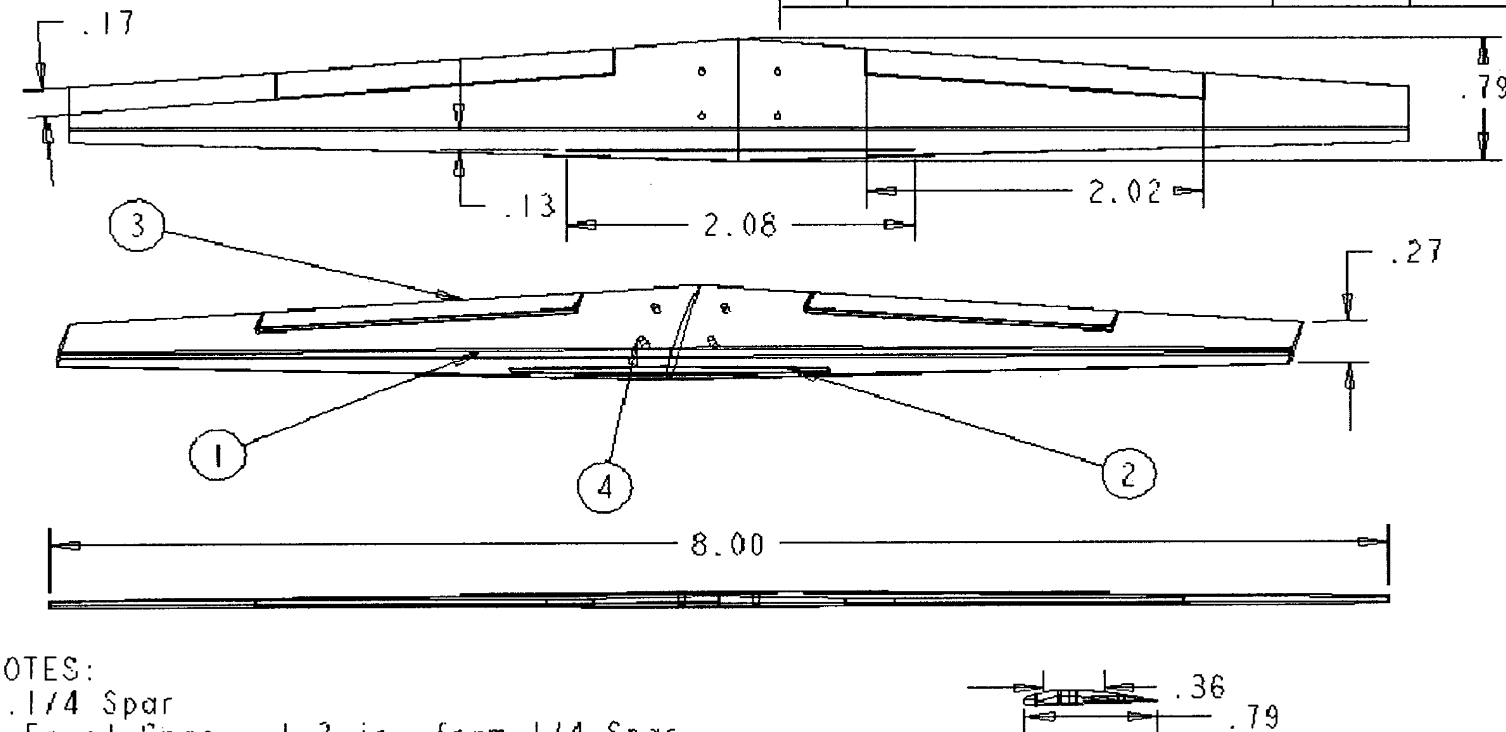
5.2 Drawing Package

The drawing package for the aircraft and main individual components can be seen in the following pages.



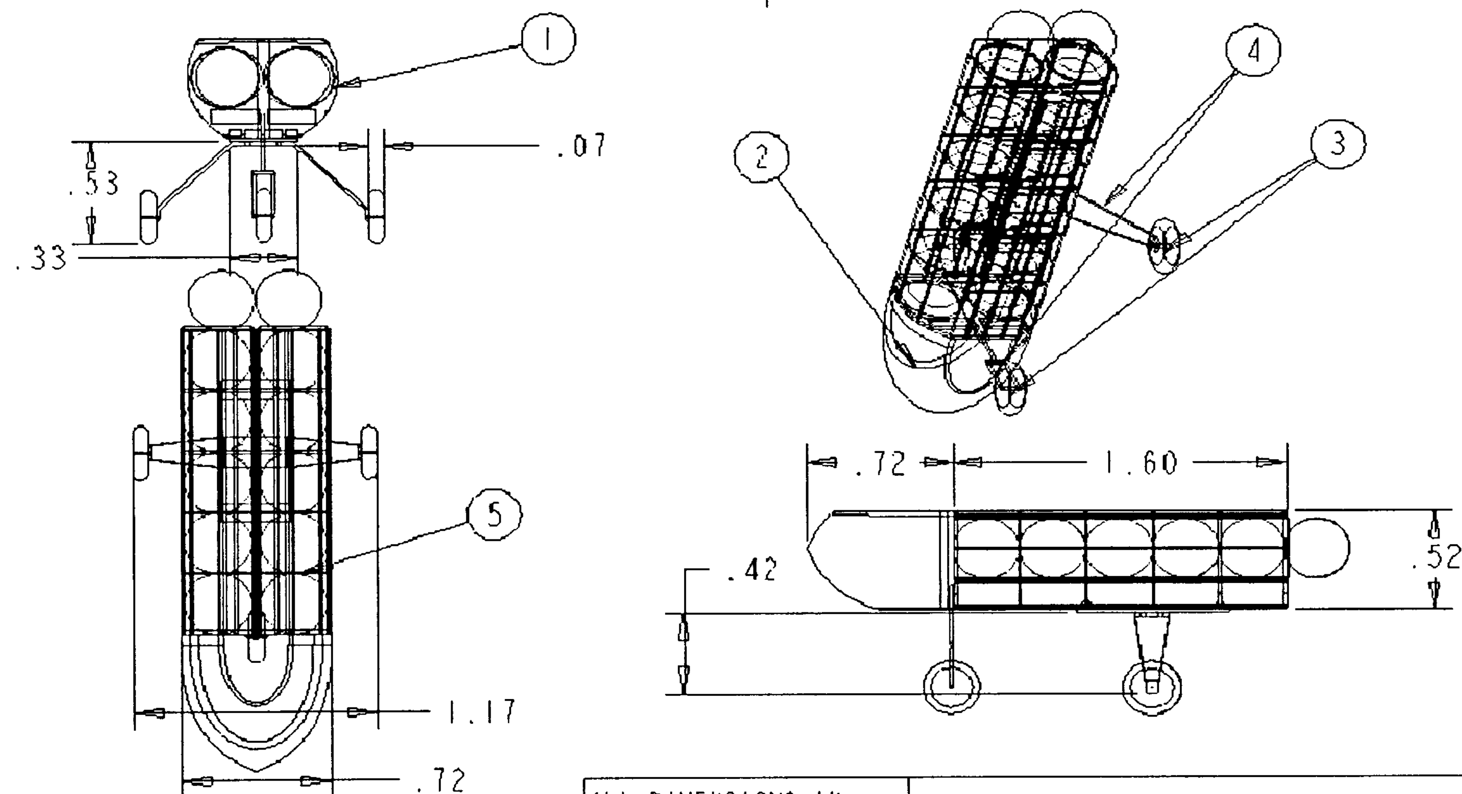
NOTES:
1. Wing Assembly
2. Fuselage Assembly
3. Empennage Assembly
4. Landig Gear Assembly

ALL DIMENSIONS IN FEET UNLESS OTHERWISE STATED	UCF DBF 2002 University of Central Florida
	Plane Assembly
DRAWING NO. UCF-1	size A · SHEET 1 OF 5 · REV. 0
DRAFTED BY: BERNIE SAM	MARCH 1, 2002



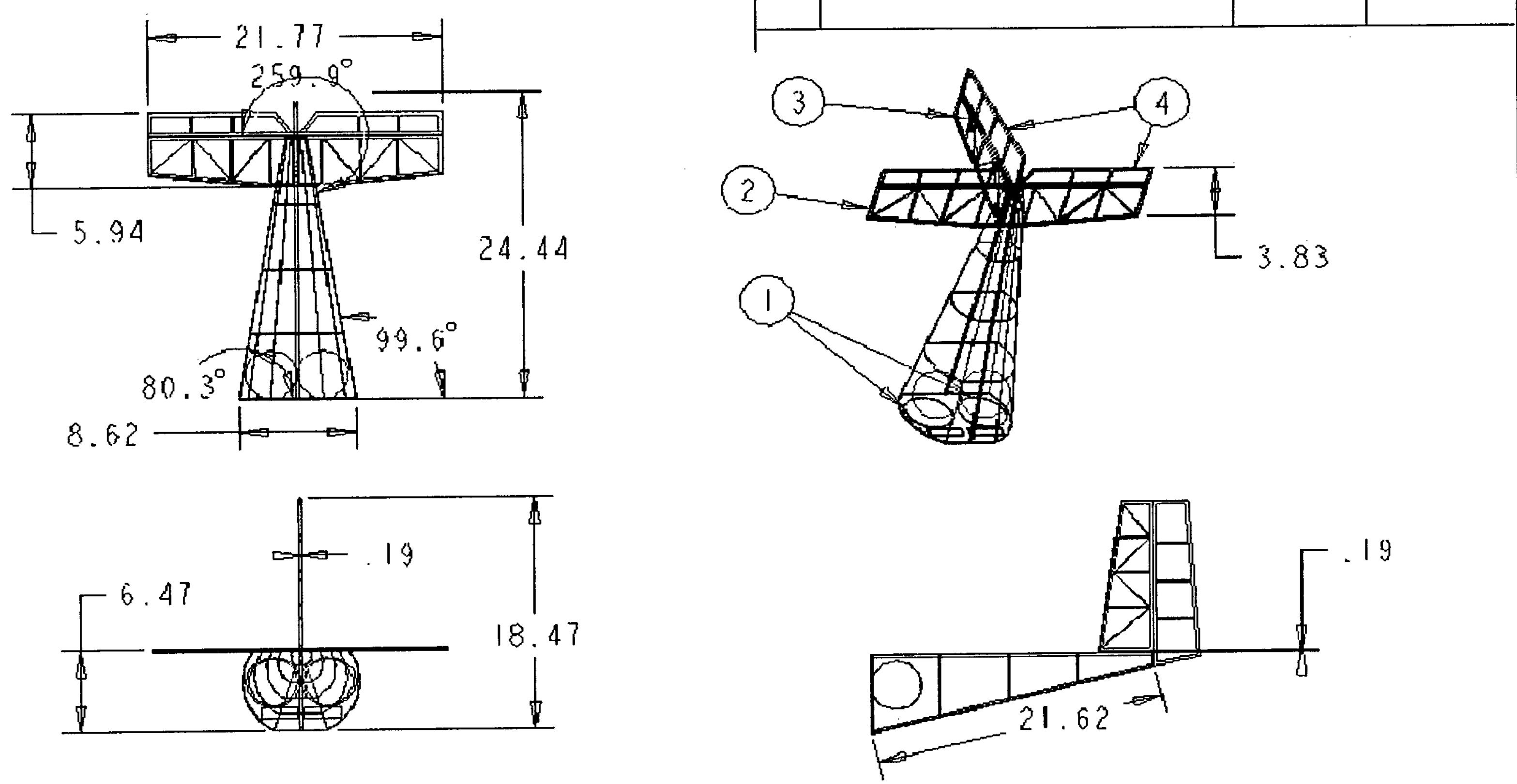
NOTES:
1. 1/4 Spar
2. Front Spar - 1.3 in. from 1/4 Spar
3. 0.15c Flaperons
4. 1/2 in. Holes for Nylon bolts

ALL DIMENSIONS IN FEET UNLESS OTHERWISE STATED	UCF DBF 2002 University of Central Florida		
	Wing Assembly		
DRAWING NO. UCF-1	size A	SHEET 2 OF 5	REV. 0
DRAFTED BY: BERNIE SAM		MARCH 1, 2002	



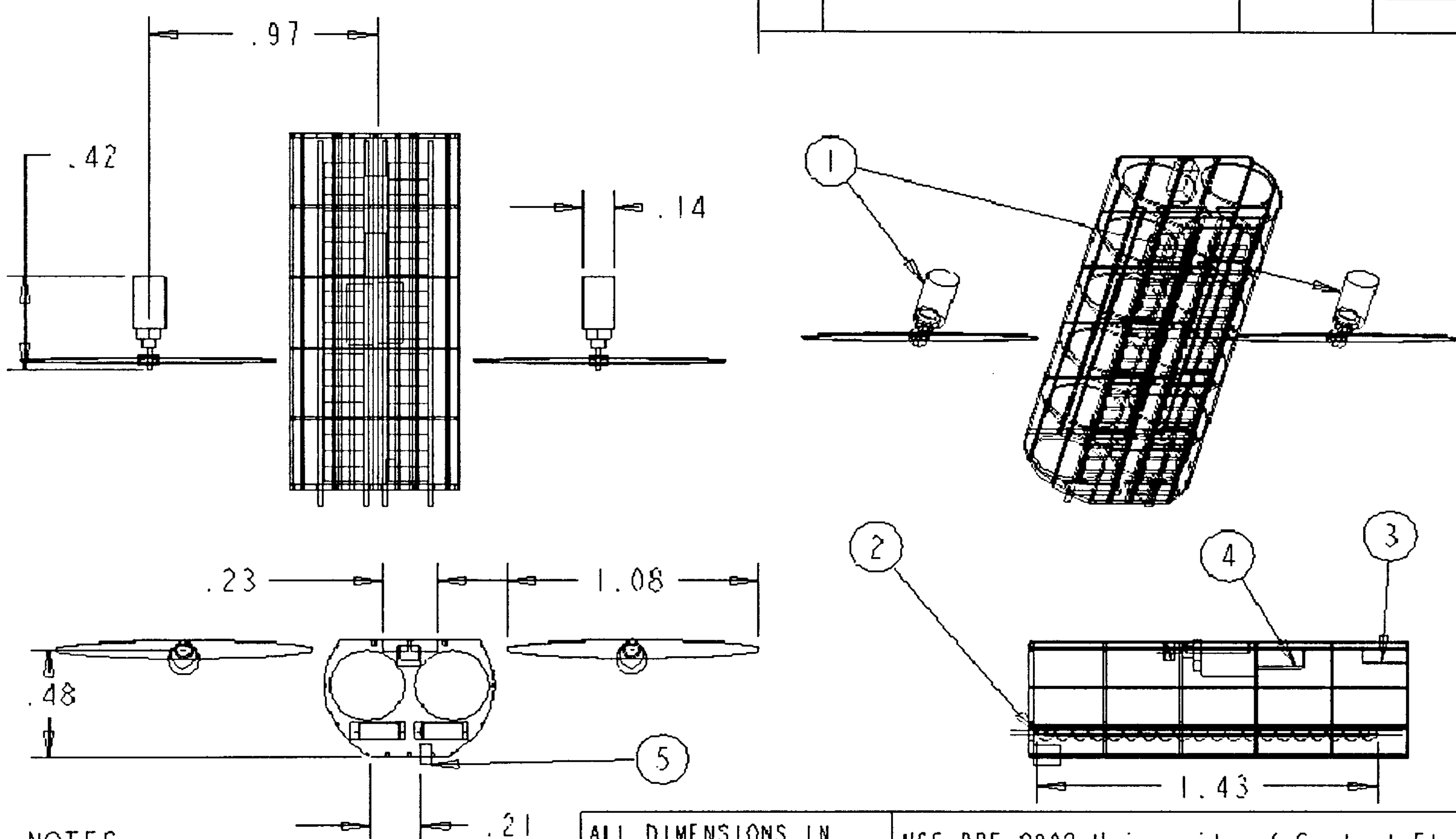
NOTES:
1. Payload Bay
2. Nose Structure
3. 3" Tires
4. Landing Gear Assembly
5. Bulkhead

ALL DIMENSIONS IN FEET UNLESS OTHERWISE STATED	UCF DBF 2002 University of Central Florida
	Fuselage Assembly
DRAWING NO. UCF-1	size A — SHEET 3 OF 5 — REV. D
DRAFTED BY: BERNIE SAM	MARCH 1, 2002



NOTES:
1. Payload Bay
2. Horizontal Tail Surface
3. Vertical Tail Surface
4. Control Surfaces

ALL DIMENSIONS IN FEET UNLESS OTHERWISE STATED	UCF DBF 2002 University of Central Florida		
	Empennage Assembly		
DRAWING NO. UCF-1	size A	SHEET 4 OF 5	REV. 0
DRAFTED BY: BERNIE SAM		MARCH 1, 2002	



NOTES:
1. Astroflight Motors
2. Battery Packs
3. Receiver
4. Receiver Battery
5. Landing Gear Servo

ALL DIMENSIONS IN FEET UNLESS OTHERWISE STATED	UCF DBF 2002 University of Central Florida		
	Power Plant Assembly		
DRAWING NO. UCF-1	SIZE A	SHEET 5 OF 5	REV. 0
DRAFTED BY: BERNIE SAM		MARCH 1, 2002	

5.3 Flight Performance

The final stage of the detailed design phase was to calculate the performance of the aircraft while in flight. These calculations include take off and landing distances and speeds, rate of climb, velocity at cruise, range and endurance, and handling characteristics.

5.3.1 Take off and Landing

According to calculations done in Mathcad, the take of distance for our aircraft is 137.594 feet reaching a take off velocity of 40.49 miles per hour. Calculations also show that the landing distance is 102.253 feet at a landing velocity of 48.84 miles per hour. The stall speed of the aircraft is 28.534 miles per hour. The distances calculated for take off and landing are well below the requirement of a take off and landing within 200 feet.

5.3.2 Rate of Climb

The rate of climb was obtained by graphing the rate of climb curve as a function of velocity using Mathcad and was found to be 14 feet per seconds.

5.3.3 Range, Endurance, and Cruising Speed

The range and endurance of the power system was calculated with Mathcad. Using the eighteen, 1700mah batteries per Astro 640G motor, the voltage that each motor will see will be 21.6 volts, not taking into account the loses due to connectors, wires, speed controller, etc. Estimating the flight distance for each sortie and flight speed per sortie at different throttle positions, the total time to complete the course including one minute for loading and unloading would take approximately 7.79 minutes. This time does not take into account any problems that might occur during loading and unloading. Employing these parameters, the total battery amperage consumed would be roughly 1600mah to complete the course, leaving 100mah available. According to the calculated data shown below in figure 5.1, the aircraft should perform as predicted leaving time available for any obstacles that might alter the operation of the aircraft, such as weather conditions.

Flight Score:

$Course\_Length = 3604ft$ With loop

$Time\_to\_Load := 60s$

$Velocity\_of\_Plane := V \quad V = 50\frac{ft}{s}$ or 30 mi./hr

$$Time\_with\_Loop := \left(\frac{Course\_Length}{Velocity\_of\_Plane}\right) \qquad Time\_no\_Loop := \frac{(Course\_Length - 702ft)}{Velocity\_of\_Plane}$$

$Time\_with\_Loop = 72.08s$ $\qquad Time\_no\_Loop = 58.04s$

$$Total\_Mission\_Time\_One := (4 \cdot Time\_with\_Loop) + (2 \cdot Time\_no\_Loop) + Time\_to\_Load$$

$Total\_Number\_Laps\_One := 6 \qquad Total\_Mission\_Time\_One = 464.4s$

$$Single\_Flight\_Score\_One := \frac{(Total\_Number\_Laps\_One + Number\_of\_Balls) \cdot s}{Total\_Mission\_Time\_One}$$

$Single\_Flight\_Score\_One = 0.039$

$$Total\_Mission\_Time\_Two := (4 \cdot Time\_with\_Loop) + (2 \cdot Time\_no\_Loop) + Time\_to\_Load$$

$Total\_Number\_Laps\_Two := 6$

$$Single\_Flight\_Score\_Two := \frac{(Total\_Number\_Laps\_Two + Number\_of\_Balls) \cdot s}{Total\_Mission\_Time\_Two}$$

$$Total\_Mission\_Time\_Three := (4 \cdot Time\_with\_Loop) + (2 \cdot Time\_no\_Loop) + Time\_to\_Load$$

$Total\_Number\_Laps\_Three := 6$

$$Single\_Flight\_Score\_Three := \frac{(Total\_Number\_Laps\_Three + Number\_of\_Balls) \cdot s}{Total\_Mission\_Time\_Three}$$

$$Total\_Flight\_Score := Single\_Flight\_Score\_One + Single\_Flight\_Score\_Two + Single\_Flight\_Score\_Thre$$

$Total\_Flight\_Score = 0.116$

$$Overall\_Score := \frac{Written\_Report\_Score \cdot Total\_Flight\_Score}{Rated\_Aircraft\_Cost}$$

$Overall\_Score = 0.777$

Power/Range:

battery_amps := 1700mA

motor_max_amps := 35amp

motor_static_amps := 2.5amp

battery_voltage := 1.2volt

motor_voltage := 19.9volt

number_of_batteries := 18

RPM := 6987

Prop_Pitch := 8in

Prop_diameter := 13in

Max_Speed := 53mph

(APC Prop Size: 13x8)

M_V := (battery_voltage)·(number_of_batteries)

$M_V = 21.6V$

wing_loading := 59.472

Stall_Speed := $3.7\sqrt{\text{wing\_loading}}$

Stall_Speed = 28.534

*Batteries will be in series to reach voltage of 21.6 volts and maintain a millamperage of 1700 mah. (Flight distance estimated.)

flight1_distance := 2·3604ft

flight2_distance := flight1_distance

flight3_distance := 2·2900ft

(All flight speeds are estimations of how fast the aircraft will actually fly.)

*flight1 speed consists of plane running at 70% throttle unloaded

flight1_speed := Max_Speed·0.70

flight1_speed = 37.1 mph

*flight2 speed consists of plane running at 55% throttle and loaded

flight2_speed := Max_Speed·0.55

flight2_speed = 29.15 mph

*flight3 speed consists of plane running at 70% throttle unloaded

flight3_speed := Max_Speed·0.70

flight3_speed = 37.1 mph

$$\text{time1} := \frac{\text{flight1\_distance}}{\text{flight1\_speed}} \qquad \text{time1} = 2.208\,\text{min}$$

$$\text{time2} := \frac{\text{flight2\_distance}}{\text{flight2\_speed}} \qquad \text{time2} = 2.81\,\text{min}$$

$$\text{time3} := \frac{\text{flight3\_distance}}{\text{flight3\_speed}} \qquad \text{time3} = 1.777\,\text{min}$$

$total\_time := time1 + time2 + time3$

$total\_time = 6.794\,min$ **(Total time does not include time to load and unload aircraft.)**

$Total\_time\_load\_unload := total\_time + 1min$ $\qquad Total\_time\_load\_unload = 7.794\,min$

$full\_throttle := 22.8amp$ (Prop and motor combination draw in amps)

$throttle1 := 0.70 \cdot full\_throttle$

$throttle2 := 0.70 \cdot full\_throttle$

$throttle3 := 0.55 \cdot full\_throttle$

$$range := \frac{time1 \cdot throttle2 + time2 \cdot throttle3 + time3 \cdot throttle1}{3600s}$$

$$range = 1647.101\,mA$$

$range1 := \frac{time1 \cdot throttle2}{3600s}$ $\qquad range2 := \frac{time2 \cdot throttle3}{3600s}$ $\qquad range3 := \frac{time3 \cdot throttle1}{3600s}$

$range1 = 587.273\,mA$ $\qquad range2 = 587.273\,mA$ $\qquad range3 = 472.556\,mA$

$remaining\_amps := battery\_amps - range$

$remaining\_amps = 52.899\,mA$

(Total speed with APC propeller installed.)

$Pitch := 8$ $\qquad RPM := 6.987$

$Total\_Speed\_13x8\_Prop := Pitch \cdot RPM$

$Total\_Speed\_13x8\_Prop = 55.896$

(Fuse amperage)

$Fuse := 22.8A \cdot 1.2 = (27.36A)$

$$Duration := 60min \cdot \frac{1.7A}{22.8A}$$

$Duration = 4.474\,min$ (Duration = 4.474 minutes at full throttle)

Figure 5.1 Battery Life and Endurance

5.4 Rated Aircraft Cost

The total score that team can achieve is calculated by three variables: the written report score, the total flight score, and the rated aircraft cost. The cost form given to compute the rated aircraft cost (RAC) is utilized to compute the estimated cost of the aircraft in thousands of dollars. The RAC is composed of three sections: the manufacturers empty weight (MEW), the rated engine power (REP), and the manufacturing man-hours (MFHR).

Using the following formula seen below, the rated aircraft cost was calculated for the aircraft. The RAC for the aircraft was determined to be 13.027.

<u>Rated Aircraft Cost</u>:

Rated Aircraft Cost $ (Thousands) = (A*MEW + B*REP + C*MFHR)/1000

A = $100

B = $1500

C = $20/hr

Manufacturers Empty Weight (MEW) = 13.308lb

Rated Engine Power (REP) = (1+. 25*(# engines-1)) * Total Battery Weight = (1 +. 25*(2-1))*4.35

REP = 5.4375

Manufacturing Man Hours (MFHR) = Sum of Work Breakdown Structure:

Wing Span: 8hr/ft = 8hr * 8ft = 64hr

Control Surfaces: 3hr/control surface = 3hr * 2 control surfaces = 6hr

Fuselage: 10hr/ft maximum body length = 10hr * 3.07ft = 30.7hr

Empennage: 5hr/Vertical Surface = 5hr * 1 = 5hr

10hr/Vertical Surface (Active Control) = 10hr * 1 = 10hr

10hr/Horizontal Surface = 10hr * 1 = 10hr

Flight Systems: 5hr/servo = 5hr * 5 = 25hr

Propulsion System: 5hr/engine = 5hr * 2 = 10hr

5hr/propeller = 5hr * 2 =10hr

RAC = ($100/lb * 13.308lb + $1500/lb * 5.4375lb + $20/hr * 177.004hr)/1000

<u>RAC = 13.027</u>

ADVISOR SIGNATURE Roger Johnson

6.0 Manufacturing Plan and Processes

Once the design process was complete, the manufacturing of the aircraft was carefully planned. The manufacturing of the different components had to be done carefully and with much patience. All the manufacturing for all the components was planned ahead of time in order to ensure inexpensive, efficient, and correct assembling of the aircraft. Considerations in cost, skill levels required, availability, and manufacturing processes were taken into account before the actual building began.

6.1 Figures of Merit

Several figures of merit were investigated to come up with the best manufacturing plan. Figure 6.1 shows how cost, skill levels required, availability, and time required influenced manufacturing.

6.1.1 Cost

While the teams' score at the competition will not be affected by the cost of manufacturing, the team must work within the allotted budget set by school monies. In order to stay within this budget, the team ranked several manufacturing processes according to its associated cost as shown in figure 6.1. Figure 6.2 shows the total cost of manufacturing in detail.

6.1.2 Availability

The availability of materials and processes had to be also taken into consideration. Many of the materials were not also very expensive but were also hard to work with and difficult to acquire.

6.1.3 Required Skill Levels

Some of the components and processes required higher skill levels than others. Many of the team members were actually building for the first time. This limited the types of manufacturing chosen since we had to keep processes skills needed at a minimum.

6.1.4 Time Required

As the team was small in size so was our time available to complete the construction by the beginning of March in order to have enough time for testing and final modifications. Since some processes in manufacturing take longer than others we had to also analyze how long each process lasted.

Component	Process	Cost	Skill Level Required	Availability	Time	Totals
		Figures of Merit				
Wing	Foam Core w/ Spar and Balsa	+	-	+	-	0
	Foam Core/Spar/Carbon Fiber	+	0	+	0	2
	Ribbed Structure and Monokote	+	-	-	-	-3
	All Composite	-	-	+	-	-3
Fuselage	Balsa Ribs and Stringers	+	+	0	0	2
	Composite Shell	-	-	-	-	-4
	Box Beam	0	-	0	+	1
Tail Boom	Truss	+	-	+	-	2
	Balsa Ribs and Stringers	+	0	+	+	3
Vertical and Horizontal Stabilizers	Foam Airfoils	-	-	0	-	-3
	Stringers	+	+	+	+	4
	Ribs and Monokote	+	-	0	-	-2
Landing Gear	Composite	-	-	-	-	-4
	Metal	+	+	0	+	3
Motor Mounts	Plywood Mount	+	+	+	0	3
	Outsource	-	+	0	-	1

Figure 6.1 Figures of Merit

6.2 Wing

The wing, which is maybe the most important component of the aircraft, had to be constructed so that its surface finish would provide an excellent flow characteristic in order provide the necessary lift while being able to endure the aerodynamic forces applied to it. Due to the size of the team, ample supplies, and the limited experience of wing manufacturing, the team had to resort to using a blue foam core wing with two light ply spars for wing and motor support. The wing was then reinforced with carbon sheathing. The wing deflection was at maximum 0.9007 in. The cost for the manufacturing of one of these wings is approximately $323.00.

The construction of the wing was begun by making three separate airfoil templates for a total of 4 sections for the wing. The four sections were a left inboard section, a left outboard section, a right inboard section, and a right outboard section. The foam was then cut using a hot wire, which was guided by the templates. Once the foam pieces were cut, they were sanded down to remove any imperfections left over from the hot wire cutting. Once this was done, a cut to each one of the four pieces was made at the location where the quarter chord spar would be placed and a cut at 1.3" in front of the quarter chord location was also made for the placement of the second spar. The spars were then attached into place using epoxy and the wing was then coated with carbon fiber sheeting and epoxy using a vacuum and Mylar in order to achieve a smooth surface. Once the epoxy was cured, the flaperons and servo motor seatings were cut out using a Dremmel tool. Four holes were then drilled thru the wing for the attachment to the fuselage.

6.3 Fuselage

The fuselage being the part that binds all other parts together is the most complex structure of all since it also must carry the payload and most power components. The fuselage must also be able to handle the most amounts of strains and stresses experienced during normal operation and therefore must be structurally adequate. The fuselage consists of five 1/8 in. thick light ply bulkheads spaced at 3.37 in. apart, one ¼ in thick light ply bulkhead at the head of the fuselage for the attachment of the front landing gear and nose cone, seven 3/16 in. stringers for the top and bottom surfaces, eight 1/8 in. stringers for the side walls, two ¼ in. thick light aircraft ply pieces for wing attachment, one ¼ in thick light aircraft ply piece for the main landing gear attachment, two tube shaped compartments made of Mylar to be used as the payload bays, and 1/16 in. thick balsa sheet to cover the fuselage which would then be monokoted. The approximate cost for the manufacturing of the fuselage is $50.00.

The construction of the fuselage began by binding together five pieces of 1/8 in. light ply sheets and one ¼ in. light ply sheet together to ensure the similarity between all of the bulkhead shapes, which were cut using a band saw. A pattern for a single bulkhead was printed out from an assembly drawing done to scale and attached to the top piece of ply which was then used to guide the band saw in cutting the bulkheads and notches for the stringers to the right dimensions. The hole patterns for the payload and battery compartments were then cut using a router bit. The bulkheads were then sanded down to ensure smooth surfaces. Once the bulkheads were finished, the stringers were attached in their perspective positions. The three pieces of ¼ in. thick light aircraft ply were cut and drilled with corresponding holes for wing and landing gear attachment and attached to the underside and the top of the fuselage with epoxy. The minor power components were then attached to the fuselage in their corresponding spots by way of glue and screws and the fuselage was then skinned with a thin 1/16 in. thick balsa sheet. The final detail consisted of simply applying Monokote to the exterior of the fuselage.

The nose cone of the fuselage, which is 9 in. long in diameter for aerodynamic purposes, was constructed by using white foam, shaped to the correct dimensions and coated with fiberglass sheeting and epoxy. Once the epoxy cured, the foam was removed leaving only the fiberglass. A 1/8 in. piece of light ply was then attached to the nose cone in order to be able to hinge it to the fuselage.

6.4 Empennage

The single tail boom of the aircraft was made in a similar manner as the fuselage in that it consists of similarly shaped 1/8 in. thick bulkheads and assembled in the same fashion. The difference is that since the empennage would be carrying hardly any payload, the bulkheads were spaced farther apart than the bulkheads in the fuselage and the amount of stringers used were less than half of those that were used on the fuselage. The bulkheads in this case, which were spaced five inches apart, were cut out using the same techniques as the fuselage bulkheads but instead of securing them together, they were done separately since the tail boom tapers as it goes back. In order to do this, scale drawings were printed and glued to five pieces of light ply. Due to the tapering of the tail boom, the band saw was angled to ensure proper cutting and fitting of the stringers and balsa skin.

The horizontal and vertical stabilizers were constructed out of ¼ in. and 1/8 in. balsa stringers according to detailed drawings. The stabilizers were basically plat plates configured to minimize weight. Once the tail boom and stabilizers were constructed, they were coated with Monokote and assembled together with epoxy. The stabilizers would simply sit atop the tail boom. The empennage was then attached to the fuselage with epoxy by bringing the last bulkhead of the fuselage together with the beginning bulkhead of the empennage.

6.5 Landing Gear

The landing gear, being that of tricycle configuration similar to the one used last year was not very difficult to manufacture. As a matter of fact, the front landing gear used would be the same off the shelf landing gear used last year since it met our requirements. The main landing gear was designed and tested using IDEAS finite element analysis with several different materials. The material chosen was aluminum alloy 6061-T6 since it met our requirements with minimum dimensions and weights. The landing gear was ordered to be manufactured by a model aircraft landing gear manufacturing company. Once the fuselage was built the landing gear was attached. The total cost of manufacturing for the landing gear was $25.00.

6.6 Power Components

The main components of the power system, which are the motors and batteries, were assembled after the wing, fuselage, empennage, and landing gear were finished. The motors were attached in the following manner. Once the wing was built, two cuts were made in the positions where the motors would be attached. This is at 11.5 in. away from the root chord of the wing underneath the wing and where the wing spars are located. Two pieces of light ply were then attached vertically from the spars. Motor mounts were attached to these two pieces of ply and the engines were secured to these motor mounts. The batteries for the motors were bought assembled and shrink-wrapped by the same manufacturer. The batteries were assembled as mentioned before, in series and side by side. Two-¼ in. dowels were then secured to the sides for easy insertion into the fuselage battery compartments.

Once the power components were attached and assembled, the wing was attached to the fuselage by using four nylon screws and locking helical inserts supported in two light aircraft ply pieces. All the final power component connections were made final and the aircraft was ready for testing.

Aircraft Spruce and Specialty Company

Product #	**Description**	**Unit Price**	Quantity	Total
01-10200	Styrofoam (Blue) 4"x24"x48"	$26.40	4	$105.60

TNR Technical: The Battery Store

Product #	**Description**	**Unit Price**	Quantity	Total
Sanyo N-1700	Sanyo N-1700 1x18, soldered, shrink-wrapped batteries	$86.00	4	$344.00

Hobby Lobby International

Product #	**Description**	**Unit Price**	Quantity	Total
HLFK394	Kavan Motor Mounts	$7.90	4	$31.60

Balsa USA

Product #	**Description**	**Unit Price**	Quantity	Total
81	42 Inch Balsa Sheets (1/16)*3	$0.54	**5**	**$2.70**
82	42 Inch Balsa Sheets (3/32)*3	0.65	**5**	**$3.25**

Tower Hobbies

Product #	**Description**	**Unit Price**	Quantity	Total
LX1584	APC 13x8 Prop	$4.69	4	$18.76
LXH354	Futaba Dry 4 Cell Receiver	$4.99	2	$9.98
FUTM4382	Futaba SWH12 Mini Switch	$8.59	4	$34.36
LX3559	Astroflight Cobalt 40G Motor	$194.99	2	$389.98
LXTF93	Astroflight 204D Speed Controller	$109.99	4	$439.96
LL4031	Great Planes Pro CA – Glue Thin	$6.49	2	$12.98
LL4039	Great Planes Pro CA – Glue Thick	$6.49	2	$12.98
LL4177	Tower Hobbies 6-Minute Epoxy	$7.99	1	$7.99
LL4179	Tower Hobbies 30-Minute Epoxy	$7.99	1	$7.99
LLJC80	Tower Hobbies #11 Blades (100)	$7.99	1	$7.99
LL4061	Great Planes Easy-Touch Sandpaper 150 Grit	$4.99	2	$9.98
LL4062	Great Planed Easy-Touch Sandpaper 180 Grit	$4.99	2	$9.98

Aerospace Composites

Product #	Description	Unit Price	Quantity	Total
WF-19	5.6 oz. Woven carbon plain weave 50" wide	$32.00	1	**$32.00**
WF-16	3.7 oz. S-Glass 27"	$7.50	3	**$22.50**
WF-15a	1.4 oz. S-Glass 50"	$12.00	2	**$24.00**
V-11	Mylar Bagging Field 60" wide per yard	$1.50	3	**$4.50**
V-12E	Vac bagging Sealant tape 250 temp 25 ft.	$6.50	1	**$6.50**
V-16	0.014 Dupont Mylar 6ft	$11.75	5	**$58.75**
V-22	Breather Cloth 60"wide	$3.00	1	**$3.00**
V-25A	China Bristle Epoxy brush	$1.00	1	**$1.00**
V-26	Epoxy spreader 5" wide	$1.00	1	**$1.00**
V-28	3" Epoxy Roller Kit	$10.00	1	**$10.00**
E80-R	Red Epoxy Pigment Paste	$6.00	1	**$6.00**
E60-03	EZ-Lam Epoxy 1-1/2 Qt. Kit	$38.00	1	**$38.00**

TOTAL	**$1657.33**

Figure 6.2 Cost Analysis of Manufacturing

Task Name	Start
Manufacturing	Thu 1/17/02
Wing	Fri 1/18/02
Aifoil Sections	Fri 1/18/02
Foam Sections	Mon 1/28/02
Spars	Mon 2/11/02
Carbon Fiber Coating	Sat 2/23/02
Motor Attachment	Sat 3/2/02
Fuselage	Mon 2/11/02
Bulkheads	Mon 2/11/02
Assembly	Thu 2/21/02
Attachment parts for Wing and Landing gear	Tue 2/26/02
Minor Power components	Wed 2/27/02
Balsa Sheething	Thu 2/28/02
Monokote	Fri 3/1/02
Nose Cone Manufacturing	Mon 2/11/02
Nose cone attachment	Fri 3/1/02
Landing Gear Attachment	Fri 3/1/02
Empennage	Mon 2/11/02
Tail Boom Bulkheads	Mon 2/11/02
Assembly	Mon 2/18/02
Balsa Sheething	Mon 2/25/02
Monokote	Thu 2/28/02
Attachment to Fuselage	Fri 3/1/02
Vertical and Horizontal stabilizer construction	Fri 1/18/02
Final connections	Sat 3/2/02

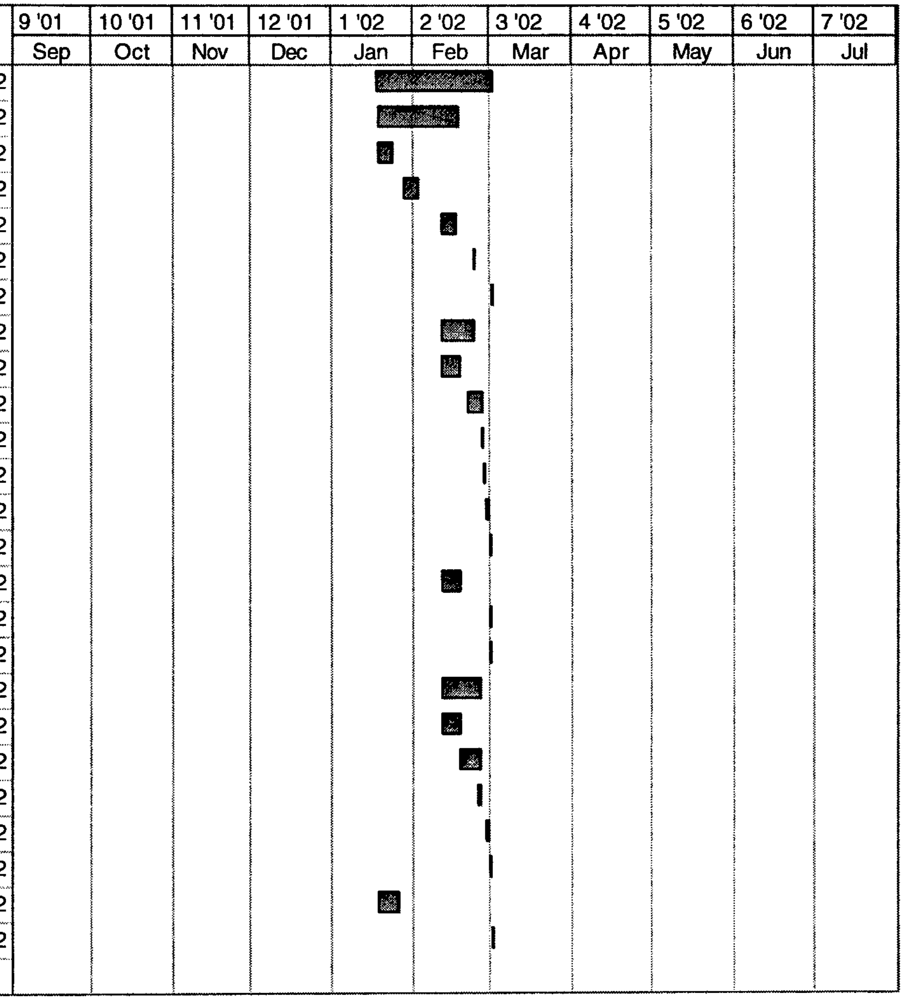


Figure 6.3 Manufacturing Timeline

7. References

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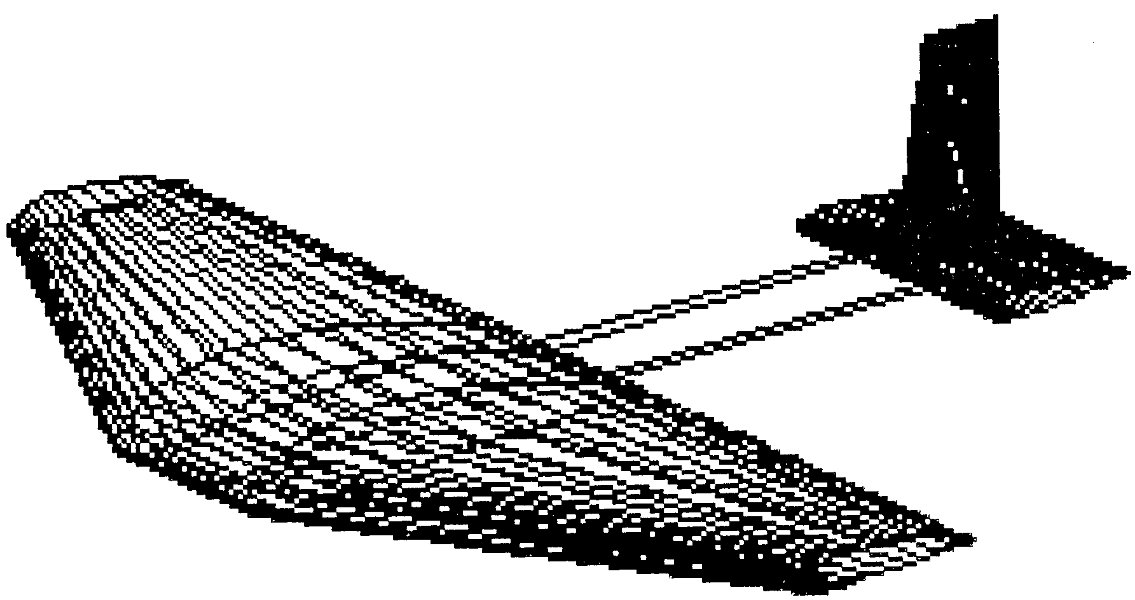
2001/2002 AIAA Foundation Cessna/ONR

Student Design Build Fly Competition

Design Report – Proposal Phase

The University of Alabama

March 2002



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1. EXECUTIVE SUMMARY

The senior design class of the University of Alabama chose its project to be the Cessna/ONR Design Build Fly Competition. The objectives of the competition were to fly the course in both loaded and unloaded conditions carrying a payload of between 10 to 24 softballs. A team was then organized that included the members of the senior class along with interested underclassmen. Starting from the initial investigation of nine of possible configurations, three were selected for further investigation and later only one was selected for final design. The final design selected was that of a flying wing planform; it would later be modified to include a tail for stability purposes.

For the conceptual design phase, textbooks, documents, and journal articles were examined to determine the advantages and disadvantages of the nine possible designs based on pre-selected criteria. During the preliminary design phase, the three designs that rated best based on the criteria were then investigated in further detail. In the detailed design phase all efforts were concentrated on finalizing the specifics of the chosen design, the flying wing.

1.1 Conceptual Design

To facilitate the investigation of the initial of possible design planform concepts, each of the nine students in the senior design class was assigned a concept to research. These design concepts were ring-wing, flying wing, lifting body, C-wing, blended wing body, braced/typical wing, biplane, pusher canard, and joined wing. Each was evaluated based upon a common set of judging criteria comprised of the following: structural weight, high speed, internal volume to external area ratio, maneuverability and stability, ease of manufacturing, Lift to Drag ($^L/_D$) ratio, payload access and transport, cost and cost penalties, durability, and slow landing speed. Each of the criteria was then rated on a scale of +2 to –2 by the researcher based on its advantages or disadvantages when compared to the conventional or typical aircraft configuration.

1.2 Preliminary Design

After comparing each of the initial concepts, three were selected by the team to be pursued in further detail during the preliminary design phase. These three concepts were the braced/typical wing, the joined wing, and the flying wing. Groups comprised of three senior class members each were then formed and underclassmen were enlisted to assist in the analysis. Computer programs such as WINGBODY, JKVLM, VLMPC, MATLAB, IDRAG, and Excel were all used as design tools during this phase. Each group was responsible for progressing the design of each of their respective aircraft as much as possible so that the three could later be compared.

1.3 Detailed Design

With three partially complete and distinctly different designs under consideration, it became necessary to down select to only one final design. Each group gave presentations briefing the entire team on the feasibility of their design. The team members then wrote individual selection reports and submitted them to the team advisor. The flying wing configuration was selected by those reports as the final design. The efforts of the entire team were then dedicated to the finalization of that design. The computer programs

used during the preliminary design were again used as tools for the detail design. Tests were also performed to gather further information to help with the design process.

2. Management Summary

At the beginning of the fall semester of 2001, the senior design class of the University of Alabama selected as their senior design project this year's Cessna/ONR Student Design/Build/Fly Competition. The team consisted of nine seniors, all majoring is aerospace engineering, and nine interested underclassmen. The senior design class's professor, Dr. Zeiler, was also the team's advisor.

2.1 Architecture of the Design Team

The organization of the design team changed during each phase of the design. During the conceptual design phase, each member was independently researching. In the preliminary phase, however, the team was broken into three design groups. This breakdown appears in Table 2.1.1.

Table 2.1.1 Preliminary Design Phase Team Organization

Team leader: Ryan Davidson Treasurer: Adam Sampley		
Joined Wing Leader: Tim Wingenter	**Flying Wing** Leader: Chris Lane	**Typical Wing** Leader: Adam Sampley
Ashley Moore Hyungkeyn Yi	Yohei Yamaoka Ryan Davidson	Mizuho Aoyagi Mark Thornblom Sam Carter

In the detailed design phase the team was re-organized into four disciplinary groups. Each senior team member was allowed to suggest the final team organization to the advisor. To facilitate the division of the team members among different groups, all members was asked to evaluate themselves on a scale of 1 to 5 as having proficiency in several different areas. The skill areas and individual member ratings appear in Table 2.1.2. The advisor then organized the team as appears in Table 2.1.3 after taking into consideration the suggestions of the team members.

Table 2.1.2 Skills Matrix of Senior Team Members

	Adam	Ashley	Tim	Mizuho	Yohei	Mark	Ryan	HK	Chris
Programming	4	1	4	2	3	3	1	2	4
Writing	4	4	3	1	1	4	4	1	2
Wood Working Skills	3	1	4	1	3	4	4	4	1
Fiberglass Skills	1	1	1	1	4	5	1	1	1
Machining & Tools	3	2	4	1	4	5	4	4	3
Fine Motor Skills	4	3	4	1	4	4	3	3	4
"Leader"	1	3	3	1	1	4	3	2	4

Table 2.1.3 Detail Design Phase Team Organization

Team leader: Ryan Davidson Treasurer: Adam Sampley			
Layout, Weights, & Inertia	**Aerodynamics, Performance, & Controls**	**Structures & Manufacturing**	**Systems & Power**
Ashley Moore Mizuho Aoyagi Yohei Yamaoka	Adam Sampley Chris Lane Mizuho Aoyagi Tim Wingenter	Chris Lane Mark Thornblom Yohei Yamaoka	Ashley Moore Hyungkeyn Yi Mark Thornblom Ryan Davidson

Each group was allowed to perform its own testing as required by coordinating with the team leader. Many of the underclassmen participated routinely in the tests that needed to be performed. They were also allowed to support any of the four groups as required. Table 2.1.4 illustrates the personal assignments and involvement in team functions.

Table 2.1.4 Personal Involvements

	Mizuho Aoyagi	Ryan Davidson	Christopher Lane	Ashley Moore	Adam Sampley	Mark Thornbolm	Tim Wingenter	Yohei Yamaoka	Hyungkeyn Yi	Underclassmen
Conceptual Design										
Ring Wing	0	0	0	0	5	0	0	0	0	0
Flying Wing	0	5	0	0	0	0	0	0	0	0
Lifting Body	0	0	0	0	0	5	0	0	0	0
C - Wing	0	0	0	0	0	0	5	0	0	0
Blended Wing Body	0	0	0	0	0	0	0	5	0	0
Strut-Braced Wing	5	0	0	0	0	0	0	0	0	0
Biplane	0	0	0	0	0	0	0	0	5	0
Canard/Pusher	0	0	5	0	0	0	0	0	0	0
Joined Wing	0	0	0	5	0	0	0	0	0	0
Preliminary Design										
Typical Wing	5	0	0	0	5	5	0	0	0	1
Joined Wing	0	0	0	5	0	0	5	0	5	0
Flying Wing	0	5	5	0	0	0	0	5	0	0
Detailed Design										
Aerodynamics	5	0	5	0	5	0	5	0	0	0
Performance & Controls	5	0	4	0	5	0	5	0	0	0
Layout	3	0	0	1	0	0	0	5	0	0*
Weights & Inertia	3	0	0	1	0	0	0	5	0	0*
Structures	0	0	5	0	0	3	0	5	0	0
Manufacturing Planning	0	0	3	0	0	5	0	3	0	0
Systems	0	3	0	3	0	3	0	0	3	0*
Power	0	5	0	0	0	3	0	0	4	0*
Manufacturing	0*	0*	0*	0*	0*	0*	0*	0*	0*	0*
Documentation of Design										
Journal	5	0	0	1	2	0	0	0	0	0
Letter of Intent	0	0	0	5	0	0	0	0	0	0
Final Report	3	2	3	3	4	3	3	3	1	0
Addendum Report	0*	0*	0*	0*	0*	0*	0*	0*	0*	0
Drafting Package	3	0	1	0	2	0	0	4	0	5

* No work has been done in these areas at this time, but upcoming work has been planned.

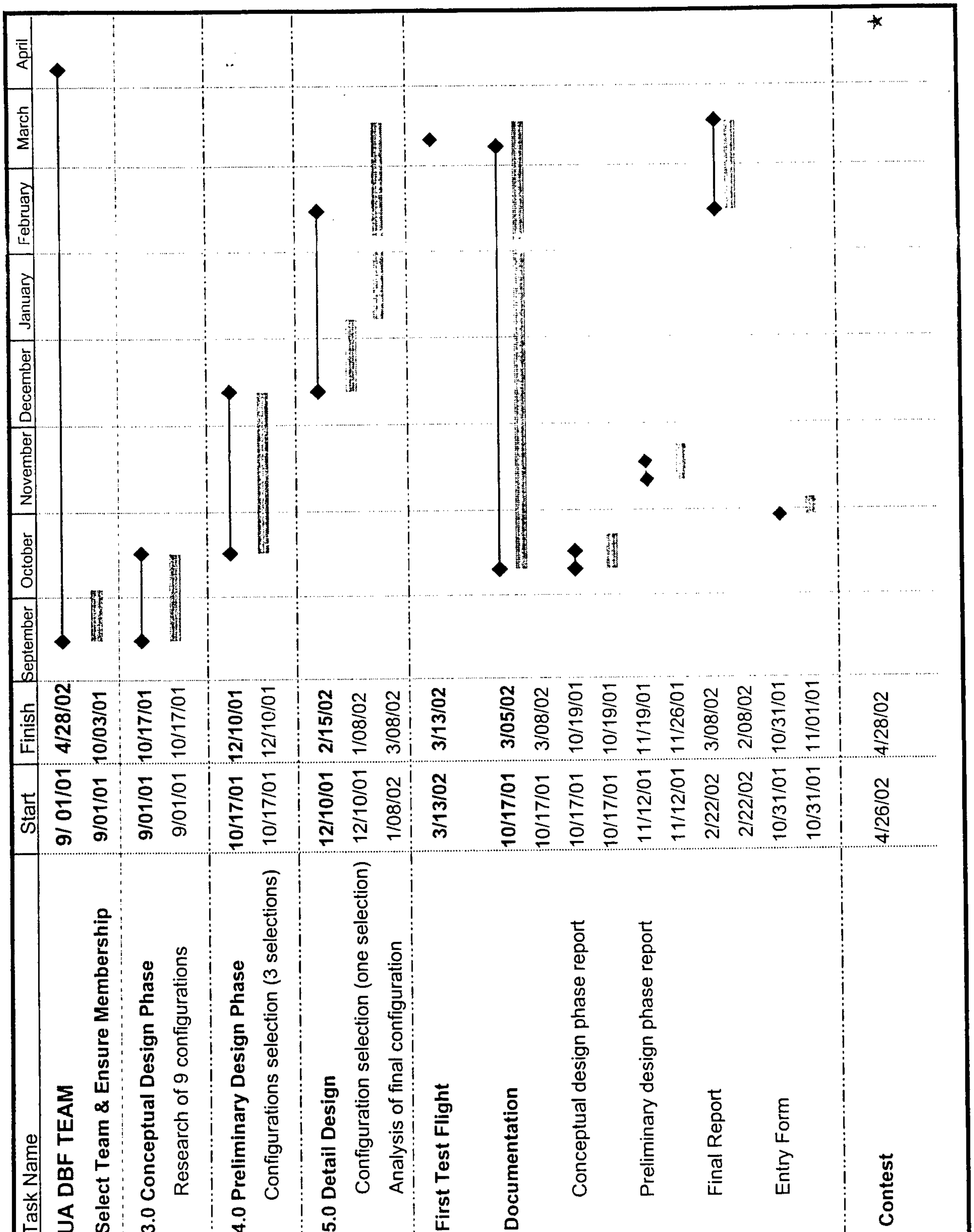


Figure 2.2 project Schedule. This schedule illustrates the project milestones and time periods designated for each design phase. The black line represent the planned schedule while the blue bars show the events actually occurred. The contest data is marked as star sign.

3. Conceptual Design

In the conceptual design phase, the team members analyzed nine different design concepts. They then rated those concepts based on the criteria the team had previously selected. The three concepts, which rated the best, would then be selected for the second phase of the design. No hard calculations were performed during this phase. Each design was evaluated on a somewhat subjective basis and compared to the typical aircraft on each of the criteria.

3.1 Design Criteria and Concept Performance

The team had decided to rate the concepts base upon the following criteria: structural weight, high speed, internal volume to external area ratio, maneuverability and stability, ease of manufacturing, Lift to Drag ratio ($^L/_D$), payload access and transport, cost and cost penalties, durability, and slow landing speed. Each concept was then assigned a value of +2 to –2 for each criterion when compared to the performance of a typical aircraft.

3.1.1 Ring-Wing

A ring-wing planform is one in which the wings are looped over and connected together to form a closed section. It is considered a very radical concept and no airplanes have ever been flown successfully that used this concept. The ring-wing concept was found to have a large disadvantage when compared to the typical aircraft. It's rating was an overall –8. The advanced nature of the concept also meant that there was a lack of information when researching its feasibility. In the end, this design would be rejected.

Ring-Wing structure forms a closed loop thus adding to the rigidity of the structure and reducing the weight to surface area ratio of the wing. It received a slight plus for light structural weight. No significant difference could be noted that would affect the speed of the design over that of a conventional aircraft except for possible engine mounting issues, thus it scored lower for high speed. Since the fuselage of the ring wing and typical aircraft are very similar, this concept had no volume to area ratio advantages. The curved surface of the wing could increase difficulty it positioning the controls surfaces thus impeding the maneuverability and stability of the aircraft. The unconventional curvature also greatly increases the manufacturing difficulty. $^L/_D$ ratio was the main benefit of the design. The unusual wing shape could also lead to complicated internal structure thus complicating payload access. The circular ring could also greatly increase the overall height of the design thus incurring stiff cost penalties. The durability of the aircraft could also be reduced because of the difficulties involved in the repair of the complicated structure. The curved wing could also cause landing gear issues and thus affect its landing speed.

3.1.2 Flying Wing

By definition, a flying wing is an aircraft that has no other components except the main lifting surface. Taken literally, this is an absurd configuration for an aircraft since it does not allow for engines, or other components to break the clean line of the wing. If the definition is relaxed a bit to allow propellers, controls and other useful components, it becomes a viable concept.

When compared with the typical aircraft configuration in many different areas, the flying wing shows that it is superior in some categories, equal in some categories, and inferior in other categories. In the category of structural weight the flying wing is much better than the typical aircraft. With a typical aircraft the lift created by the wing and the weight of the fuselage and cargo must be connected with heavy structures. With a flying wing the loads are already connected due to the fact that the cargo is carried in the wing, and there is no fuselage. In the category of high speed, the flying wing does marginally better than the typical aircraft due to a lower parasite drag than the typical aircraft since there are fewer components in the air stream. When comparing the ratio of internal volume to external area, the flying wing does much better than the typical aircraft. The typical aircraft has a large external area compared to the internal volume due to the tail, external pylons, fuselage, wings, etc. The flying wing however, has a small external area compared to its internal volume. This is due to its lack of a tail, fuselage and other external components. In the category of stability, the flying wing is greatly inferior to the typical aircraft. Tail surfaces give the typical configuration a large amount of stability. The fact that a flying wing has no tail necessitates that its stability come from other places. The main source of stability in a flying wing is the sweep and twist of the wing. Sweep moves the aerodynamic center of the aircraft aft, while wing twist allows the tips to create negative lift, and act much like a horizontal tail. In the category of ease of manufacture, the typical aircraft is marginally better. This is due to the fact that the flying wing must have precise sweep and twist in the wing. If this is not done precisely, the aircraft will not fly properly. The typical aircraft is not as sensitive to imprecision in construction as the flying wing is due to the greater level of stability given by the tail. The $^{L}/_{D}$ ratio of the flying wing is much better than the typical aircraft due the low drag seen by the flying wing. The maximum lift coefficient produced by the airfoils generally used on flying wings is not as great as the maximum lift coefficients produced by airfoils generally used on typical aircraft. Maximum lift however, is not used for most flight conditions, so this is less of a concern than the low drag. In the category of payload access, the two configurations were equal because the payload would be accessed in much the same manner for both configurations. In the area of rated aircraft cost efficiency, the flying wing did marginally better than the typical aircraft. This is due to the fact that there are fewer parts. For example there is no horizontal tail on a flying wing. Also, the total length of the aircraft will be shorter. The flying wing is more durable than the typical configuration because there are fewer components to fail. The final area of comparison was slow landing speed. The typical aircraft did better than the flying wing since flying wings generally do not have flaps to aid in landing. This is because there is no way to counter the increased pitching moment caused by use of flaps.

3.1.3 Lifting Body

The lifting body configuration is one in which no formal wings are used in producing lift for the aircraft. Instead, the body generates all the lift necessary for flight. If any protruding horizontal surface exists, it is purely for control purposes. Some examples of lifting body aircraft are the Space Shuttle, NASA's Venture Star and NASA Langley's X-43. Lifting bodies are becoming more and more popular in the design of supersonic and hypersonic vehicles. However, for subsonic applications, the lifting body

has many disadvantages over the typical configuration. The lifting body scored a –7 when compared to a typical configuration. In addition, most sources used for research of the lifting body referred to supersonic aircraft and were invalid for this investigation.

In general, the main advantage of a lifting body is that without wings, one gets a small weight savings. However, much of the weight savings created by the lack of wings is lost in fuselage reinforcement, so the advantage is slight. In addition, manufacturing a lifting body would be easier than a typical, mostly because the aircraft could be constructed in one piece: the main fuselage, rather than multiple pieces, such as each individual wing, tail sections and main fuselage. Finally, the lifting body would be capable of easy payload access. One could simply remove a single top panel and have access to the storage compartments.

Some major drawbacks for a lifting body include low speed handling, particularly at take-off and landing, stability, and controllability. Because the vast majority of lifting body aircraft are optimized for high speeds, the planforms result in poor low speed characteristics. Also, producing a lifting body aircraft with good stability is difficult to come by. Research suggests that this problem has occurred frequently in lifting body design. Finally, without the presence of wings one is unable to easily produce roll maneuvers that traditional ailerons would provide, thus increasing the difficulty of producing an aircraft with good controllability. Although the lifting body contains several advantages to that of the typical configuration, there are more disadvantages that lead to the conclusion that the lifting body configuration would not be considered for further conceptual design.

3.1.4 C-Wing

A C-wing is a configuration in which a horizontal lifting surface is attached to the top of a winglet, which is attached to the end of a conventional wing, to form what looks like the letter "C." The idea of this type of configuration has only been looked at in the past five to ten years, so research pertaining to this type of wing is still being done. As a result, there are still a lot of uncertainties associated with the C-wing. However, there are some known advantages and disadvantages for this configuration. The big advantage for this type of configuration is that the span of the wing can be reduced while keeping the lifting capabilities the same. This advantage does not really apply to this project because the length of the wings would not be restricted by runways, taxiway, and parking places as they would be with very large transport aircraft. Another advantage is the reduction of the induced drag, which primarily is due to the winglets reducing the vortex effects on the wing tips. This reduction of drag, while keeping the same lift, would result in a higher $^{L}/_{D}$. Also, since the drag would be reduced, the aircraft would be capable of flying faster. Other advantages occur as a result of configuring the aircraft in certain ways. For example, the wings could be swept back to make the upper horizontal surfaces in a position where they could act as the horizontal stabilizer. By doing this, the entire tail could be removed, and the structural weight of the airplane would be reduced. This would also cause a lower internal volume to external area ratio. One of the uncertainties for this type of aircraft is its maneuverability and stability characteristics. In theory, it could be designed to be just as controllable as the typical, or maybe even better, but there might

be some unknowns, which could cause some problems. There are some other parameters, which were neither beneficial nor disadvantageous for a C-wing. Easy payload access, low cost, and slow landing speed could not be differentiated between the C-wing and typical configurations. The disadvantages of this type of configuration would be its difficulty to manufacture, contest cost penalties (two vertical surfaces), and its durability and fatigue uncertainties. The aircraft's rating as a whole came out to be only slightly better than that of a typical aircraft, but with the uncertainties mentioned, the typical aircraft was still a more logical choice.

3.1.5 Blended Wing Body

A blended wing body aircraft is one in which there is no definite distinction between where the wing ends and where the fuselage begins. The biggest advantage of Blended Wing Body (BWB) configuration is that it can achieve very high $^L/_D$ because of its highly integrated shape. Since such a configuration does not have a horizontal tail or a typical fuselage, its total drag is greatly decreased. Also, the body section of BWB can produce a small amount of lift, and according to well-reasoned research, the $^L/_D$ ratio could be increased by up to 21% compared to the typical aircraft. However, the blended shape turns out to be a disadvantage in that it complicates the manufacturing process. The wing must be not only correctly twisted but also smoothly blended into the body. A large amount of skill and care are necessary for successful construction. Another disadvantage is poor stability. Due to the nonexistence of a horizontal tail, it is very difficult to maintain pitch stability. To be stable in pitch, the wing has to be swept back and twisted so that the aerodynamic center is aft of the center of gravity and the pitching moment coefficient becomes negative. However, to achieve a stable aircraft requires an incredible amount of design. A decrease in structural weight is one of the advantages of blended wing body. It does not have a typical tube-fuselage or a horizontal tail and generally it has 30% fewer parts compared to typical aircraft. Unlike the flying wing configuration, BWB has relatively large internal volume, which helps to carry a large amount of cargo and makes it easy to access. The distinction between the flying wing and the blended wing body however is not a very big one.

According to the available data, a successfully designed BWB is much better than typical aircraft. However, it is doubtful that the team would be able to design, build and fly a blended wing body aircraft because of the level of difficulty involved.

3.1.6 Strut-Braced/Typical Wing

The strut-braced wing uses a strut to attach the wing to the fuselage. The strut allows a lighter overall wing weight and a drag benefit through decreased wing thickness. While researching this configuration it was noted that the development of this concept would require a solution to interference drag and compression-buckling problems through the application of advanced structural and aerodynamic computer tools not available to the design-build-fly team.

A wing braced with a strut will have reduced bending moment at the wing root, allowing the wing weight to be reduced for any given thickness. Reduced thickness provides drag reduction by decreasing the amount of separation and reducing the wetted area. These drag effects allow the wing to not be swept

for an increased region of natural laminar flow and further wing structural weight savings compared to a typical aircraft.

In considering the ease of manufacture and assuming there is enough information to manufacture the typical aircraft, a braced wing aircraft would not be much more difficult to manufacture. However, the strut material, thickness, and placement could be somewhat difficult to determine due to the drag effect and compression buckling problems. Therefore, manufacturing the strut might need extra care and add complication to both the design and manufacturing processes.

One of the biggest advantages of a strut-braced wing configuration is that the strut alleviates the bending moment to the wing; this allows the reduction in wing thickness and increase span length. The longer span length makes the aspect ratio higher than that of a typical wing. This high aspect ratio helps reduce $^{L}/_{D}$.

The only difference between a strut-braced wing and typical wing is whether or not a strut is used to support the wing. The strut-braced wing is lighter, which reduces the empty weight of the aircraft. However, the span of the strut-braced wing is longer than typical wing. Also there would be some difficulties of manufacturing the strut. It is possible that there could be cost penalties due to the increased span. If the strut is manufactured properly, proper thickness, proper placement, proper material and so on, it helps to increase durability of the wing. Otherwise, the strut might break and compromise the entire wing. The complication of the strut might also increase repair time of the wing should the strut fail.

According to the information obtained in the conceptual design phase, the strut-braced wing configuration is slightly better than typical wing configuration. However, the advantages and disadvantages obtained are made from general observation. This strut-braced wing configuration was chosen for analysis in preliminary design phase as a modification of the typical wing configuration.

3.1.7 Biplane

In order to compare the biplane configuration with the typical configuration, it is necessary to review the birth of the biplane. The Wright Brothers in a biplane performed the first manned flight. At that time however, no lightweight engines were available that could produce a large enough amount of thrust to propel a mono-wing configuration. Aircraft structural theory had also not been developed yet. Thus the biplane was the only choice for aerodynamic and structural reasons known at the time. However as the technology of structures, engines, and aerodynamics improved, the biplane began to disappear and was out of use by the 1940s.

Obviously, a biplane has heavier structure than monoplane. This is due to the extra weight of multiple wings and interconnecting struts even though the biplane configuration could eliminate some of the spars inside the wing structure. Increased surface area and non-aerodynamic struts between wings also caused more drag for a monoplane. This higher drag coupled with increased wing surface area caused lower flight speeds. The only advantage of the biplane over a monoplane is its slow landing speed due to its overall slower flight operations. Modern aerodynamic technology allows for slower landings by utilizing

surfaces such as air breaks, flaps, and spoilers. The biplane was found to be inferior to the modern typical mono-winged aircraft in all respects. It was therefore not recommended for further consideration.

3.1.8 Canard Pusher

A canard/pusher aircraft is visually similar to a conventional aircraft except what would normally be referred to as the 'tail' is in front of the wing and called a 'canard.' Although an aircraft with a canard does not necessarily have to have a 'pusher' style engine (instead of the more traditional 'tractor' style engine), the two deviations from a typical aircraft configuration usually come together because, otherwise, the canard would be constantly flying in the propwash of the tractor engine. A pusher engine is exactly what it sounds like – it pushes the aircraft through the air instead of dragging it along like a tractor engine.

At first glance, the overall rating seems to eliminate the canard/pusher immediately. However, the canard/pusher loses many points because it's much easier to design a very bad canard than a very bad conventional aircraft's tail. A canard that is slightly too small or large can result in erratic and unpredictable aircraft behavior. If the canard/pusher design is employed correctly, the aircraft can outperform conventional aircraft designs in many respects (see the Rutan designs).

The usual reason a canard/pusher configuration is considered as a design option is the potential high cruise efficiency offered by both lifting surfaces acting in the same direction. This is one major advantage of the canard/pusher. The second argument put forth in favor of canard aircraft is that they can be made stall-proof. They can also be made stall-divergent. The key here is that the canard must stall before the wing. If the canard stalls first, this will cause the nose to dip, reducing the angle of attack of both the wing and the canard until the canard unstalls. The wing never stalls in this arrangement, so the aircraft never loses roll control. If the wing stalls first, the aircraft will immediately attempt to turn itself into an aft-tailed configuration, which is very bad. Great care must be taken to ensure that the canard stalls first and that the theoretical stability/maneuverability benefits of the canard/pusher configurations are realized. There is no reason the structural weight of the canard/pusher would be any greater than the structural weight of a conventional aircraft. Some argument can be made here that the wing planform for a canard/pusher design will have to be larger than the planform of a conventional aircraft causing the structural weight to increase because the canard 'spoils' some of the airflow over the wing and kills valuable lift. However, because the canard now acts in the same direction as the wing, the canard lift should help to restore some of that lost lift. The canard/pusher would offer no more internal volume per external area than a conventional aircraft.

The major disadvantages of a canard/pusher design are low cruising speed and high landing speed. The main problem here is the generation of thrust. The blades on pusher engines run the risk of striking the ground during take-off and landing. Therefore, they are typically smaller than a similar tractor engine and will develop less thrust for the same amount of engine power. Using more powerful engines that spin the blades faster can eliminate this problem. However, this is not a good alternative for our particular design because we are constrained by the type of engine we can use and the amount of power it can consume. Canard/pusher aircraft have high landing speeds because canards have difficulty developing

lift at low speeds and, as a result, they tend to have high stall speeds. Therefore, the canard/pusher aircraft generally have long landing and take-off runs because of the high speeds needed to avoid stalling the aircraft so close to the ground. There are several other minor disadvantages of the canard/pusher design. The payload will now only be accessible from the top of the aircraft (the engine is now located in the back and the canard is now located in the front). Durability will be a problem area because the prop blades must now pass in and out of the wing wake, causing considerable vibration and external noise on the blades.

3.1.9 Joined Wing

The joined wing configuration employs tandem wings that are arranged to form a diamond shape in both the front and plan views. This is accomplished by having the front wing swept back, and the horizontal tail (rear wing) mounted to the top of the vertical tail. The horizontal tail is then extended down to attach to the front wing. The rear wing is typically attached at a location between 50% and 70% of the front wing's semispan, and behind the front wing's maximum thickness point.

When comparing the joined wing configuration to the typical aircraft configuration, the advantages outweigh the disadvantages. The joined wing is much more durable due to the fact that the rear wing serves as a strut for the front wing. This results in a higher stiffness and enhanced structural strength. In addition, the joined wing has a higher L/D compared to that of the typical airplane. This configuration decreases cruise drag, increases the amount of direct lift generated, and minimizes trim drag. From experimental data, it was found that the joined wing aircraft cold typically achieve higher speeds than the typical airplane. The structural weight is reduced by 12% to 30% by using this configuration instead of the typical configuration. The joined wing configuration resists stall, has built-in side-force control capabilities, and has a higher trimmed maximum lift coefficient. This along with the use of control surfaces makes this configuration have higher maneuverability and stability.

There were a few areas of comparison in which the joined wing had no advantages or disadvantages. These include cost efficiency, manufacturing, and payload access. The cost of manufacturing the joined wing is very similar to that of the typical plane, because the total amount of materials needed for each is quite close. The joined wing has a larger horizontal tail, but the wings are thinner and require fewer internal supports. Although the joined wing's horizontal tail is larger than that of the typical airplane, and includes substantial anhedral, fewer internal stiffeners are needed, as already mentioned. By reducing the amount of internal stiffeners needed, the manufacturing of either type of configuration is nearly the same. Accessing the payload from the joined wing airplane will be just as easy as accessing the payload of a typical airplane. Lastly, the joined wing has the same capabilities of the typical plane with regards to slow landing.

The ratio of internal volume to external area was the only aspect compared that received a negative score. The same fuselage can be used for either configuration, which means that the two planes will have very similar internal volumes. The joined wing plane though, has additional external area due to the larger rear wing. This leads to a lower score when evaluating internal volume v. external area.

To conclude, the joined wing aircraft seems to be quite superior to the typical aircraft when comparing these certain airplane qualities.

3.2 Conceptual Design Concept Comparison

Each design concept was assigned values for the evaluation criteria as appears in Table 3.2.1, which can be found on the following page. Using this comparison, the team was able to select the three most promising designs for continued analysis during the preliminary design phase.

Table 3.2.1 Preliminary Criteria Results Summary

Criteria	Ring Wing	Flying Wing	Lifting Body	C-Wing	Blended Wing Body	Strut-Braced Wing	Biplane	Canard/ Pusher	Joined Wing
Light Structural Weight	+1	+2	+1	+1	+1	+2	-1	0	+2
High Speed	-1	+1	0	+1	+1	+1	-1	-2	+1
Internal Area vs. External Volume	0	+2	-2	+1	+2	0	0	0	-1
Maneuverability/Stability	-2	-2	-2	0	-2	0	-1	+1	+1
Ease of Manufacturing	-2	0	-1	-1	-2	-1	-1	0	0
High Lift to Drag Ratio	+2	+2	+2	+2	+2	+2	-1	-1	+2
Easy Payload Access	-1	0	0	0	0	0	-1	-1	0
Cost Efficiency	-1	+1	+1	0	-1	0	-1	-1	0
Durability	-2	+1	+1	-1	+1	0	-1	-1	+2
Slow Landing Speed	-2	-1	-2	0	-2	-1	+2	-2	0
Total	-8	+6	-2	+3	0	+3	-6	-7	+7

4. Preliminary Design

The three best-rated concepts from the conceptual design phase were each developed further in the preliminary design phase. These concepts were the strut-braced/typical, the joined wing, and the flying wing. A few weeks were set aside to the further development of each of these concepts. The objective was to determine which of them would best meet the requirements of the design competition and represent a feasible planform to develop, given the skills of design and manufacturing of the design team members. Each concept was to be developed to support the maximum number of twenty-four softballs allowed by the competition rules. Each group was to develop their concept independently of the others. Tools used for the design in this phase included, VLM, JKVLM, VLMPC, WINGBODY, IDRAG, Excel, and MATLAB.

4.1 Programs Used

WINGBODY is a FOTRAN based aerodynamic modeling code written in the late 1960's, but modified at the University of Alabama in the late 1990's. This code allows the user to define a model of an aircraft, in 3-D coordinates, and calculate the qualities that a real aircraft might have. The aircraft model can be broken up into different sections such as wing, body, and tail to allow a more accurate model of the real aircraft. Along with the aircraft sections, the Mach number and angle of attack at which the calculations are to be done must be defined. The program uses the Vortex Panel method to calculate pressure coefficients and control points along the defined surfaces of the wing and body at almost any angle of attack. The program also outputs lift coefficients, drag coefficients, friction coefficients, pitching moment coefficients, and center of pressures for each of the defined sections as well as for the entire model. Although the program can handle both supersonic and subsonic velocities, the basis of the program was designed to handle supersonic aircraft. Therefore, some discretion must be used in analyzing subsonic aircraft. TECPLOT is another program that can be used with the WINGBODY outputs to create plots of the aircraft and pressure distribution over the configuration.

IDRAG is a FORTRAN based program which uses a discrete vortex method based on the Kutta-Joukowski theorem and the Biot-Savart law to analyze flow properties of a given configuration. The program can be used in two different modes, design mode and analysis mode. In design mode, the aircraft geometry is defined, in 3-D space, by dividing it into sections and defining the corner points. The center of gravity point and desired lift coefficient is also required. The output of the design mode is a lift distribution over the aircraft sections to minimize induced drag at the given lift coefficient. It is then left to the user to find how to orient the aircraft to produce the given lift distribution using twist, camber, etc. For the analysis mode, the aircraft is defined the same way as in design mode, but now the load distribution is specified. The outputs for analysis mode are the actual lift and moment coefficients, the induced drag coefficient, and the span efficiency factor.

JKayVLM is a FORTRAN based computer program which uses the vortex-lattice method to calculate subsonic stability and control derivatives of a given aircraft configuration. The aircraft is defined through inputting a maximum of five trapezoidal sections, in 3-D space, which may include twist and dihedral but

no camber or thickness. Each of the sections is divided automatically into 40 panels, 8 streamwise and 5 spanwise, for the vortex lattice analysis. Control surfaces can be modeled by specifying the percent chord of the defined trapezoidal section, which is to be used as a control surface. In addition to the sections of the aircraft, the Mach number, wing area, wing mean chord, wing span, and location of the center of gravity must also be defined. The program outputs most of the control and stability derivatives for the desired aircraft configuration.

VLMpc is a FORTRAN based vortex lattice computer program. This program is designed to estimate the subsonic aerodynamics characteristics of up to two defined planform sections. The program allows dihedral, twist, and camber to be defined for the analysis. This is done by defining the planform using the coordinates of the corners of each section in 3-D space. Camber is included in the analysis by defining the mean camber line of the desired airfoil. Also, center of gravity location, Mach number, desired lift coefficient, and the number of horseshoe vortices in the spanwise and chordwise directions must need to be defined for the analysis to take place. The program outputs actual lift coefficient and the angle of attack at which it occurs. It also displays the induced drag coefficient.

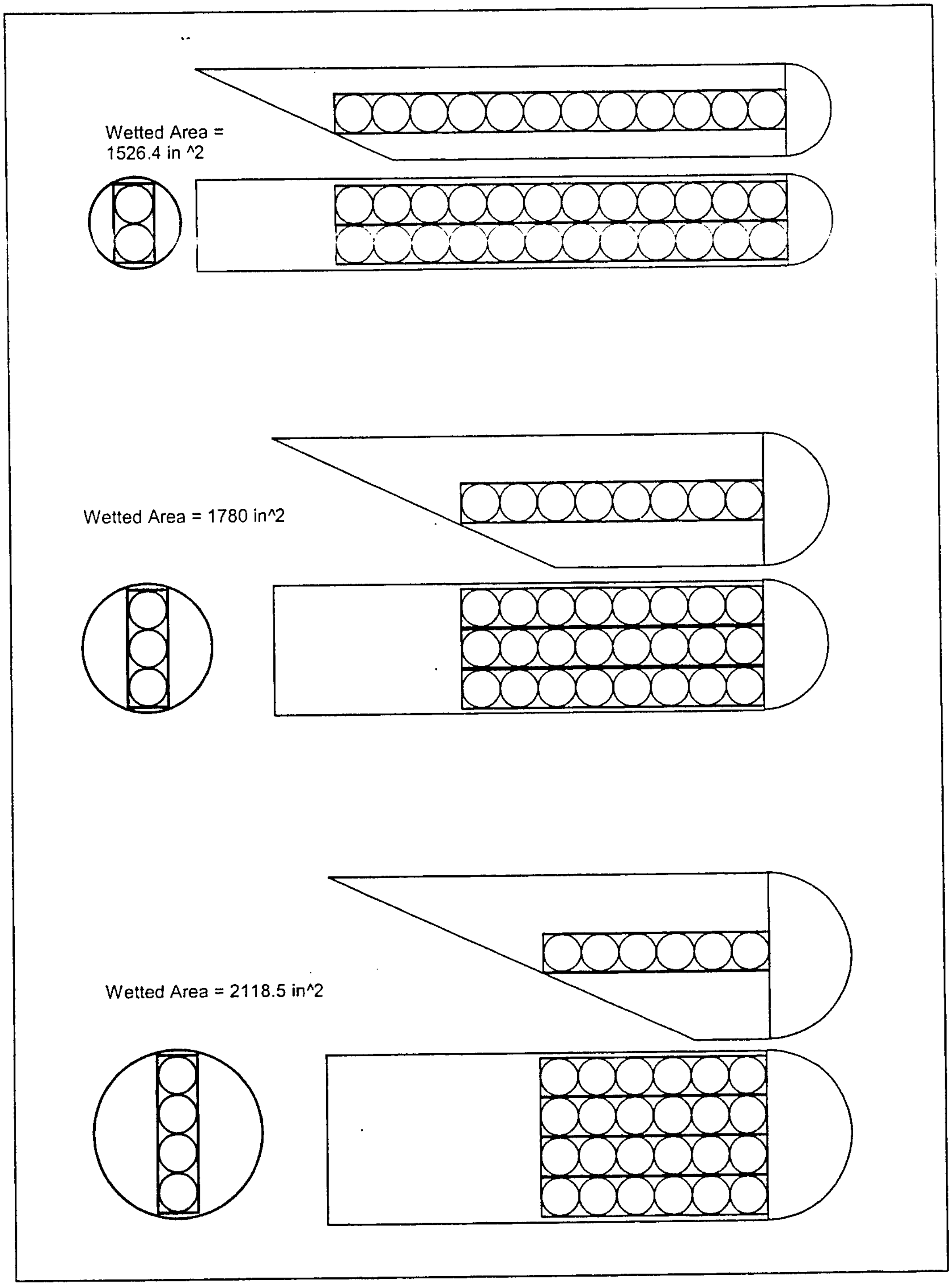


Figure 4.1.1 Typical and Joined Wing Fuselage Design

4.2 Strut-Braced/Typical Wing

From the conceptual design phase it was noted that braces would unnecessarily complicate the design and manufacture of the aircraft. The decision was therefore reached to develop the braced/typical wing solely as a typical wing aircraft. Design tools available to the team would thus be able to more accurately predict the attributes of the design.

To begin preliminary design, the aircraft was designed around the payload. The concept was to be developed with the maximum payload capacity of 24 softballs. These softballs could not be stacked and carried within the fuselage. To determine the best possible arrangement, several possibilities were considered. These included rows of softballs that were two, three, and four abreast. The balls were then covered by a circular cylinder and capped with a semi-spherical nose cone and terminated with a flat elliptical plate. The terminating plate, it was determined, could have an angle no greater that 25° to prevent flow separation from occurring. These observations were gathered from empirical relations, which referred to the transport category of aircraft. As the diameter of the cylinders increased, the termination criteria would also greatly increase their length. Payload configurations could now be compared based on their wetted surface area under the assumption that the lowest surface area would result in the least amount of profile drag. A fuselage with two rows abreast would be the optimum configuration based on that criterion. Diagrams of the fuselage design possibilities appear in Figure 4.1.1.

Before proceeding further, several initial assumptions would have to be made. From team discussions it was decided that a wingspan of about 12 feet would be a general "ballpark" estimate. A taper ratio of 0.4 was also selected in order to achieve as close as possible to the elliptical lift distribution. An initial root chord of 1 foot was also assumed. Construction materials were to be considered as foam-cored two-layer fiberglass. Two engines were to be used which would provide a constant thrust of 3.12 pounds per engine, according to their ratings. These assumptions would allow the team a starting position but would of course be refined as more accurate estimations could be obtained.

Before beginning any design, it would be necessary to obtain an estimation of weight. This would include the maximum 9.4 pounds of payload and 5 pounds of batteries. Using the assumed starting wing planform, fuselage dimensions, and construction materials it was possible to calculate an initial estimate. This value would then be multiplied by an assumed combination error and extra mass factor of 1.25 to obtain a weight of approximately 30 pounds. Modeling the fuselage as a projectile, allowing the Vortex Lattice Method Applet, VLM, to estimate the wing drag, and making allowances for landing gear etc could now also determine an estimate of drag. Thereby values for profile drag coefficient of 0.053 and induced drag coefficient of 0.033 were obtained.

In order to begin calculations for the typical aircraft configuration, a starting criterion was chosen from the mission profile. A logical consideration was that of the 200-foot runway length since the first thing the aircraft would have to do to perform its mission profile was to achieve flight within the allotted distance whether in its unloaded or loaded condition. To begin initial calculations, the aircraft was solely modeled

as a wing only in an attempt to discover a reasonable span, chord length, and airfoil shape that would allow a take-off run of less than the 200 foot maximum. This was accomplished through a combination of both externally and internally developed computer programs.

VLM was also used to obtain lift coefficient equations for the wing planform. Several airfoil sections and wing planforms were considered. Iterations were then performed using the Airplane_Takeoff MATLAB program to determine acceptable wing planforms. The angle of attack of the wing would also be an important consideration because of its significant effect on the lift coefficients. In order to avoid stall over the entire wing, each airfoil would need to be kept below its stall value. However, each wing would need a significant angle of attack to provide a large enough lift coefficient for take-off. Since the NACA 4412 airfoil seemed to have an acceptable mix of camber and stall angle, therefore it would be used as the airfoil section and the span and chord would be allowed to be the free variables instead of having all three as free variables. By making this decision, the number of iterations could be greatly reduced.

Preliminary wing dimensions were obtained for a takeoff distance of 189 feet and an 11° angle of attack. With an 11-foot wingspan, there would be room for the 1-foot diameter fuselage for a total span of 12 feet. The VLM representation of the preliminary wing's semi span may be seen in Figure 4.1.2. A result summary of the initial iterations with Airplane_Takeoff appears in Table 4.1.1.

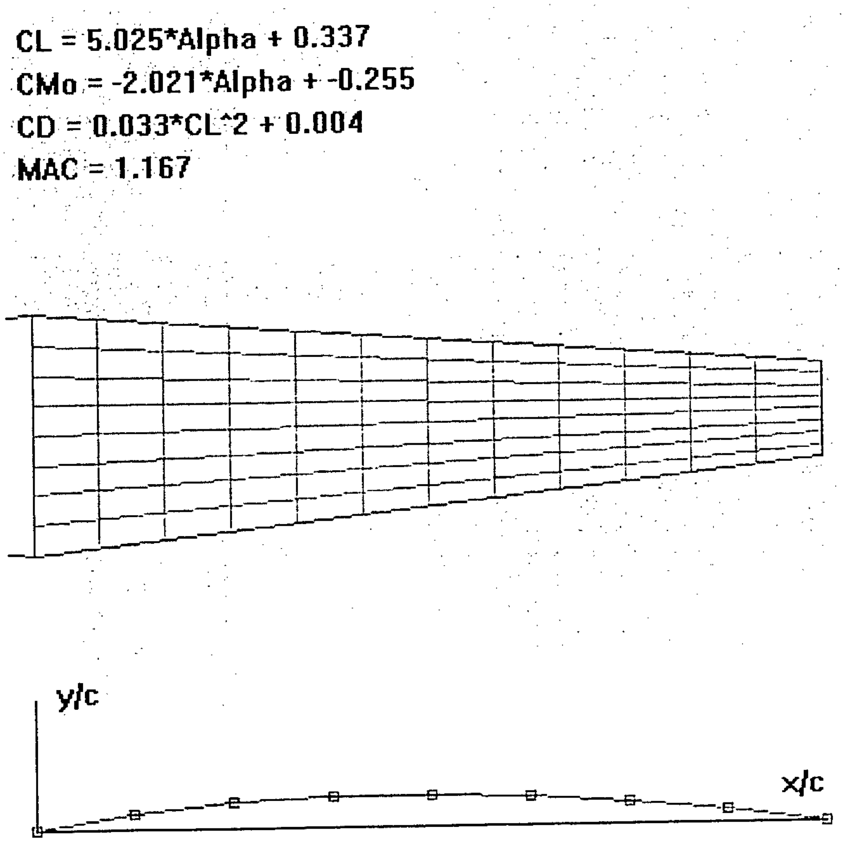


Figure 4.1.2 VLM representation of Wing Semi span with NACA 4412 airfoil

Table 4.1.1 Airplane_Takeoff Results Summary

C_{avg}	b (ft)	Taper	Sweep	W (lbs)	K	C_d	Alpha	Cl_{max}	D (ft)
1.000	10	1	0	30	0.032	0.053	10	1.198	312.01
1.050	10	0.4	0	30	0.032	0.053	10	1.212	294.82
1.050	12	0.4	0	30	0.027	0.053	10	1.257	236.25
1.050	12	0.4	0	30	0.027	0.053	12	1.439	203.28
1.050	12	0.4	4	30	0.027	0.053	12	1.443	198.72
1.050	12	0.4	4	28	0.027	0.053	12	1.443	171.69
1.167	12	0.4	4	28	0.033	0.053	12	1.415	158.10
1.167	12	0.4	4	28	0.033	0.053	10	1.236	184.12
1.167	12	0.4	4	28	0.033	0.053	11	1.325	170.90
1.167	11	0.4	4	28	0.033	0.053	11	1.301	189.24

Initial tail sizing for both the horizontal and vertical tails came from empirical formulas describing homebuilt and general aviation aircraft. To refine the tail dimensions and locations, a spreadsheet was developed in Excel called Balance. This spreadsheet utilized the lift and moment equations generated by the VLM applet for the wing, tail, and fuselage. Each surface was modeled independently in VLM and their contributions were then summed together in Balance. Through the manipulation of the lift and moment equations, the aerodynamic centers of each surface were determined. It was then at those points where the lifting force of the respective surfaces could be applied and added to the zero lift moments also calculated by Balance.

In order to properly balance the pitching behavior of the aircraft, it became necessary to estimate the center of gravity location. As part of this step, the opportunity was taken to refine the fuselage dimensions since it would carry the most massive parts of the aircraft including the payload and batteries. The circular cylinder was replaced with the more streamlined shape of an elliptic parabaloid, which would reduce the external surface area and eliminate useless internal area as well. Because of the two separate flight conditions, with and without payload, it was noted that an optimum position of the CG would be in the center of the payload compartment. Thereby, the removal of the payload would affect the weight of the aircraft but not its CG location. For convenience of calculations, all distances were referenced to the payload center location now functioning as well as the estimation of CG location. With that information in hand, the team could now proceed to define the locations for the wing and tail surfaces in relation to the CG location to ensure the pitch balance of the design.

By allowing the aerodynamic center of the wing to occur at the same longitudinal location as the CG, the challenge of achieving pitch stability could be reduced to simply obtaining accurate preliminary estimations of the tail size and location. To greater simplify the problem, the horizontal tail airfoil was chosen to be symmetric. With this selection came the realization that a negative angle of incidence of the tail might be required. Indeed, as iterations with Balance were performed, an estimation of a tail location 3 foot aft of the CG with a –4° angle of incidence was found to be a pitch balanced configuration. Some of the information calculated by Balance for this configuration can be seen in Table 4.1.2.

Table 4.1.2 Balance spreadsheet pitch balance results

Angle of Attack	C_M	C_L	V (mph)
-2	0.0864	0.0779	104.64
-1	0.0531	0.1803	68.8
0	0.0199	0.2828	54.9
1	-0.0133	0.3852	47.1
2	-0.0466	0.4877	41.8
3	-0.0798	0.5901	38.0
4	-0.1131	0.6926	35.1
5	-0.1463	0.7951	30.8
6	-0.1796	0.8975	29.2

Calculations were also performed to gain insight into how the aircraft might perform if the CG was not exactly in the location estimated. Balance was adjusted to analyze pitch behavior with the CG located either 2 inches fore or aft of the original location. Flight angle of attack under these conditions was computed and appears in Table 4.1.3. With all of the above data obtained for the typical aircraft configuration, drafts were made of the planform dimensions as appear in Figure 4.1.2. This includes fuselage design, CG location, wing location, and tail location.

Table 4.1.3 Flight angle of attack

	α	C_L	V (mph)
CG -2	3	0.5901	38.0
CG +2	-0.5	0.2316	60.7

Table 4.1.4 Pitch reaction results of 15° elevator deflection

	α	C_M	C_L	V (mph)
CG 0	10	0.0122	1.1810	26.9
CG -2	19	0.0133	2.1031	20.1
CG +2	6	0.0350	0.7712	33.3

Table 4.1.5 Airplane_Takeoff results with refined estimates

C_{avg}	b (ft)	Taper	Sweep	W (lbs)	K	C_d	Alpha	C_{lmax}	D (ft)
1.167	11	0.4	4	28	0.033	0.011	10	1.307	169.13
1.167	11	0.4	4	30	0.033	0.011	10	1.307	182.74
1.167	11	0.4	4	31	0.033	0.011	10	1.307	198.98

Preliminary planform in hand, the team proceeded to design and estimate the effectiveness of control surfaces. Specifically, the elevators were considered since they control pitch for which Balance was already set up to analyze. The last 25% of the chord for the horizontal tail was deflected by 15° and modeled again with VLM. Of course, this change affected the lift and moment equations for the tail, which are inputs to Balance. Deflection of the elevator would cause the aircraft to pitch up to a different equilibrium angle of attack. Elevator deflection data appears in Table 4.1.4. Sufficient pitch control was achieved for every CG location. However, only limited control was available if the CG was 2 inches forward of its optimum position.

The additions of the tail surfaces and the more streamlined fuselage affected the overall lift of the aircraft. The new fuselage also significantly reduced the drag due to its more aerodynamic shape. With a new C_d of 0.011 and a new C_l of 1.307 at a 10° angle of attack the Airplane_Takeoff program was re-run. Since the take-off distance had been greatly reduced, the weight estimation was increased in an attempt to estimate the total weight the given planform could lift off in the allotted distance. These results appear in Table 4.1.5, where a maximum weight of 31 lbs was estimated as the limit weight for the aircraft.

With the progress achieved so for in the preliminary design phase for the typical aircraft, a significant portion of the aerodynamic design was complete. Structural, electronic, other design aspects were left for the detailed design phase. Although vertical tail dimension were also calculated via empirical relations, analysis of their performance was also left for the detail design. Many lessons were learned and useful tools developed that would benefit any design selected for the detail design.

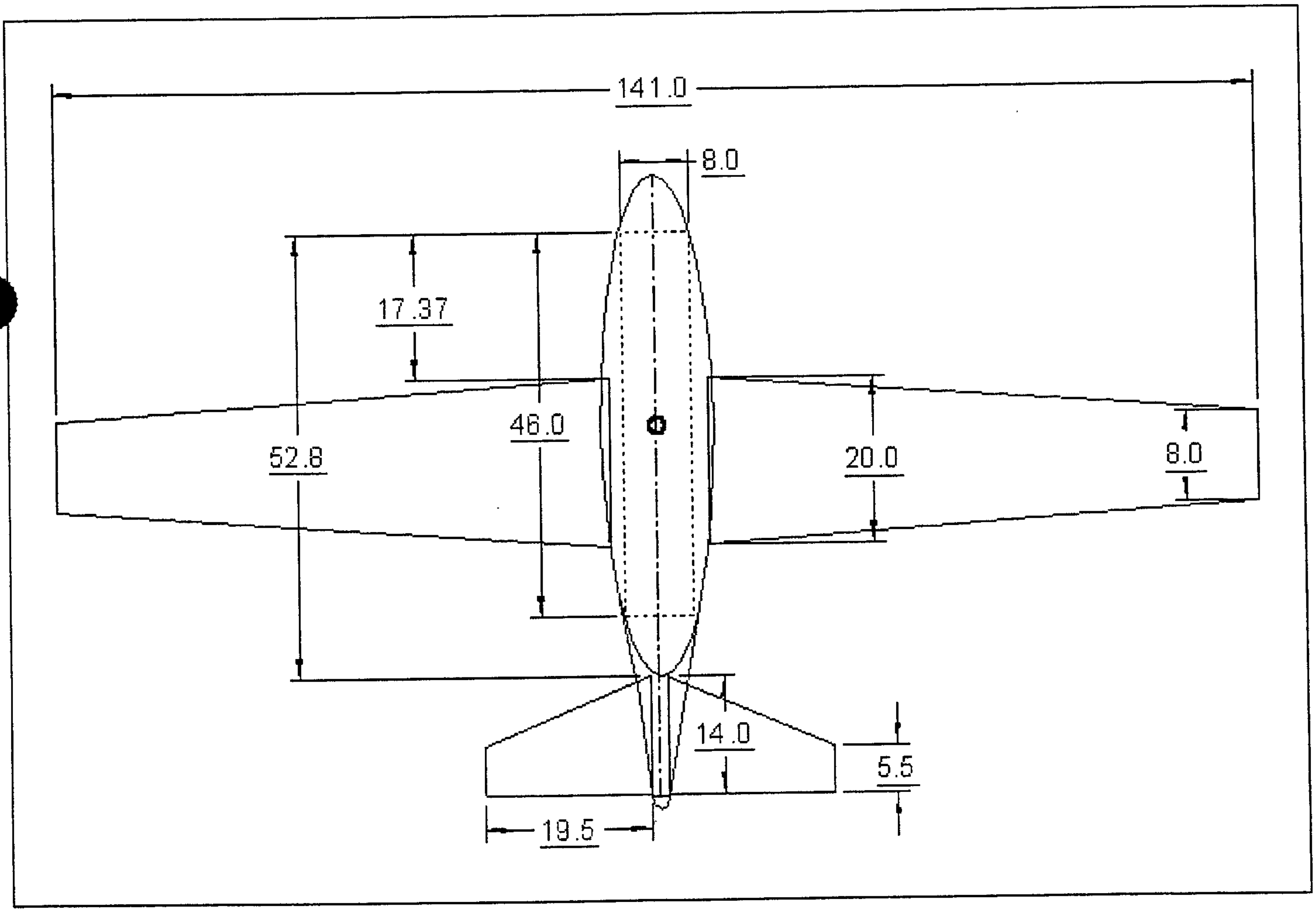


Figure 4.1.2 Typical aircraft preliminary planform (measurements in inches)

4.3 Joined Wing

The first task to accomplish for the joined wing was to design the fuselage. Three configurations for the softballs were looked at: two across, three across, and four across. For all three configurations a total

of twenty-four balls were used. The fuselage was chosen to be a cylinder wrapped around the balls and the wetted surface area was calculated for each of them with the idea being that the one with the least surface area would have the least drag and would be the best choice. The one that was chosen was the two by twelve configuration. The minimum length for this configuration was 4 feet. The diameter of the fuselage was 0.7733 feet.

To determine the remaining portions of the joined wing configuration certain simplifying assumptions had to be made. First, the relationships discussed in the aircraft design text were assumed to be applicable to the joined wing aircraft. The following equation was used to determine the size of the vertical tail:

$$Cvt = (Lvt*Svt)/(bw*Sw)$$

Where Cvt = Vertical tail coefficient

Lvt = length of the tail arm from the CG to the AC of the vertical tail

Svt = Surface area of the vertical tail

bw = span of the main wing

Sw=surface area of main wing

Once the size of the vertical tail was found, the following values were used to iterate on a solution that would give the proper dimensions of the vertical tail and length of the fuselage:

Per = percent length of fuselage for tail arm measures from the CG to the tail AC

AR = aspect ratio of vertical tail

Lambda = taper ratio of the vertical tail

Cvt, Per, and AR are values set to those from tables in the textbook that most closely resembles the aircraft for this project. For the horizontal tail, it was assumed that it would be determined from dimensions of the main wing and the vertical tail. The horizontal tail would start from the tip of the vertical tail and end at 70% of the wingspan. The chord of the horizontal tail at the vertical tail end would be the chord at the tip of the vertical tail and the chord at the tip would be the chord of the wing at 70% span. Once these values and assumptions were set, an Excel spread sheet was set up to determine all the dimensions of a joined wing aircraft given the dimensions of the main wing. The values for the main wing tip chord, root chord, sweep, span, and initial length of the fuselage were inputted and values for the vertical tail surface area, tip chord, root chord, span and new length of fuselage were calculated. Also, the horizontal tail span, forward sweep, anhedral, root chord, tip chord and surface area were calculated from the information. An initial calculation for Cd_0 was also performed once all the surface areas were defined. This was done for a range of spans, chord lengths, and taper ratios for the main wing.

Once the geometry of the aircraft was defined, certain analysis tools could then be used to determine the aerodynamic properties of the airplane. For the joined wing, these tools were limited because of the setup of the joined wing. Most of the programs available to this team were setup to analyze conventional aircraft wings. When two separate wings are joined together, analyzing the results becomes more difficult. The horizontal tail section behind the main wing is going to be affected by the flow from the main wing, especially around the joined area. Therefore, analyzing the sections separately and adding the results together is not a valid approach to this configuration. A program that could handle multiple wing sections and do an analysis for the entire airplane was needed. One of the programs used was WINGBODY. Even WINGBODY could not handle a joined wing entered into the program as is. WINGBODY used the vortex lattice method to get the properties of the aircraft. Each section is divided into panels and a vortex is placed in the panel to simulate airflow characteristics over the wing. The panels on the main wing and those on the horizontal tail needed to be lined up so that an invalid moment wasn't created on the horizontal tail due to the vortex from the front wing and moment arm from the panels not lining up. The airplane was divided up into panels with the constraints in mind. The Excel spread sheet was set up to calculate the exact coordinates needed for the WINGBODY program. Therefore, each of the configurations of the joined wing could be tested in WINGBODY. Another constraint of the program was that the aircraft couldn't have a nose. WINGBODY does not give accurate results when a nose is attached so it had to be left off to get reasonable results. WINGBODY determined the lift coefficient and the induced drag coefficient at different angles of attack of the airplane. From these results, the K of the airplane could be determined. After these values were determined, the Airplane_Takeoff program was used to see if the given aircraft could take off in the constrained distance of 200 feet. An initial weight estimate of 30 pounds was used.

This process was repeated to optimize the aircraft. Due to time constraints and technical problems in the program, the best-optimized aircraft was not determined. An aircraft with good results was found and used for further analysis. A picture of this aircraft is shown in Figure 4.2.1.

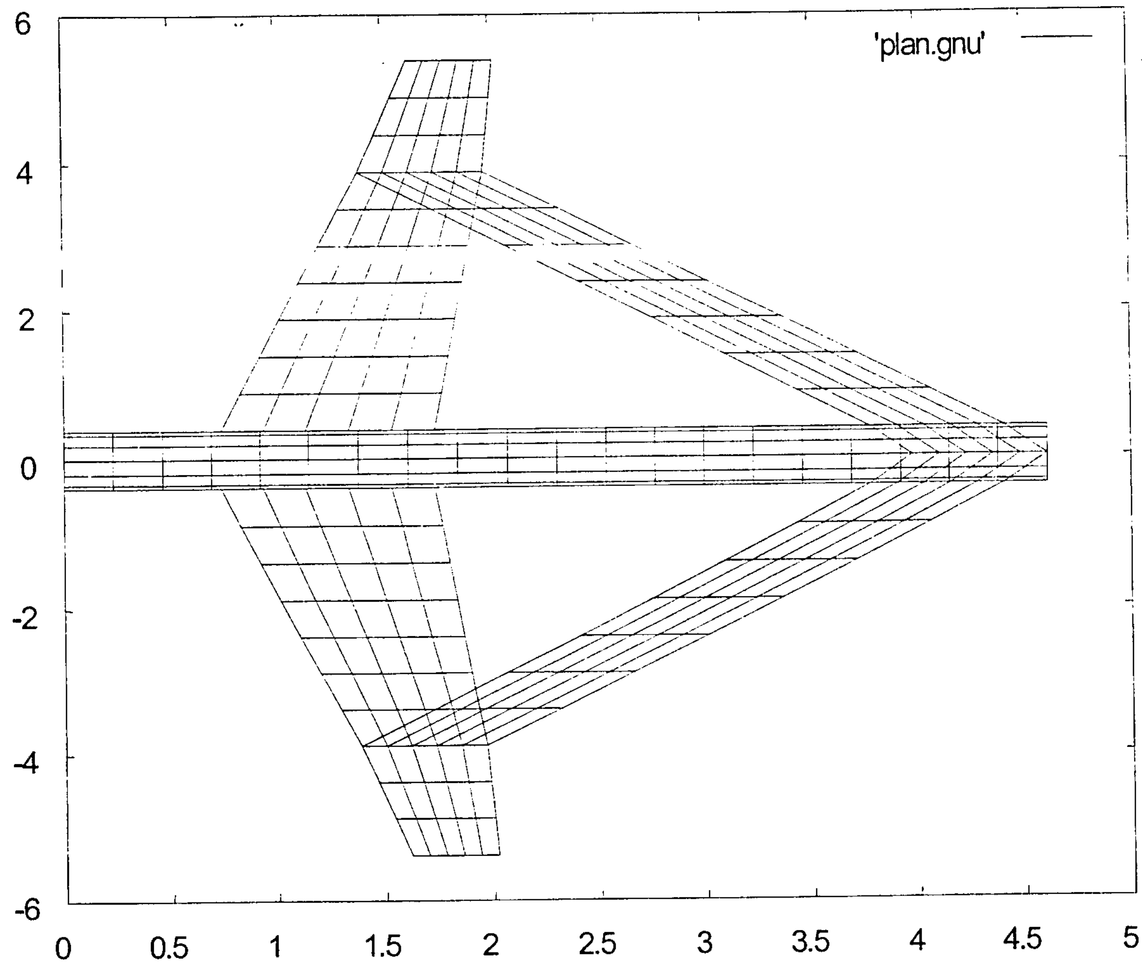


Figure 4.2.1 Joined Wing WINGBODY Illustration

The final program used to determine the characteristics of the aircraft was JKayVLM. This program is also a vortex lattice method program so the same constraints as with the WINGBODY program applied. In this program, the dimensions of the aircraft as well as control surfaces needed to be defined. The output for this program is most of the stability and control derivatives for the entered planform. The Excel spread sheet set up the data for the inputs for the longitudinal and lateral files. The spreadsheet was set up so that the position and size of the control surfaces could be determined by setting percentages of span for where each control surface begins and ends. The constraint that the panels need to line up in the section where the two wings joined still applied. Each panel is divided into 8 streamwise and 5 spanwise panels. Therefore, the panels on the main wing and the horizontal tail, which are joined, need to be the same length spanwise in order for the panels to line up. This means that the end of the aileron and the end of the elevator need to occur at the same spanwise location. This limits some of the design possibilities if this program were to be used.

Due to time constraints, the analysis could not be completed for any given configuration. To do this, a complete structural evaluation must also be done on top of the aerodynamic, stability, and control analysis. To accomplish these tasks will require more time and resources than were available. The problems associated with the aerodynamic, stability, and control information has already been discussed. There is a different set of problems for the structural part of the evaluation. For a joined wing the problem becomes statically indeterminate. In theory, a joined wing is structurally stronger than a conventional aircraft wing. To do an evaluation for skin thickness, rib placement and all other structural would become relatively more difficult than for un-joined wings. The problem looks like the model in Figure 4.2.2. Each component cannot be individually considered, but the entire wing structure needs to be evaluated at once.

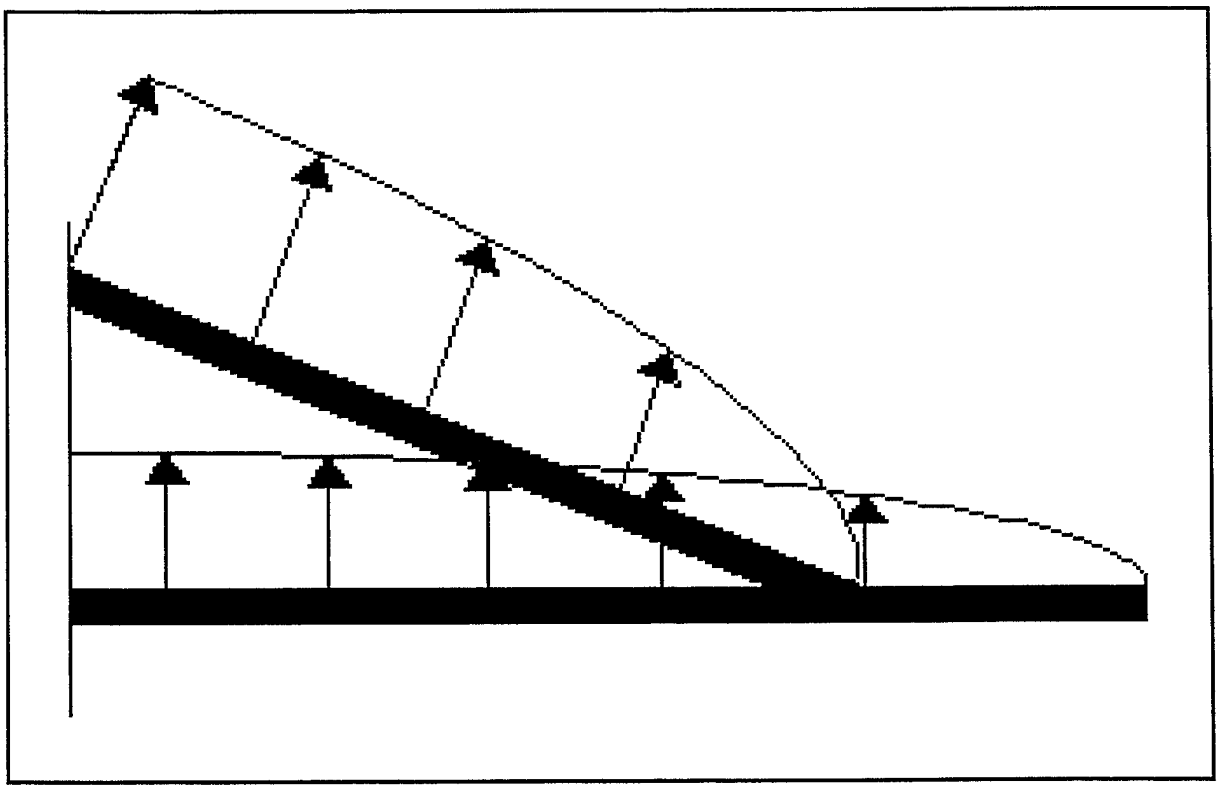


Figure 4.2.2 Illustration of Static Indeterminacy

The joined wing has many theoretical advantages over a conventional wing. However, coming up with numbers for an actual joined wing aircraft proved to be much more difficult. Most of the programs that were used as tools were set up with typical conventions in mind and needed to be modified or forced to work in a way to handle a joined wing. For further evaluation of both aerodynamic and structural properties, new tools need to be created. For aerodynamics, the programs used already are pretty much the best can be done right now. For the structural question, a program would probably have to be created specifically for the joined wing configuration. Since programs already exist that can be used to

design other aircraft, it would not be worth the time and effort to create a set of programs just for one type of configuration.

4.4 Flying Wing

The Flying Wing Group (FWG) was given the task of completing the preliminary design on the flying wing configuration. The FWG divided the project into three parts: 1) structures, 2) performance, and 3) stability. Performance and stability are related and could have been included folded into one group, however the FWG felt that the inherent instability of the flying wing configuration necessitated that the stability of the configuration be a major concern (and, consequently, its own group).

The FWG realized almost immediately that the DBF team had no quick method of analyzing the structure of a wing. The FWG spent the first several weeks writing a MATLAB computer program, WINGSTRUCTURE.M, which would accomplish this very goal. This would allow the FWG optimize the wing structure for lowest structure weight (calculations could be performed quickly). It then accepts as input airfoil geometry, wing geometry, and an IDRAG loading distribution (although the program could be modified to accept loading distribution of any sort, IDRAG loading distributions were the most readily available to the FWG). The program calculates running lift, shear force, bending moment and torque at a specified number of stations along the span of the wing. It also calculates longitudinal stress due to bending moment, transverse stress due to shear force and torque, and critical buckling stress of upper skin at each station. The program then compares the longitudinal stress to the buckling stress at each station and decides if ribs are needed and there placement. The principle stress is then calculated using the Strain Energy of Distortion Theory (von Mises) and compared to the yielding stress. The user is alerted if the principle stress is larger than the yielding stress. At this point the program is ready to make an estimation of wing weight based on the number of ribs and changes in airfoil dimensions (structural volume). These calculations were all based upon the assumption that the airfoil was a thin-walled closed section (i.e., the shear flow along the skin was not constant). An example of the graphs of Loading vs. Span, Shear vs. Span, and Moment vs. Span that WINGSTRUCTURE.M produces can be seen in Figures 4.3.1, 4.3.2, and 4.3.3 respectively.

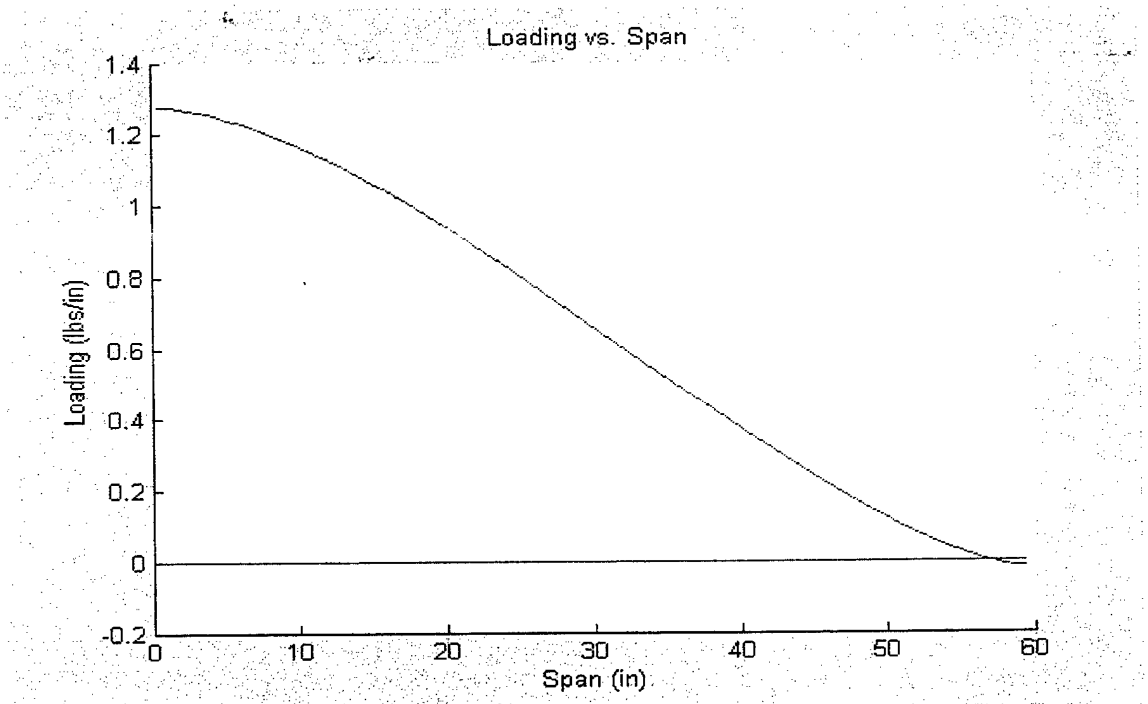


Figure 4.3.1 Loading vs. Span

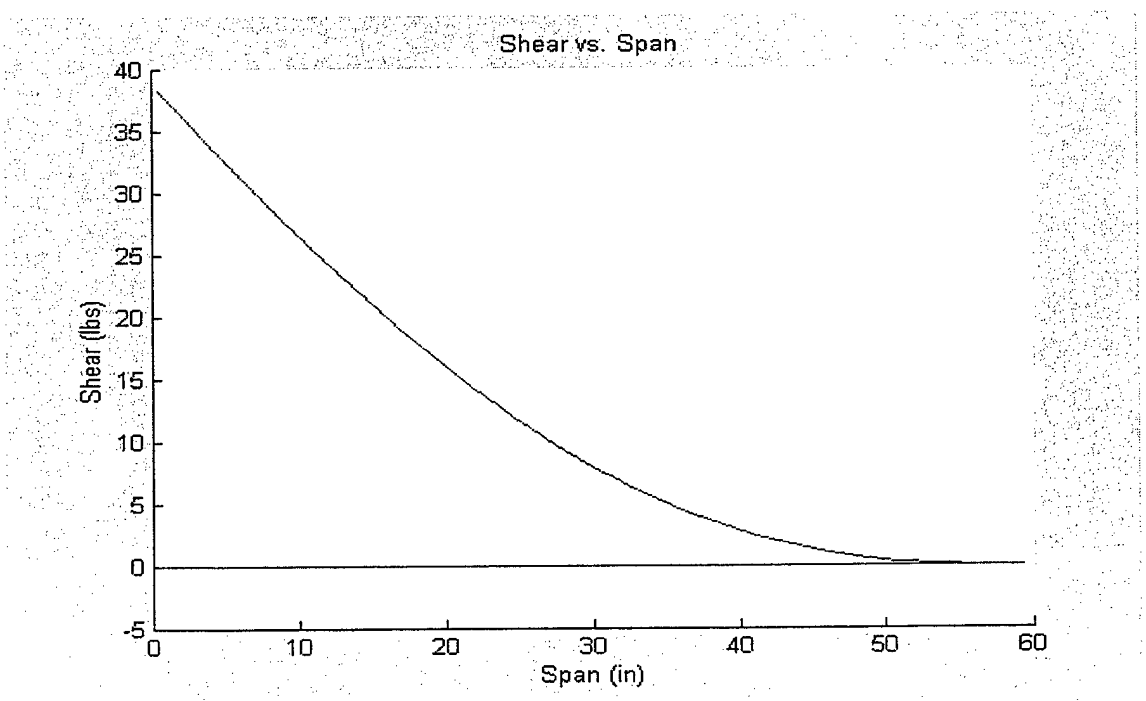


Figure 4.3.2 Shear vs. Span

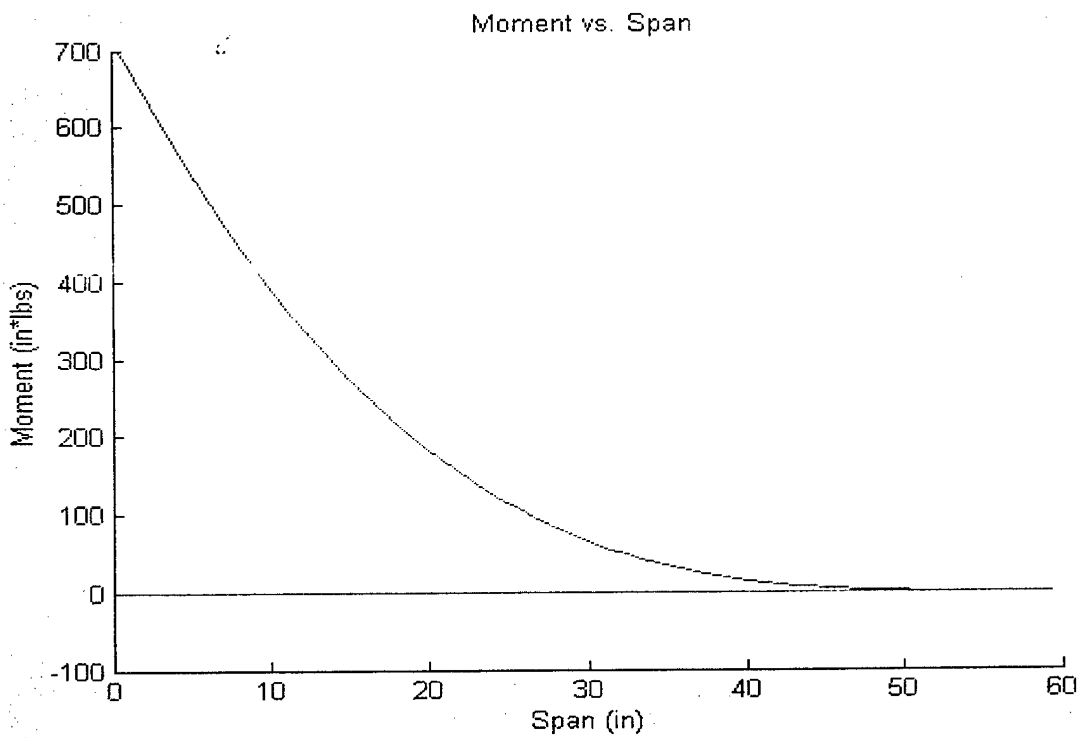


Figure 4.3.3 Moment vs. Spa n

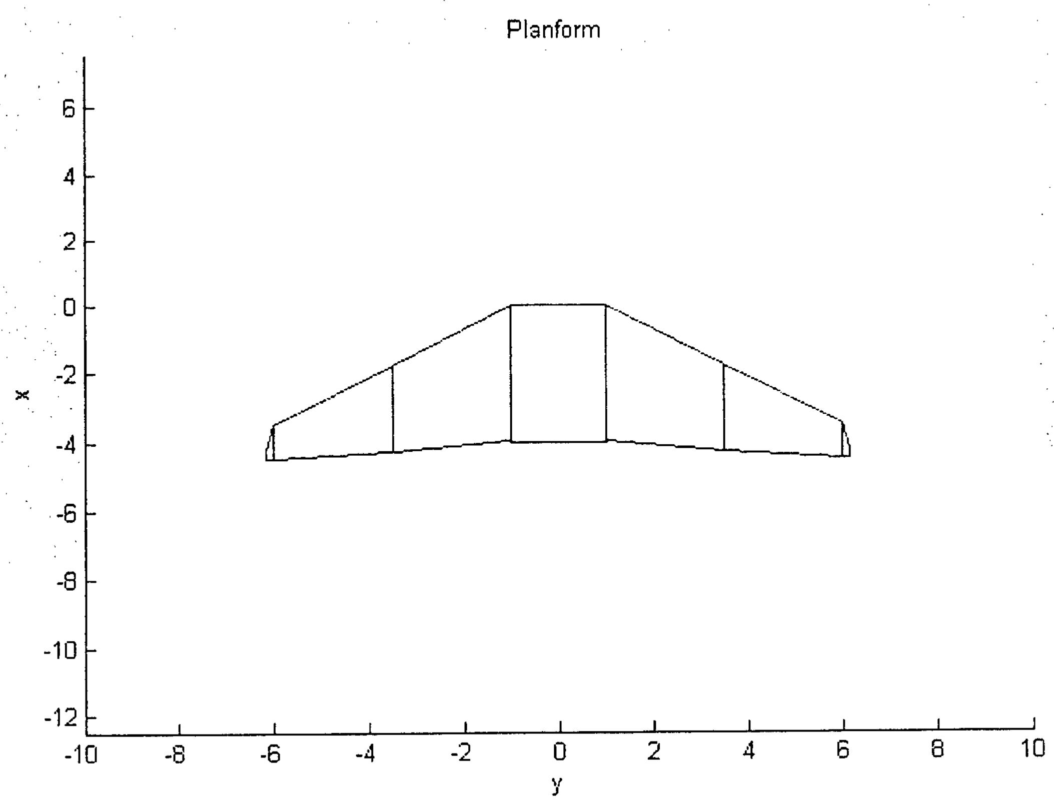


Figure 4.3.4 Final Planform

To make use of this new analytical tool, the FWG still needed to specify a few of the input requirement of WINGSTRUCTURE.M. The planform needed to be sized. The center-section of the flying wing (the 'body' of the configuration) was sized by determining number of balls the FWG wanted to design for and adding to this capacity the extra room needed to carry such items as batteries and radios. The FWG decided to attempt to carry the maximum number of balls, 24. They were arranged into 6 chord-wise rows and 4 span-wise columns. This meant that the center-section had a 2-foot span and a 4-foot chord. The wing sizing followed in a similar manner. The wing root needed to match the 4-foot chord of the center-section. A taper ratio of 0.25 was selected. A tip root of 1-foot achieved the desired taper ratio. A wing sweep angle of 35° was settled on after several trial-and-error experiments using IDRAG and comparing the loading, shear, and moment generated at a variety of wing sweep angles. The wing sweep helps to move the aircraft's aerodynamic center aft. To determine the span of the wing, the FWG started with a lift coefficient (C_L) of 1.37, a weight of 55 pounds (the maximum allowed), and velocity (V) of 40-miles per hour. Once these variables were known, a planform area of 25 square feet was calculated by setting lift equal to weight and solving for planform area. This set are wing half span at 5-foot. The final planform with winglets (see later discussion in this section) can be seen in Figure 4.3.4.

With the planform geometry as described, the airfoil geometry needed to be determined. The FWG group decided that this was a detailed design consideration and, for this stage in the design process, just simply picked an airfoil shape. Finally, the properties of the construction material, fiberglass, were needed. The design textbook had the needed information.

The FWG could now optimize the X_{cg} position, where X is measured down the center of the aircraft body from the nose to the tail, weight, and wall thickness. The fiberglass comes in sheets that are 0.0055-inches thick – meaning the minimum wall thickness was 0.0055-inches. An X_{cg} position was then choosen to optimize the weight of the aircraft at each wall thickness (ranging from one layer to ten layers of fiberglass), IDRAG was run with a starting $C_L = 1.37$. (X_{cg} positions, ranging from 1-ft to 2.5-ft from the nose of the aircraft, were considered). WINGSTRUCTURE.M was then run to determine the weight of the aircraft structure at this wall thickness, X_{cg}, and C_L value. This new weight was then used to determine a new C_L and IDRAG was run again using this new C_L. This would start the process anew. In this manner, FWG could converge on a weight for each wall thickness and X_{cg} position. A flow chart of this process can be seen Figure 4.3.5.

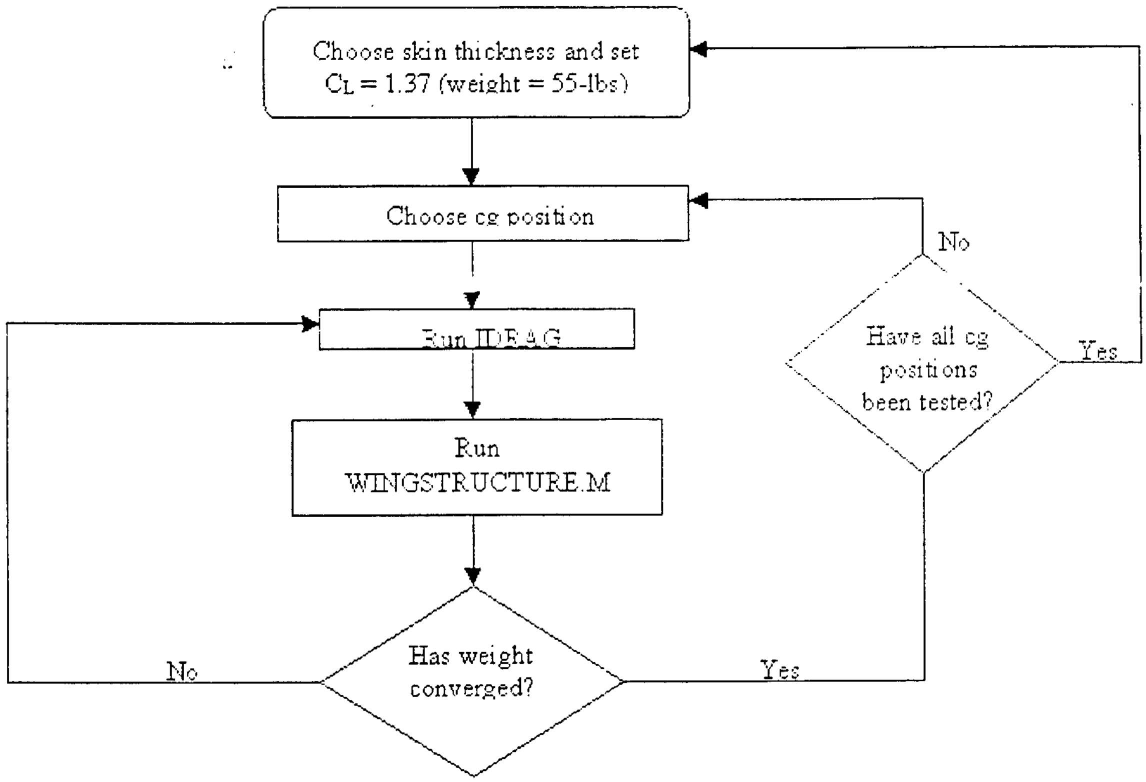


Figure 4.3.5 Process Flow Chart

The FWG could now determine the best X_{cg} position. Wall thickness, span efficiency, induced drag coefficient, material yielding margin of safety, and weight could each be plotted versus X_{cg} position. A plot of span efficiency, induced drag coefficient, and material yielding margin of safety vs. X_{cg} position and a plot of weight vs. X_{cg} position for 8-layers of fiberglass (0.044-inches thick) can be seen in Figures 4.3.6 and 4.3.7 respectively.

The optimal X_{cg} position was determined to be 2-ft from the nose. This position tended to have the highest span efficiency, margin of safety and lowest induced drag for the lowest weight for each of the wall thickness in the included range. The optimal wall thickness could now be determined by comparing span efficiency, induced drag coefficient and margin of safety for each wall thickness at x_{cg} equals 2-feet. A plot of span efficiency, induced drag coefficient and margin of safety for wall thickness ranging from four layers to eight layers of fiberglass can be seen in figure 4.3.8. A plot of weight vs. wall thickness over the same range can be seen in figure 4.3.9.

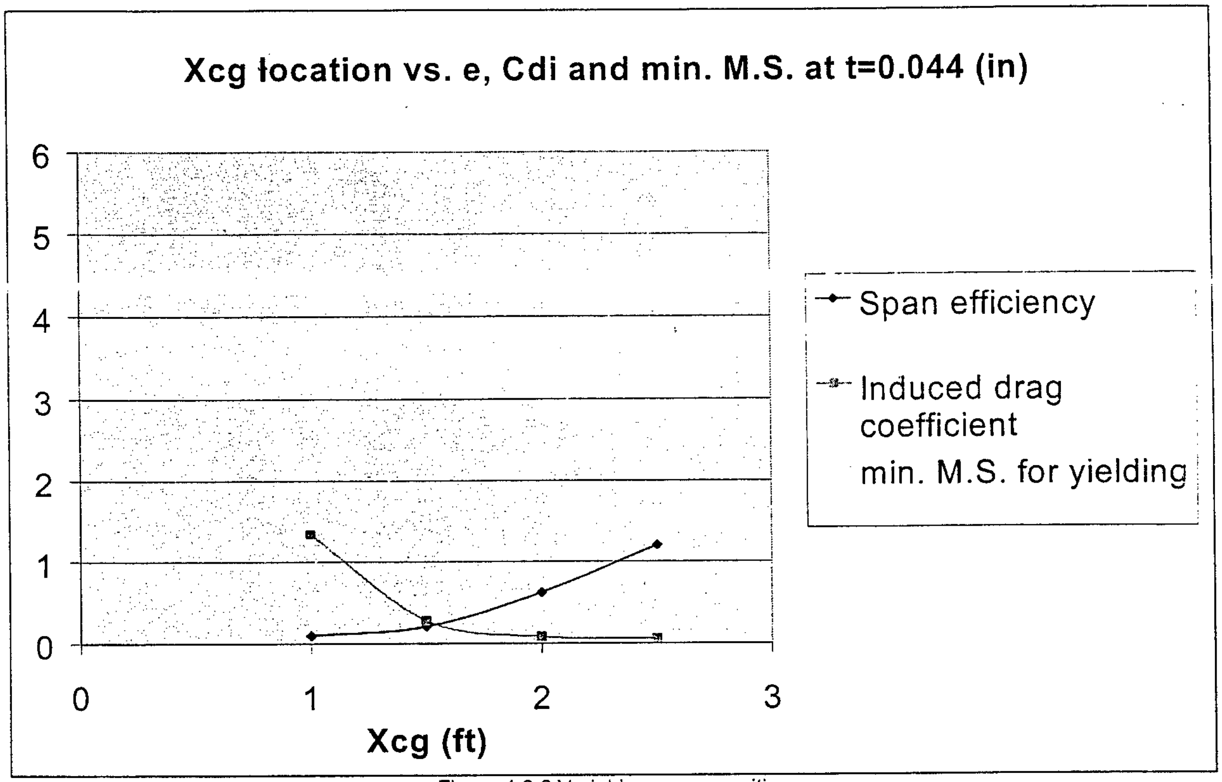


Figure 4.3.6 Variables vs. x_{cg} position

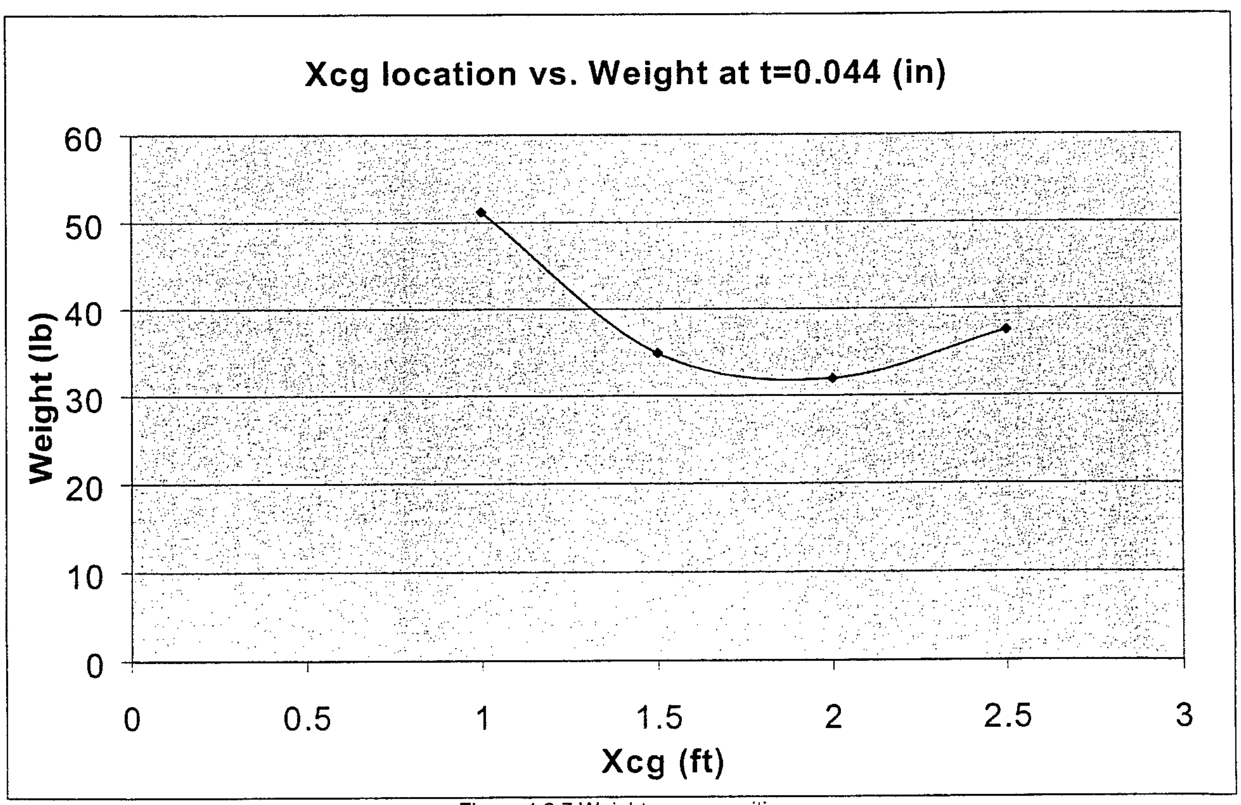


Figure 4.3.7 Weight vs. x_{cg} position

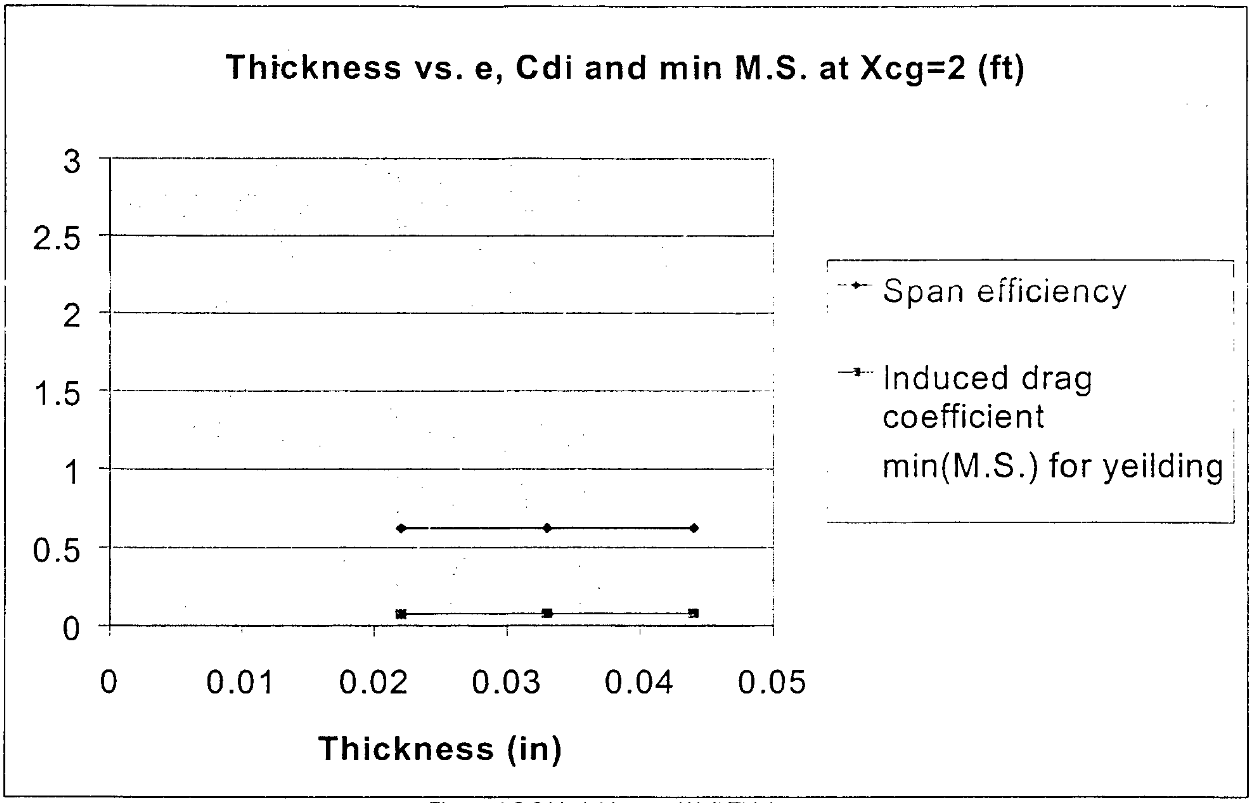


Figure 4.3.8 Variables vs. Wall Thickness

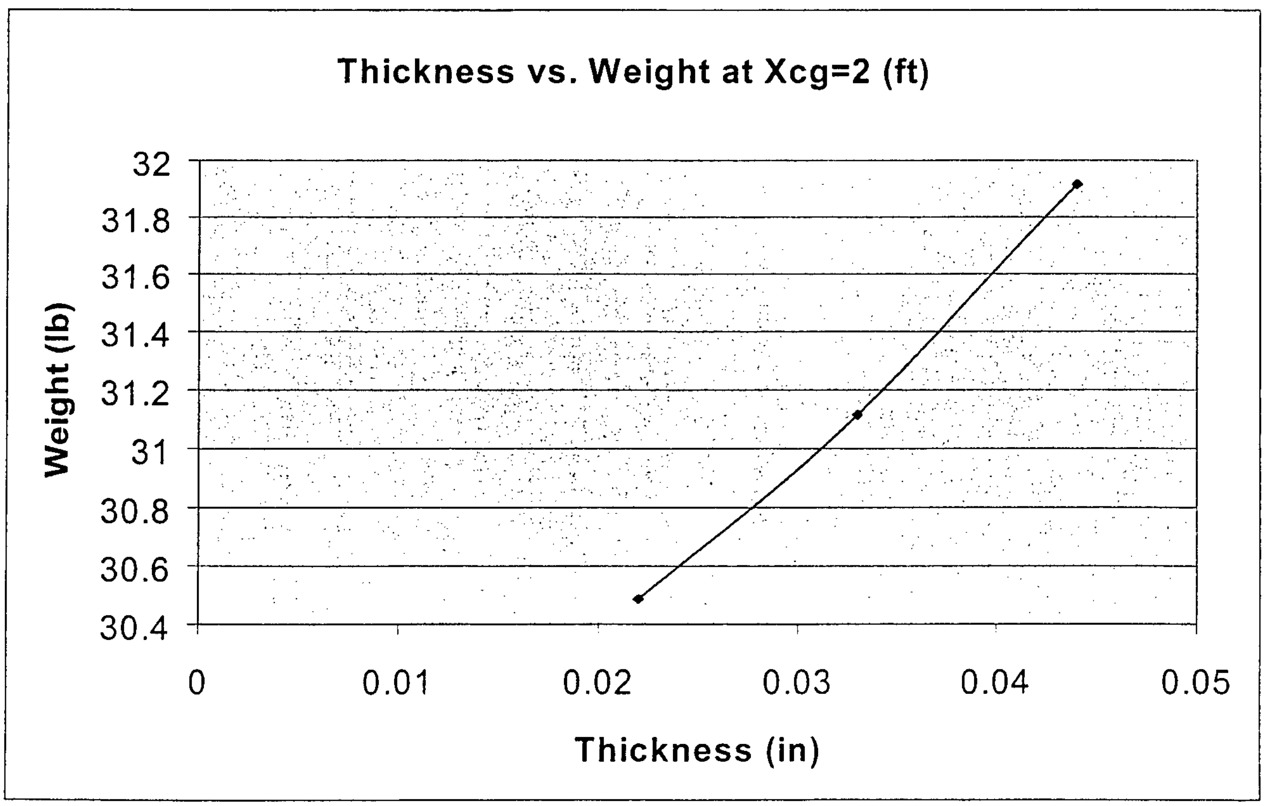


Figure 4.3.9 Weight vs. wall thickness

The optimal wall thickness was determined to be six layers of fiberglass (0.033 inches thick). The weight for six layers of fiberglass and an X_{cg} position of 2 feet was 31.11399 pounds. The span efficiency was 0.623, and the induced drag coefficient was 0.07675. The margin of safety for yielding was 2.5366. This concluded the FWG analysis of the flying wing configuration's structure.

To describe the performance of the aircraft accurately, the FWG needed to first describe the drag polar of the aircraft. The FWG was familiar with IDRAG from the structures part of the project. IDRAG provides an induced drag coefficient and a lift coefficient and therefore the induced drag coefficient factor (K) could be backed out from these values. Another analysis program, JKayVLM, had the ability to directly calculate the induced drag coefficient factor. The two program gave different values for induced drag coefficient factor. The FWG made the decision to refine this value in the detailed design phase. The FWG would base all of its calculations on the higher of the two values (from IDRAG), setting K = 0.1318. The second part of the drag polar, the zero-lift drag (Cd_0), proved just as problematic to find as the induced drag coefficient. For the flying wing configuration, the zero-lift drag is highly dependent on the airfoil shape – a consideration the FWG left for the detailed design phase. With no analysis techniques to guide the FWG, an educated guess based on available model aircraft data was made. The FWG set Cd_0 = 0.01.

Next, an estimation of the minimum and maximum velocity of the aircraft needed to be established. For prop aircraft, the minimum and maximum velocity are determined by the intersection of the power required and power available curves. The power required would vary according to how much the aircraft weighs. This means there were two cases to consider: the unloaded case (without the softballs) and the loaded case (with the softballs). For the engines available to the DBF team, power available – loaded and power required vs. velocity can be seen in Figure 4.3.10. Power available – unloaded and power required vs. velocity can be seen in Figure 4.3.11.

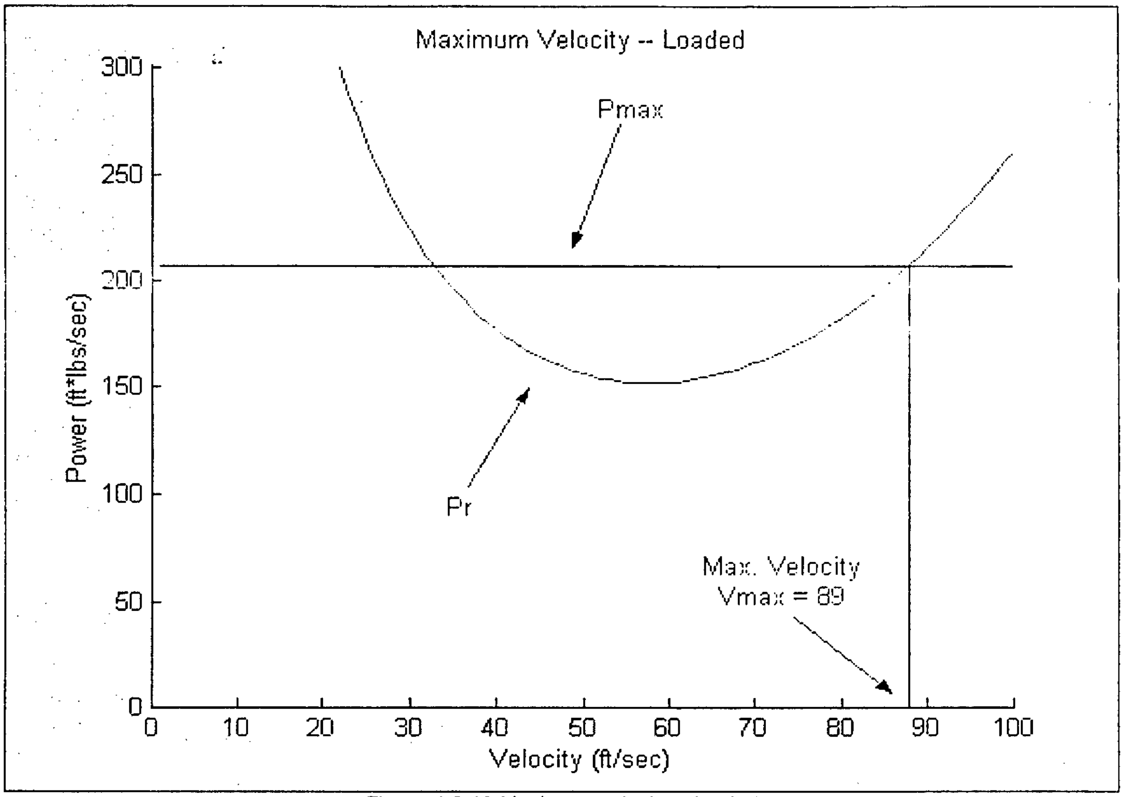


Figure 4.3.10 Maximum velocity – loaded

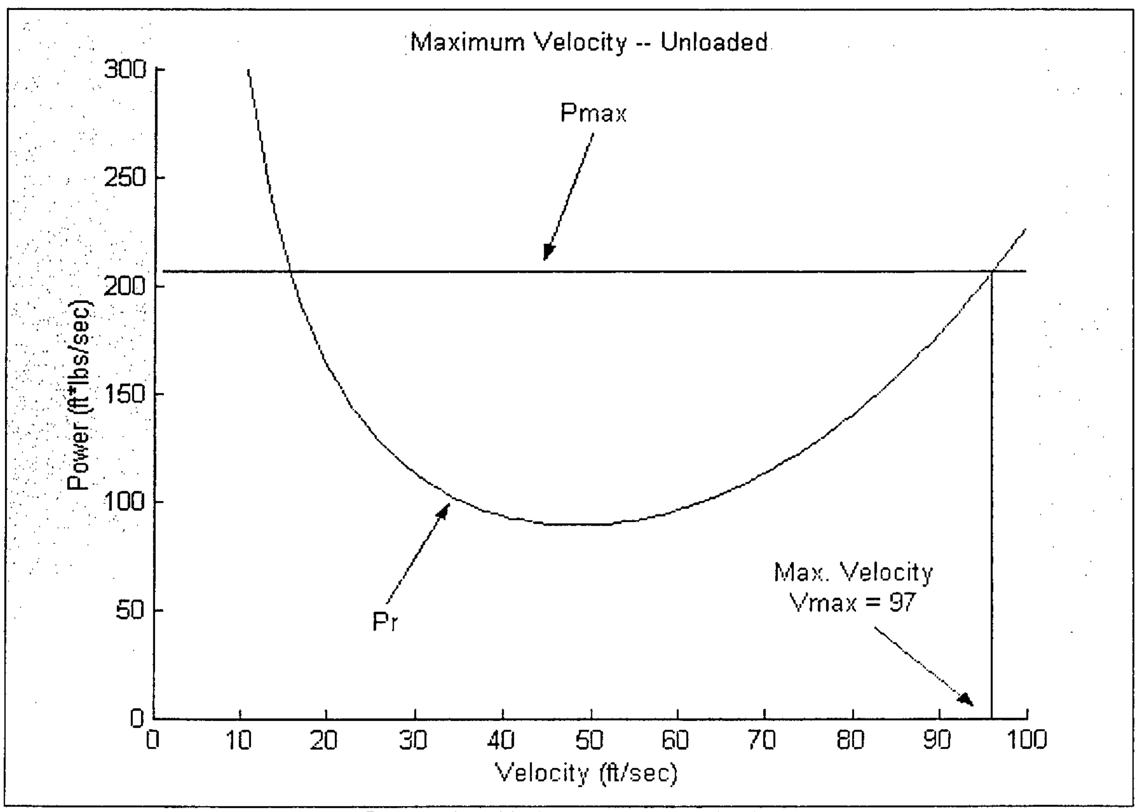


Figure 4.3.11 Maximum velocity – unloaded

Several other important velocities were calculated during this portion of the design phase: the velocity for maximum range and the velocity for maximum endurance for both the loaded and unloaded configurations. Where fuel considerations where an important component in the equation, battery power was substituted. A summary of the velocities can be seen in Table 4.3.1.

Table 4.3.1 Velocity Summary

Case	Velocity (ft/sec)
Max. Velocity Loaded	89
Max. Velocity Unloaded	97
Velocity for Max. Range Loaded	40.3139
Velocity for Max. Range Unloaded	48.1466
Velocity for Max. Endurance Loaded	30.6319
Velocity for Max. Endurance Unloaded	36.5835

A MATLAB program, TIMES.M, was then written to compute the time around the pre-defined course accounting for the differences in distances due to the loading configuration. The velocity and minimum turning radius changed based on the loading configuration. This program accepts as input the velocity and output the time around the course. The time around the course was computed for the aircraft flying at maximum velocity, velocity for maximum range, and velocity for maximum endurance. A summary of these times can be seen in Figure 4.3.2.

Table 4.3.2 Time Summary

Case	Time (min)
Max. Velocity	5.41
Velocity for Max. Range	8.31
Velocity for Max. Endurance	9.98

The final piece to the performance puzzle was to ensure the flying wing aircraft could take-off and land in the allotted 200-foot space. Another MATLAB program, Airplane_Takeoff, was written to analyzing the take-off and landing performance of the flying wing aircraft. The program accepts as input several performance characteristics and outputs ground roll distances and take-off velocities. A summary of this data for the flying wing configuration can be seen in Table 4.3.3. This concluded the FWG analysis of the flying wing configuration's performance characteristics.

Table 4.3.3 Data Summary

Case	Ground Roll (ft)	Take-Off Velocity (ft/sec)
Loaded	103.018	33.21
Unloaded	50.919	28.74

The stability of the flying wing configuration was a major concern from the beginning, particularly at the relatively slow speeds that our aircraft would be flying. These low speeds would serve only to aggravate the already unsteady stability of the flying wing. A previously mentioned program, JKayVLM, was used to analyze the stability of the flying wing. This program provides static stability derivatives based on wing planform geometry and several performance characteristics. Several of the more important stability derivatives can be seen in Table 4.3.4.

Table 4.3.4 Stability Derivatives

Stability Derivative	Value
C_{m0}	0.0
C_{ma}	0.02908
C_{la}	4.06934
C_{mq}	-0.71235
C_{Lse}	0.66192
C_{mse}	-0.22371
$C_{n[illegible]r}$	0.00009

As can be seen in the table, many of the stability derivatives are within acceptable ranges. The moment coefficient in level flight is zero. All of the derivatives have the correct sign for stability except for C_{ma}, which is only slightly positive. The stability problems of this configuration can be overcome in the detailed design phase.

5. Detailed Design

For the detailed design phase the team was reorganized into four disciplinary groups. These groups were the following: Layout Weights and Inertia, Aerodynamics Performance and Controls, Structures and Manufacturing, and Systems and Power. Ryan Davidson was appointed as the group leader responsible for inter-disciplinary group interaction and overall team progress. After each of the three preliminary design concepts were presented to the team, the senior members of the team were allowed to vote by secret ballot via independent technical progress evaluations for the design that they believed could best meet the requirements of the competition. By a vote of 7 to 2 the flying wing design was selected primarily because of the extensive amount of structural calculations that had been perform during its preliminary design. The typical aircraft garnered the second place position while all team members summarily rejected the joined wing. Some simplifying assumptions and adjustment were then made by the team in order to expedite the detailed design process. One such adjustment was the addition of a horizontal tail to augment pitch stability rather than the reliance on a reflexive airfoil such as the MH-64 on a wing that would have to be both highly swept and twisted.

5.1 Aerodynamics, Performance, and Controls

Due to the intensive effort directed toward structural design for the flying wing during the preliminary design phase, existing aerodynamic related calculations were primarily concerned with lift coefficient only. Concerns of airfoil selection, aircraft stability, control placement, and performance had not yet been addressed in sufficient detail. Tools developed during the typical aircraft's aerodynamic design would be very useful in helping to obtain aerodynamic design details for the flying wing. The addition of the horizontal tail and loss of wing twist allowed the previously used design tools to still be applicable.

Once the planform for the wing section was determined from the preliminary design, the specific airfoil for the wing had to be chosen. The chord at the root of the wing is 3.5 feet. The thickness of the balls is 3.82 inches, but to encompass the balls and surrounding structure, 5 inches is required. From these two numbers, the maximum thickness to chord ratio at the root was found to be 12%. Since data for NACA airfoils have been tested, tabulated, and are easily accessible, the airfoil for this design was

going to be picked from the NACA selection. Therefore, airfoils with the NACA notation of XX12 would be investigated. The specific camber of the airfoil would be determined from the required lift coefficient for the airplane. The IDRAG program was used to determine an ideal load distribution and lift coefficient for the given wing. The camber of the airfoil and, thus, the specific NACA airfoil, would be determined by matching the lift coefficient from IDRAG and the actual lift coefficient of the wing planform with a cambered airfoil section using Balance. The selection was narrowed down to two airfoils, the NACA 4412 and NACA 5512. The NACA 5512 had a better lift coefficient, but not enough needed data was present for this particular airfoil. The NACA 4412, however, is a very common airfoil section and lots of data can be found on this airfoil, so the NACA 4412 was chosen. To make sure that this airfoil would be acceptable, the manufacturing group was consulted to make sure the manufacturability of this particular section would not be to difficult. The actual airfoil was then compared to the structure in the body, which it must encompass. The entire ball section did not fit into the actual airfoil section, so, for the body section, a modified NACA 4412 was used just for the section encompassing the balls. Figure 5.1.1 shows an example of how the airfoil was modified. A short transition would then have to occur from the modified 4412 at the body to the actual 4412 at the wing root.

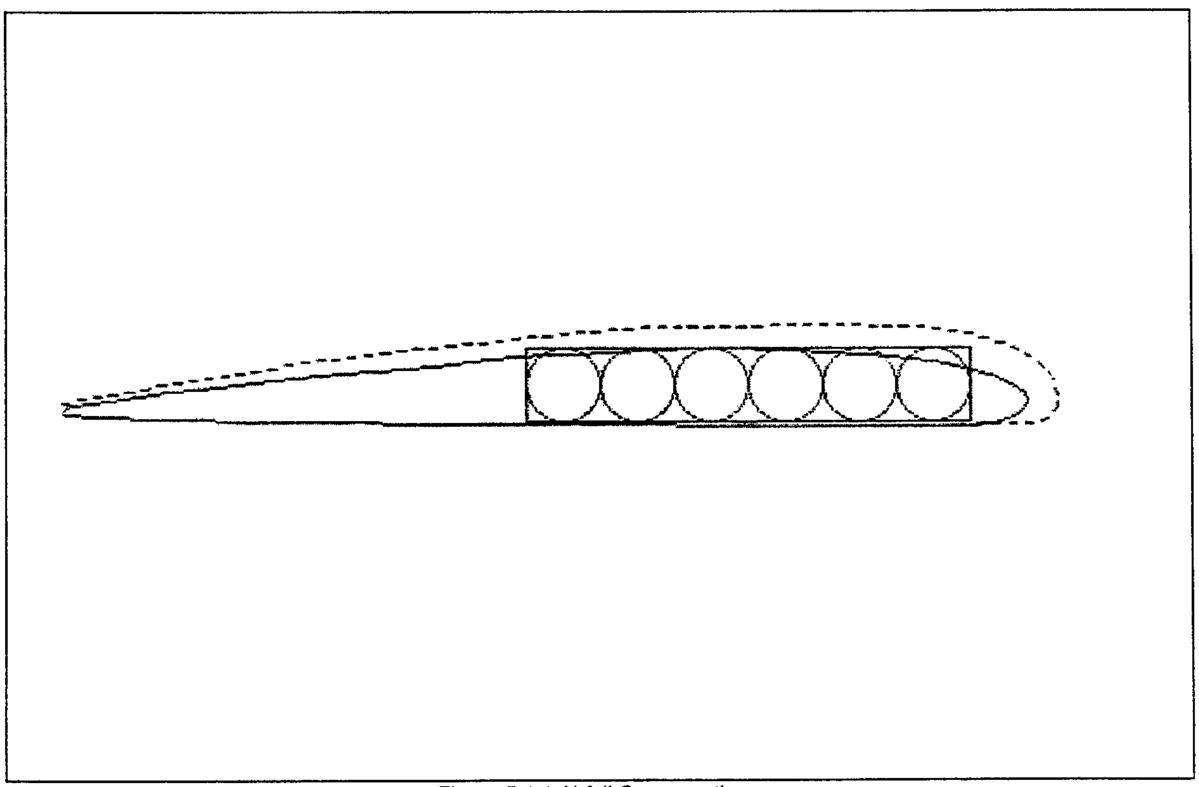


Figure 5.1.1 Airfoil Cross sections

After the airfoil selection process was completed the group could revisit the take-off calculations used as a primary design criterion during the preliminary design phase to assure that it would be met. The Balance spreadsheet was improved to incorporate the functions of the Airplane_Takeoff program so that

pitch stability and take off distance could simultaneously be reported for each planform adjustment made. Existing planform dimensions for the flying wing were used with the exception of the elimination of sweep. It was found that even with the preliminary dimensions, the flying wing was able to meet the takeoff criteria with the NACA 4412 airfoil. The wing and fuselage were both modeled with VLM and entered into Balance to obtain the takeoff distance result. Although takeoff distance was well within tolerance, the aircraft had no pitch stability due to the lack of a horizontal tail surface, sweep, and twist.

The next step was to attempt to provide the aircraft with pitch stability through the addition of a horizontal tail. With the same empirical relations used earlier on the typical aircraft, tail dimensions and locations were calculated for the flying wing. Another simplification was made at this point at the request of the manufacturing group to design a single rectangular surface rather than two trapezoidal sections. For simplicity, the airfoil section for both the horizontal and vertical tails was chosen to be symmetric. Both were selected to be NACA 0012 airfoils. It was discovered, however, that the horizontal tail could not provide pitch stability for the aircraft at any reasonable distance or inclination. The reason for the persistent instability was observed to be the location of the aerodynamic center of the wing in relation to the initial assumed CG location for the flying wing aircraft provided by the weights and inertia group. It was calculated to be several inches in front of the CG when for stability it should have been aft.

In order to impart pitch stability to the aircraft, the aerodynamic center (AC) of the wing would have to more aft in relation to the CG location. This could have been accomplished by several methods including adding weight to the nose, moving the fuselage section forward, or adding a slight sweep to the wing. Of these options, the addition of a slight wing sweep seemed to be the optimum choice. Therefore, enough sweep, 28° of leading edge sweep, was added to the wing to cause the wing's AC to become co-incident with the CG. The effects of the tail surface would then move the aerodynamic center of the entire aircraft to slightly aft of the CG position. Through the use of a horizontal tail and wing sweep pitch stability was achieved eliminating the need for complex airfoil shapes, high sweep, or wing twist.

With the aircraft now able to take off in an acceptable distance and to be stable in pitch, some optimization could be performed. It was noted that, due to the large surface area of the wing and fuselage, the flying wing was flying more slowly at its design lift coefficient that was desired. In order to increase the speed of the aircraft and to eliminate excess wing weight, the root chord of the wing was reduced to the minimum required to support the payload in center section. The center section airfoil was changed to a modified NACA 4412 so that it would better transition into the wings as well as help generate more lift. With these adjustments, the wing surface area was reduced and thus the flight velocity increased. Balance detected no adverse affects on pitch stability and calculated that take off would still occur within a reasonable margin. Even when the aerodynamics group was presented with a lower estimate of engine thrust the takeoff distance was well under the designated 200 feet.

With current estimates of planform, maximum lift coefficient (0.8395), and thrust (5.25 pounds), graphs were generated for the acceleration, velocity, and distance during the takeoff run. These graphs

appear in Figure 5.1.1. Takeoff calculations also required estimations of C_d and K. These were found using the VLM applet. A value of 0.02 was used as the ground friction coefficient.

Other analysis was also performed on the configuration. A graph was generated to describe the relationship between weight and takeoff distance for a 5.25 pound constant thrust. Another was generated to illustrate the relation between weight and thrust required for a 200 foot takeoff run. These graphs would allow the team to determine limiting values for total weight and minimum thrust. They appear in Figure 5.1.2. It was determined that the aircraft could weigh up to 40 pounds if provided with 5.25 pounds of constant thrust during its takeoff run. However, if weighing only 30 pounds as expected, the aircraft would only require 2.1 pounds of thrust per engine or a total of 4.2 pounds of thrust. These values would serve as guidelines until engine test results and refined weight estimations became available.

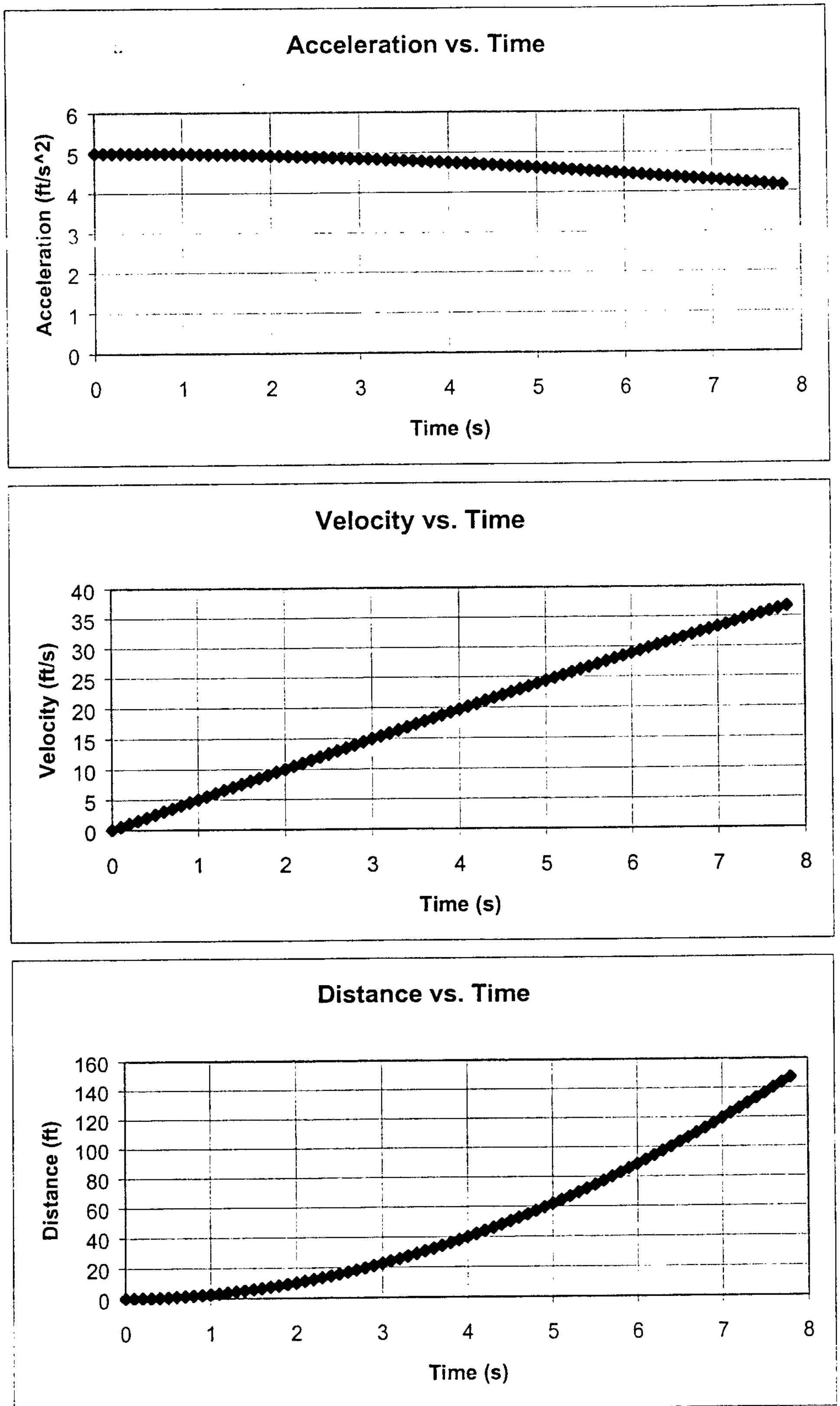
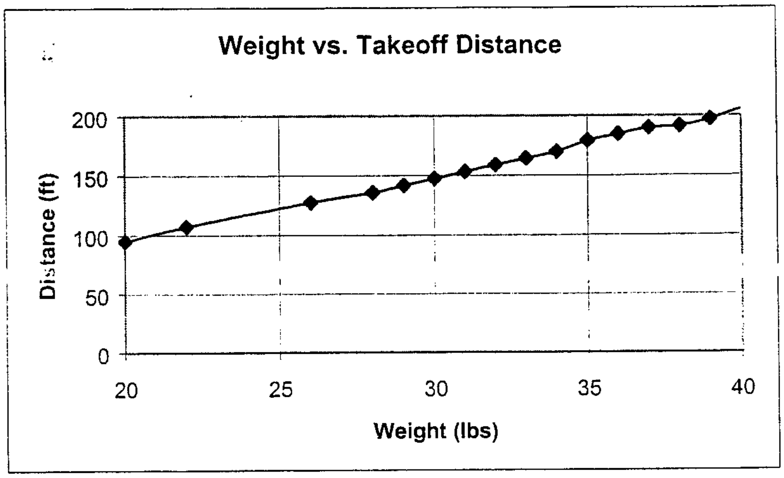


Figure 5.1.1 Takeoff roll estimations



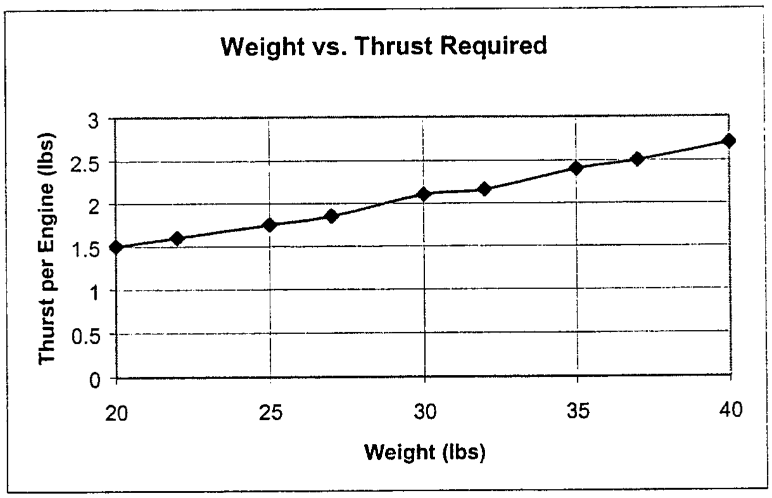


Figure 5.1.2 Weight Relations

The pitch performance of the aircraft was computed by again utilizing Balance as was done for the typical aircraft. Monitoring the behavior of the moment coefficient insured stability. Lift coefficients and flight velocities were also computed and appear in Table 5.1.1. The aircraft was found to have an equilibrium angle of attack of 4° with a C_l of 0.607 and a flight velocity of 27.8 miles per hour with no control surface deflection. However, no calculations had yet been performed to determine how long it would take the engines to propel the aircraft to such velocities in flight.

It would now be necessary to determine the appropriate elevator deflection to cause a flight angle of attack of 0°. This would be the process used to estimate elevator control power. As done with the typical aircraft's analysis, the last 25% of the horizontal tail was deflected and re-entered into the VLM applet. VLM output could then be entered into Balance to observe the pitch reaction of the aircraft. This was done until the point was reached where the elevator deflection angle could counteract the initial pitching moment of the aircraft. Deflections between –4 and 10° were analyzed. A tabulated summary of the result of the elevator control power analysis appears in Table 5.1.2. From this, it can be estimated that an elevator deflection angle of 5° would trim the aircraft into a 0° angle of attack flight path. Under that

condition, the flight velocity would be 38.3 miles per hour, over 10 miles per hour faster than with an undeflected elevator.

Table 5.1.1 Pitch Behavior of Flying Wing

Angle of Attack	C_m	C_l	V (mph)
7	-0.4052	0.8395	23.7289
6	-0.2605	0.7621	24.8799
5	-0.1159	0.6847	26.2265
4	0.0288	0.6073	27.8285
3	0.1735	0.5299	29.7756
2	0.3181	0.4525	32.2093
1	0.4628	0.3751	35.3685
0	0.6075	0.2977	39.6976
-1	0.7522	0.2203	46.1503
-2	0.8968	0.1429	57.3131
-3	1.0415	0.0655	84.6796

Table 5.1.2 Elevator Control Summary

Deflection	Angle of Attack	C_m	C_l	V (mph)
-4	7	0.0225	0.8214	23.9891
-2	5	0.098	0.6757	26.4016
0	4	0.0288	0.6073	27.8285
2	2	0.1043	0.4616	31.891
4	1	0.0352	0.3932	34.544
5	0	0.0747	0.3203	38.2734
7	-1	0.0054	0.2519	43.1567
10	-4	0.1238	0.0331	119.1771

Calculations were next performed to determine the acceptable range for the location of the center of gravity. Since the initial value of 18 inches was only a preliminary estimation, it became necessary to determine for what range of values the aircraft would remain stable. Balance was again modified slightly to now accept a value of CG location off of design value and to report the pitch behavior of the aircraft under those circumstances. The weights and inertia group had indicated that the CG location could be manipulated slightly by using battery location as a variable. The aerodynamics group could then proceed to determine not only the CG range but also the optimum CG location for pitch stability. A plot was generated to illustrate both the equilibrium angle of attack and the elevator deflection required to trim to zero angle of attack for a CG range of 0 to 23 inches. This plot can be seen in Figure 5.1.3. It was found that above a distance of 22 inches from the tip of the wing the moment coefficient became positive and thus the aircraft became unstable. This correlates to Balance's estimation of an aerodynamic center of approximately 22 inches. One result of this analysis was the realization that at a CG location of 14 inches no elevator deflection was required to trim the aircraft. Another was that at locations above 19 inches non-realistic angles of attack were reported. Because locations less than 12 inches would be difficult to achieve, a CG range of 12-18 inches would be a realistic bound.

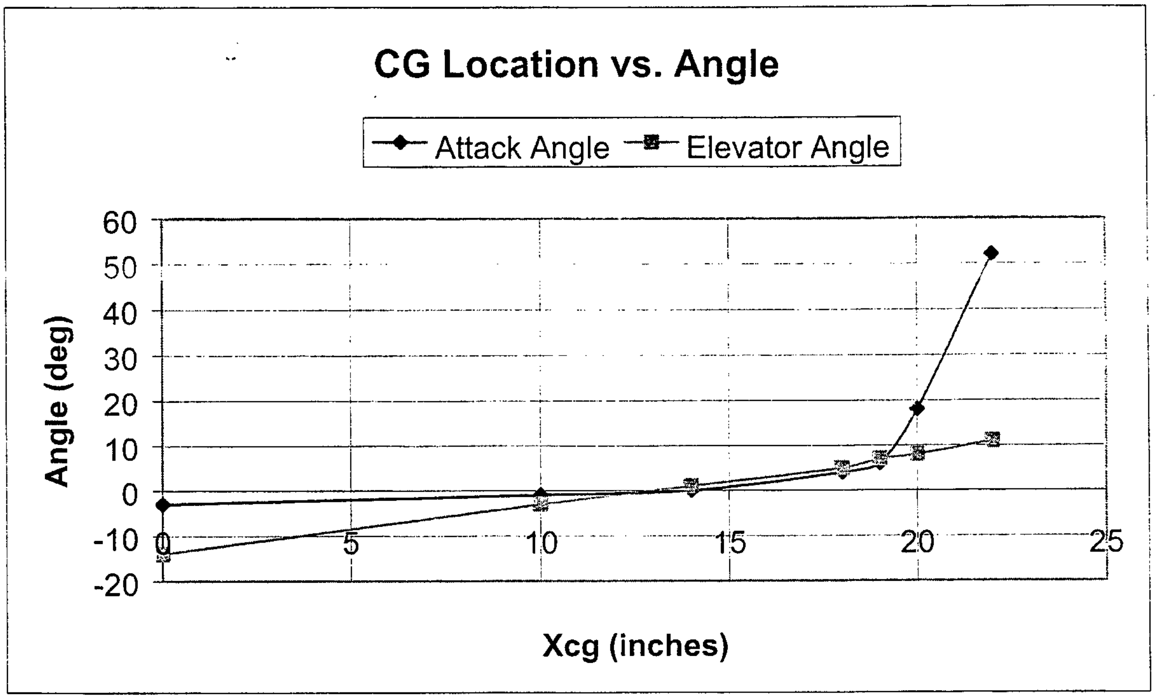


Figure 5.1.3 Center of Gravity Angle Effects

With the elevator dimensions and control behavior in hand, the group proceeded to design the other control surfaces. According to empirical relations found in the design text, the aileron sizing could be up to 80% of the span of the wing. The group chose to make the ailerons to the maximum size to provide maximum roll control and so that they might be able to function as flaps as well. The rudder was also sized with the aid of the design text; it covered the entire span of the vertical tail at 40% of the mean chord. Only one vertical surface was deemed to be necessary and it was placed at the center of the horizontal tail. The root chord of the vertical fin was set to the chord of the horizontal minus the elevator so that the movements of the elevator and rudder could not interfere.

Dynamic analysis of the final aircraft was investigated for steady flight conditions by using JKayVLM. The geometry of the aircraft is defined into a series of zero thickness trapezoidal sections. That program does not calculate effects of camber. The results of that analysis appear in Tables 5.1.3 and 5.1.4 below.

Table 5.1.3 Longitudinal Dynamic Results

Longitudinal Derivatives			
Derivatives	*Values*	*Derivatives*	*Values*
C_{L0}	0.00000	$C_{L\delta}$ [Aileron]	1.94813
C_{m0}	0.00000	$C_{m\delta}$ [Aileron]	-0.56090
C_{d0}	0.00000	$C_{L\delta}$ [Elevator]	0.29916
$C_{L\alpha}$	5.42564	$C_{m\delta}$[Elevator]	-0.57753
$C_{m\alpha}$	-0.13116	$C_{l\delta}$ [Aileron]	-0.41655
K	0.03897	$C_{n\delta}$ [Aileron]	-0.00636
C_{Lq}	7.59596	$C_{l\delta}$ [Elevator]	-0.01072
C_{Mq}	-4.55556	$C_{n\delta}$ [Elevator]	-0.00026

Table 5.1.4 Lateral Dynamic Results

Lateral Derivatives			
Derivatives	*Values*	*Derivatives*	*Values*
$C_{y\beta}$	-0.10192	C_{lp}	-0.49031
$C_{n\beta}$	0.03320	C_{np}	-0.06207
$C_{l\beta}$	-0.00491	$C_{y\delta}$	-0.04253
C_{yr}	0.08270	$C_{l\delta}$	-0.00273
C_{nr}	-0.03410	$C_{n\delta}$	0.01877
C_{lr}	0.00432		

5.2 Structures and Manufacturing

The structures group (SG) first task was to obtain an accurate measure of the material properties of the particular fiberglass being used by the DBF team. The preliminary design had been completed using the material properties found in the design textbook. The SG felt that these numbers were not accurate enough for the detailed design phase and that strength of material tests need to be completed. These tests were of paramount importance, particularly if the fiberglass being used by the DBF team was found to have a lower modulus of elasticity (E) and ultimate yielding stress (F_u) values. This could lead to a possible re-design of several key structural areas of the wing and, therefore, needed to be completed as soon as possible.

To determine the modulus of elasticity and ultimate yielding stress, five 1-in by 9-in specimens and five 2-in by 9-in specimens of six-layer fiberglass were constructed. These were of the exact build quality that could be expected during actual aircraft construction. Aluminum grip plates were bonded to each end and side of the 10 specimens. Each grip plate was 1.5-in in length and matched the width of the particular specimen. The specimens were then placed in a 5000-lb tensile test machine and stressed to the point of fracture. Stress and strain values were taken continuous during the experiment and then plots of stress vs. strain were generated for each specimen. An example of the type of graph generated can be seen in Figure 5.2.1.

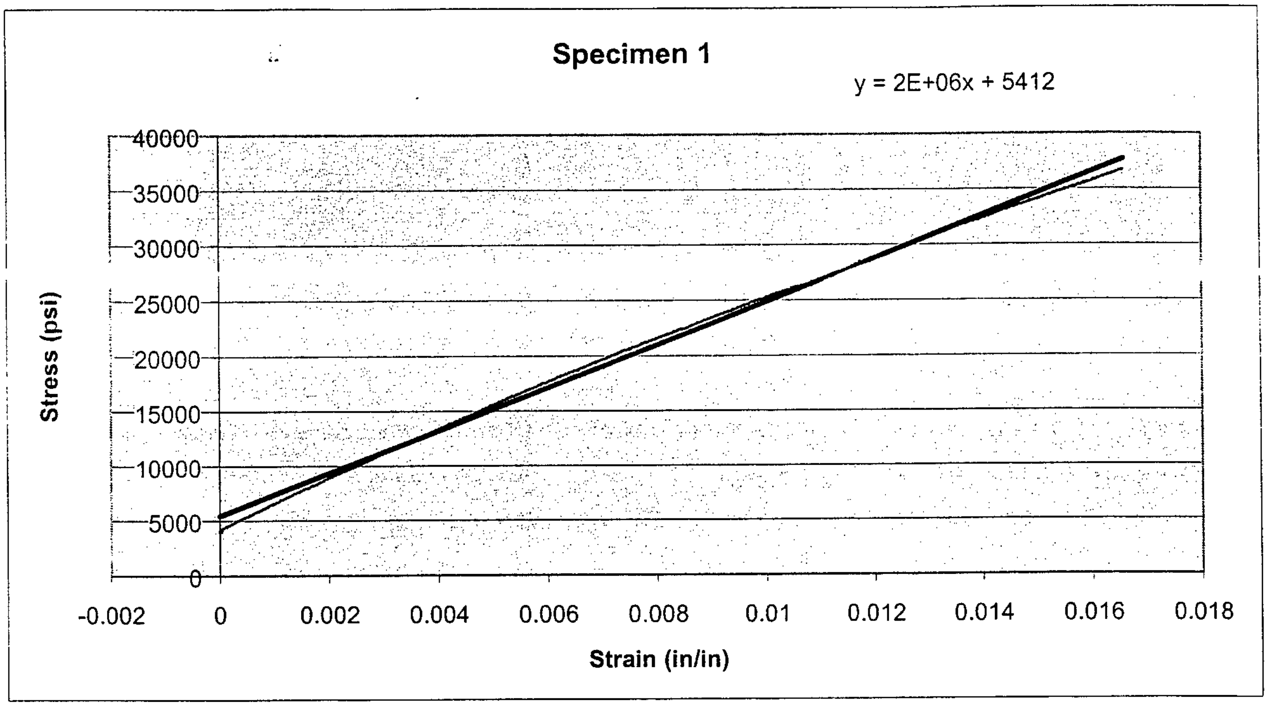


Figure 5.2.1 Stress vs. strain curve for specimen 1 (a 1-in by 9-in specimen)

The 10 plots produced were essentially identical with only the exact numbers of each plot varying slightly. The modulus of elasticity for each specimen was identical, $E = 2 \times 10^6$. This was well below the modulus of elasticity predicted by the Design Book for fiberglass – the predicted value was $E = 7.70 \times 10^7$. The experimental value was just 2.6% of the predicted value. This, obviously, represented a major modification to the internal structure of the wing. Armed with this new, more accurate value of the modulus of elasticity, the SG re-ran the wing structure program, WINGSTRUCTURE.M, which was written last semester (see section 4.2). The number of ribs needed to prevent cripplingly of the upper skin and the number of layers of fiberglass were all significantly increased. More importantly, however, the weight was increased by almost 10-pounds, from 31.1139-pounds (the loaded weight estimate at the conclusion of the preliminary design phase) to 40.237-pounds. This meant a serious deterioration of almost every performance and stability characteristic of the aircraft and was clearly not acceptable. Another method of wing construction would have to be realized.

The method that was decided upon, after much deliberation, was to use a type of sandwich structure. The SG would use 0.25-inches thick foam sandwiched between two layers of fiberglass. This would eliminate the need for ribs to combat the cripplingly of the upper skin (the skin material would now accomplish this) and rid the DBF team of the extra weight associated with those ribs (the weight of the foam was added). This would also easy the construction process because this was the method of construction most familiar to the group member with the most experience working with fiberglass. This also meant that the wing structure analysis program created last semester, WINGSTRUCTURE.M, would

have to be modified (or, re-written altogether – which is what ended up happening). Once the program was changed to be able to analyze the type of sandwich structure the SG now wanted to build, the improvements were clear. The weight of the aircraft dropped by 6-pounds to 33.9 pounds (although this was still more than 5-pounds over the preliminary phase estimate). This savings, although not huge, was significant and allowed many of the performance and stability characteristics to fall back to acceptable levels. The SG felt, however, that the predicted improvement needed to be compared against experimental data on a real wing (having learned a lesson from the modulus of elasticity experiment). The SG decided that a prototype wing needed to be constructed and that stressed to the point of failure. These experimental values could then be compared to the predicted values and the predicted values could be modified accordingly ('fudged' to better represent the details that the SG was not accurately modeling with the wing structure program). At the time of the writing of this report, the experiment was still underway and was yet to be completed. The results will be included in the addendum.

The second major task of the structures group was to design the center-section of the flying wing (the 'body'). This section would carry not only the 24 softballs but also the batteries and other necessary components. As mentioned in the preliminary design phase, the softballs would be broken up into 4-spanwise columns and 6-chordwise rows. The softball carrying portion of the body would be constructed of fiberglass and simply be a box with the necessary division made with fiberglass inserts (much like a fishermen's tackle box). This section would be accessible from above with a removable panel. An airfoil shape decided on by the Performance Group was wrapped around this section making sure to leave 4-inches on both sides in the spanwise direction for other components. This brought the overall spanwise length of the center section to 2-feet. The chordwise length is 3.5-feet (as dedicated by the Performance Group). The wings would be mounted to the center-section via 'hard points' built into the 4-in section on either side of the softball carrying portion. These hard points would match hard points on the wing and in this way the wings and center-section could be mated together with bolts. To accomplish this the center-section would have to be 0.25-inches thicker than the wings. This would allow the wings to slide into the center section. The hard points would be made of aircraft grade spruce wood bonded into the sandwich structure. The discontinuity created by having to step the center-section up 0.25-inches above the wings would be relieved by wedge-shaped transition piece bonded to the wings at the root. This can be seen in the drawing package.

5.3 Systems

The function of the systems group was to develop and test the propulsion and control systems for the aircraft, and the operations needed for cargo loading and unloading. The requirements for the propulsion system were taken from many different places. First, from the contest rules the motors had to be of the brushless variety, and the batteries to drive the motors had to have a total weight of under five pounds. Next, the aero-performance group required that the total thrust must be at least five pounds, and the systems group decided that the batteries must produce a full throttle current for at least seven minutes.

Finally, the motors that were available to the design team required a total voltage of the battery pack to be about 7.2 Volts.

The first step in the development of the propulsion system was the choice to use two motors. This decision was made because the systems group felt that five pounds of thrust from one motor was not feasible. With the decision to use two motors, the thrust required from each motor was cut to 2.5 pounds. Also, the weight for each battery pack was limited to 2.5 pounds.

The first step in the development of the propulsion system was the selection of a propeller. The requirements for the propeller were that it must produce at least 2.5 pounds of thrust for takeoff. It was found that the larger the diameter and pitch of the propeller the greater static thrust it produced. The decision was made to select a 16 X 7 inch propeller because it gave good static dynamic thrust.

The second step in the development of the propulsion system was the selection the battery pack. The type of cell was selected using the program to compare a host of different battery packs. The program showed that the higher the milliamp rating of the cell the longer it would produce enough power. Unfortunately however, the higher the milliamp rating the more the cell weighed. The constraints narrowed the list of viable cells to only a few choices. The SR 5000 MAX cell was the one with the longest full throttle current time and was chosen. This cell was rated at 5000 milliamp·hr and 1.2 volts. A pack constructed from seven of these cells gave the proper performance and an acceptable voltage for the motor, and fell within the limit of battery weight.

Another responsibility of the systems group was to test the chosen propulsion system. Information that was to be found or verified in this testing was the static thrust produced by motor, the dynamic thrust produced by the motor, and the length of time that the motor could be run at full throttle before the batteries lost their charge.

Due to some difficulty manufacturing a shaft to connect the propeller to the motor these tests were postponed until one can be made. The results will be in the addendum.

The layout of the servos to power the control surfaces was another responsibility of the systems group. The decision was made to put the servo as close as possible to the control surface that it would power. Because of this decision all servos were located within three inches of the leading edge of the control surface. Also, the systems group felt that the best spanwise location for the servo would be at the semi-span of the control surface. This was possible for only for the ailerons. The servos to for the elevator, and rudder were placed in the horizontal tail, on each side of the vertical tail. This was done to ease access to the servos and so they would counterbalance each other. The servo used to control the front wheel was located in the nose of the aircraft, near the front wheel.

The last of the systems group's responsibilities were to coordinate the payload operations. This included sizing the payload door, laying out the soft ball holders, and making a procedure to load and unload the balls.

The payload door was decided to cover the entire upper surface of the center section from near the leading edge to just aft of the ball holders. The payload door not only gives access to the cargo bay, but

also gives access to the propulsion system. The best way to secure the softballs was determined to be a fiberglass grid about four inches high with one softball to each square. When the payload bay door is closed, the grid will insure that the balls are properly restrained. A diagram of the ball holder can be seen in the drawing package.

To load the softballs, the propulsion system fuses must be removed to safe the aircraft. Next, the cargo bay door is opened. Next, a ground crew member will stand on each side of the tail boom and load soft balls into the ball holder. Then the door will be closed and the fuses reinserted. The method for unloading the balls will be very similar to that of loading the balls, with the only exception being that the balls are removed instead of being loaded.

5.4 Weights and Inertia

To predict accurate aircraft performance and stability, the Weights / Inertia group developed a Matlab computer program entitled "Inertia.m", and an Excel spreadsheet to estimate the aircraft's total weight, mass moment of inertias, and product moment of inertias.

First, "Inertia.m" receives aircraft dimensions, airfoil dimensions, and material properties for each part (ex. wing or vertical tail) as input, and then divides into small finite elements so that we can assume each element has a prismatic shape. After calculating the moment of inertias of each element, the parallel axis theory is applied with respect the origin, which in this case is located at the aircraft's nose. Not only does "Inertia.m" write an output file, "output.out", containing the results, but also plots the aircraft's planform and the center of mass locations in a 3-D graph. A sample of this graph can be found as Figure 5.4.1, which is located on the following page.

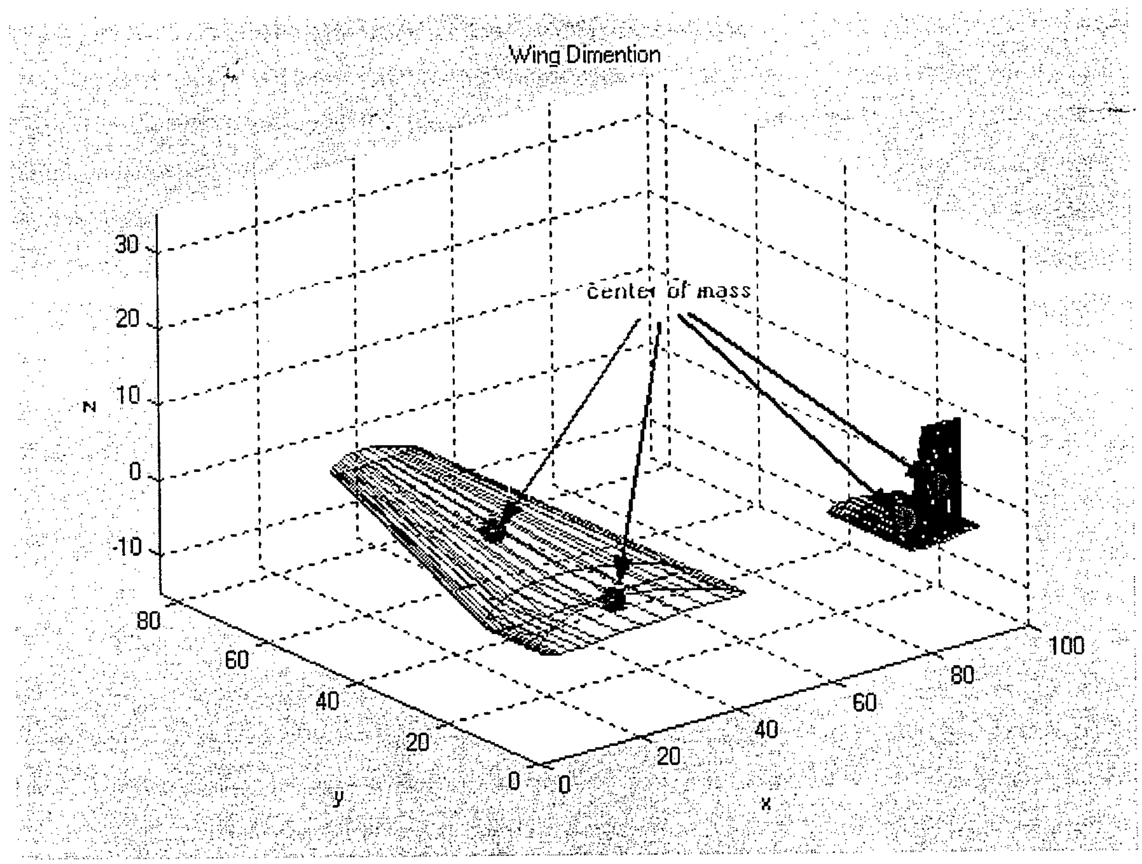


Figure 5.4.1: Sample output graph

Next, we measured the weight and dimensions of each component of the aircraft. For the components that were available, an electronic scale was used to obtain the weights. In order to get a weight estimation for the structure itself, sample pieces of the sandwich structure were used to determine the weight per area ratio, and as a result, it was possible to determine the weight of the foam/epoxy/fiberglass structure. Since it is very difficult to find the exact moment of inertia of a component about its center of mass, we assumed the shape to be a simple structure such as solid cylinder or rectangular prism in order to apply known the moment of inertia equations. Using the data and output from the computer program, "Inertia.m", the spreadsheet calculates the total weight, CG location of the aircraft, moment of inertia, and product moment of inertia. Below, in Table 5.4.1, is a portion of the Excel spreadsheet, which gives the CG location, and moments of inertias for the aircraft.

Table 5.4.1: Center of Gravity and Moment of Inertias

Moment of Inertia	(slug-in^2)	(slug-ft^2)
Ix	159.51816	1.107765008
Iy	870.51257	6.045226169
Iz	1540.7558	10.6996299
Ixy	0	0
Iyz	0	0
Izx	6.5545334	0.045517593

CG Location	(inch)	(ft)
Xcg	26.0805179	2.17337649
Ycg	0	0
Zcg	0.65545225	0.05462102

The total weight of the aircraft, including all the components obtained at this point, but excluding the softballs, was found to be 24.52 pounds. This results in a weight of 33.9 pounds when the softballs are included. According to the aerodynamics group, the X_{cg} location has to be less than 18 inches from the nose, so that the aerodynamic center is located aft of the X_{cg}. However, currently the X_{cg} location is far behind the aerodynamic center. Therefore, further concerns surface, and additional research in the areas of aerodynamics, structures, and layout must be performed.

6. Manufacturing Plan

The design of the aircraft has been carefully governed by the ability for the design to be manufactured in a quick and timely manner, as well as a cost effective manner. In order to complete the aircraft by the desired date, a manufacture's timeline has been constructed. In addition, processes have been chosen to facilitate the builders' abilities. Figure 6.1 shows the manufacturing schedule that will be followed to complete the aircraft.

In addition to building the aircraft in a timely manner, cost and availability of materials has also been greatly considered. In order to know that all materials are present, a Bill of Materials was created. Table 6.1 shows the Bill of Materials for the aircraft. Fortunately, the team has been supplied with enough funds to fabricate the aircraft. Additionally, many components needed for this project were available to the team from past projects, therefore it was not necessary to purchase them. These items include: transmitter, receiver, servos, engines, and landing gear.

Because the aircraft is more or less a composite "shell" the fabrication process will be quick and proceed with ease. The main structure, wings and fuselage will be constructed in six main panels. The panels will be constructed from ¼" foam core laminated with two layers of 3.74-ounce S-glass on each side. These panels include the following: left wing upper surface; left wing lower surface; right wing upper surface; right wing lower surface; center section top surface; center section bottom surface. The surfaces are numbered 1-6, respectfully. Each panel will be cut using a hot wire foam cutting tool with the templates provided. Laminating each panel takes a day per panel. This is because the team only has

enough resources for one curing at a time, this is mainly due to the fact that only one vacuum pump exists for the vacuum bagging process.

After the panels are complete, the corresponding upper and lower surfaces can be connected. The tail section and boom will be connected to the fuselage at this time. In addition, any needed service hatches will be cut after the panels have been joined together. These service hatches will provide the team access to needed sections of the airplane. The wings will not be permanently bonded to the fuselage. Instead, the wings and fuselage ends are made such that the wings will be attached via a set of nylon screws. This allows the wings to be detached when the need arises. This provides for easy transportation of the aircraft.

After the wings and fuselage have been assembled, installation of all other components will proceed. This includes engines, receivers, servos, batteries, wiring, landing gear, and payload compartments. The wiring will be laid out according to the wiring diagram that be found in Figure 6.3

The final step will include sanding the outside surfaces as smooth as possible and applying as many coats of paint as desired.

If all goes as planned, the aircraft should be able to be built in a matter of 16 days.

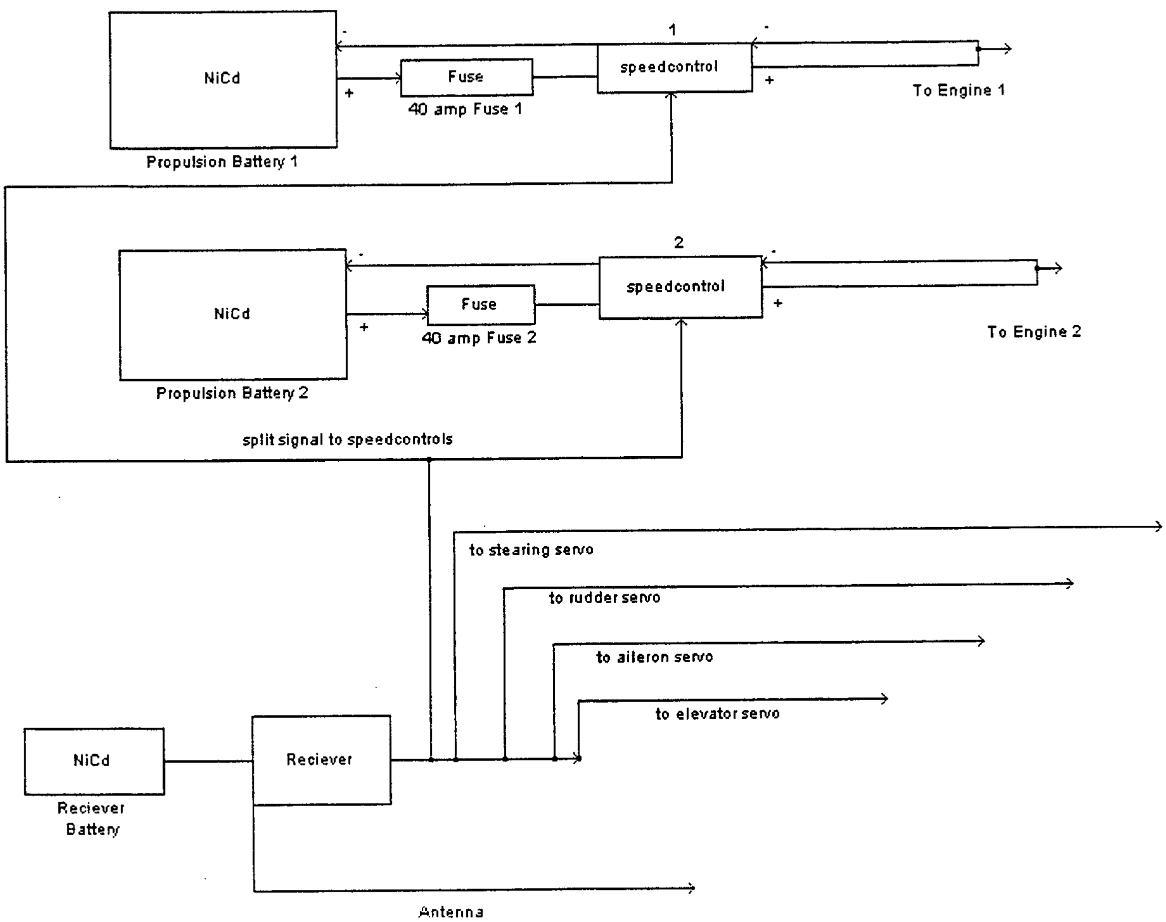


Figure 6.3 Wiring Diagram

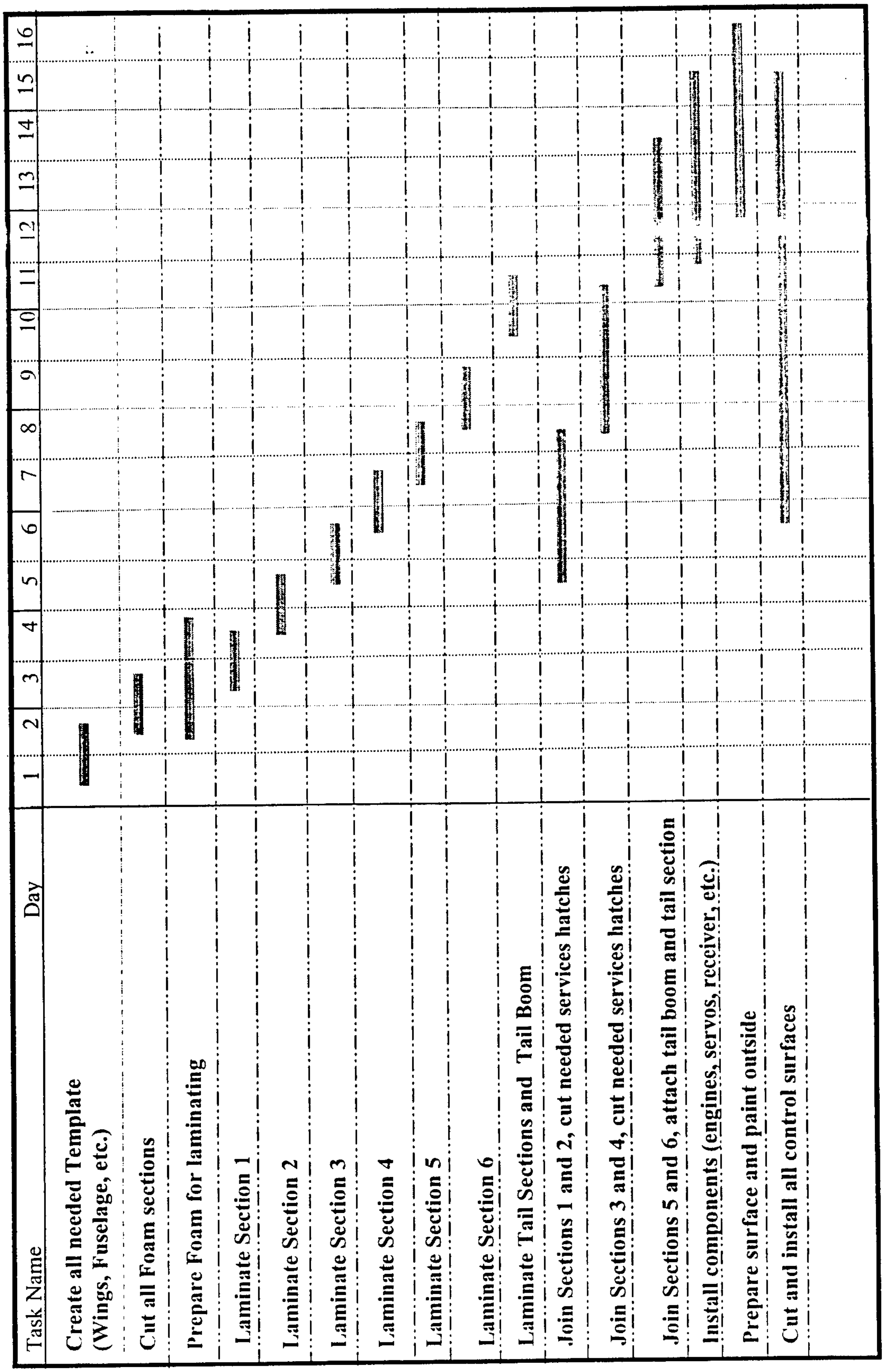
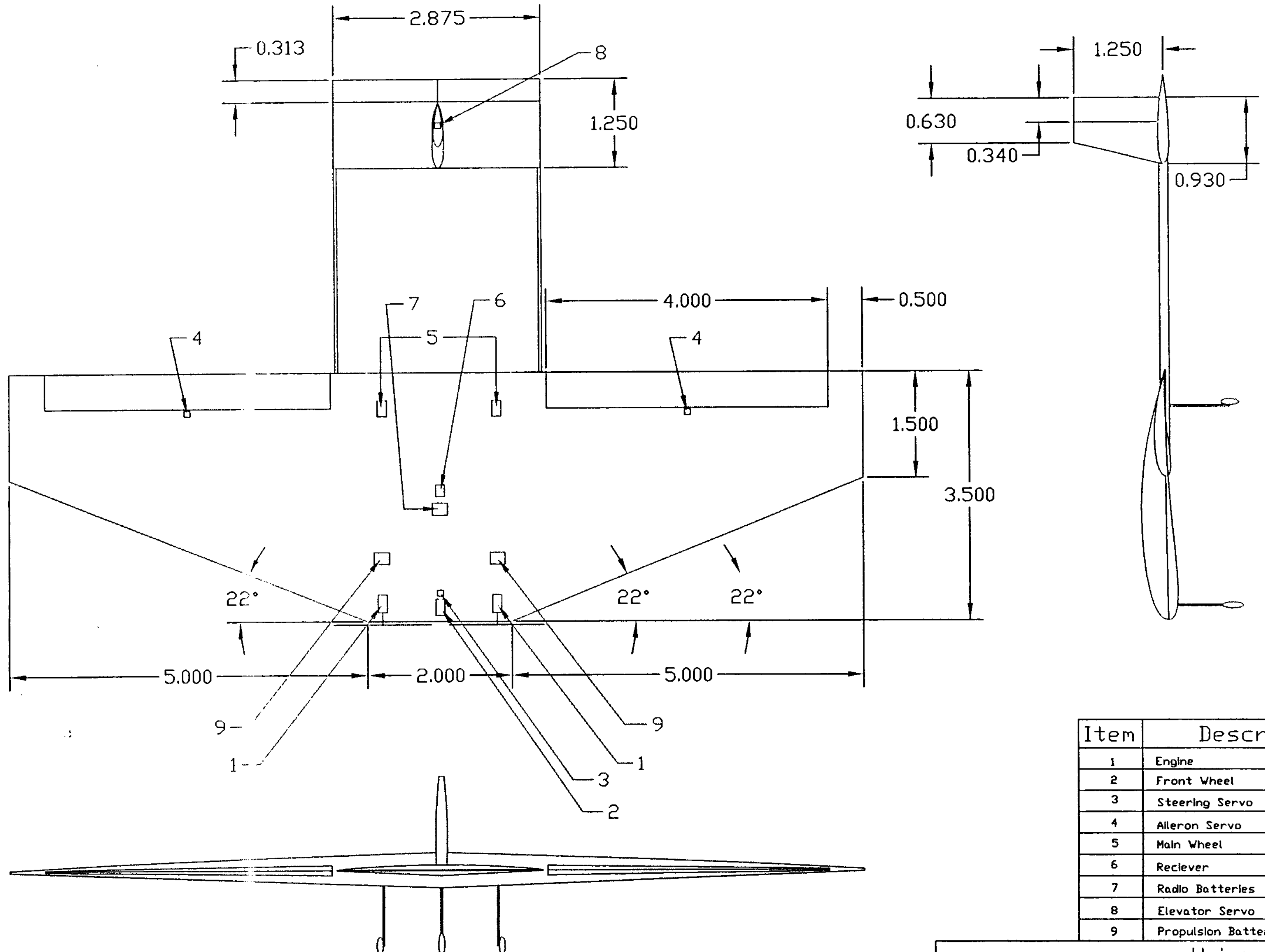


Figure 6.1 Manufacturing Schedule. This schedule illustrates time periods designated for each building phase.

Table 6.1 Bill of Materials

	Quantity		Cost per Unit	Total Cost
Construction Materials				
3/4" Foam Insulation Board	35		$9.23	$323.05
5.79 oz. Standard E-Glass Cloth	19	yards	$5.50	$104.50
5" Wide Graphite Fiber Tape	5	feet	$1.37	$6.85
West System Epoxy Kit, 205/206	1		$86.25	$86.25
Vacuum Bagging Film	15	yards	$1.11	$16.65
Perforated Release Sheets	15	yards	$3.25	$48.75
Vacuum Valve, 2 piece	1		$17.95	$17.95
Full Width Peel Ply	15	yards	$2.90	$43.50
Glass Bubbles	1	pound	$6.45	$6.45
Flocked Cotton Fiber	1	pound	$1.90	$1.90
Spray Adhesive	3	cans	$8.60	$25.80
2 x 2 ft 1/4" spruce wood	1		$5.00	
Aircraft Components				
Batteries	24		$7.50	$180.00
Copper Braid Nicad Interconnector	1		$6.50	$6.50
Digital Speed Control	2		$50.00	$100.00
Aircraft Wiring	20	yards	$1.50	$30.00
Frequency Module, Channel 36	1		$50.00	$50.00
Softballs (Competition Payload)	24		$2.50	$60.00
16 x 7 propellors	2		$6.00	$12.00
Graupner speed 600 Engines	2		*	*
Servos	5		*	*
Landing Gear	1		*	*
Receiver Batteries	1		*	*
model aircraft wheels and tires	3		*	*
NiCd battery charger	1		*	*
Tooling Equipment	10		$0.59	
Paint Brushes	2		$3.60	$5.90
Rubber Squeegee	1	package	$11.65	$7.20
50 Epoxy Mixing Cups	1	package	$8.45	$11.65
500 Mixing Sticks	5	packages	$0.85	$8.45
Sandpaper, 100 grit	5	packages	$0.75	$4.25
Sandpaper, 200 grit	1	package	$10.75	$3.75
100 Latex Gloves	1		$13.75	$10.75
Linear Spring Scale	1		$30.00	$13.75
Dremel Tool				$30.00



Item	Description
1	Engine
2	Front Wheel
3	Steering Servo
4	Aileron Servo
5	Main Wheel
6	Reciever
7	Radio Batteries
8	Elevator Servo
9	Propulsion Batteries

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UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE FOOT

Title ASSEMBLY

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MIT DBF Team:
Better than Jack

Presents . . .

The Blue Light Special

Executive Summary

The Better than Jack aircraft is the result of a team-rebuilding year. Given MIT did not participate in the AIAA Design Build Fly competition last year, all of the team's experienced members graduated and new people were not trained. The team name was chosen both in jest and to symbolize the team's determination to make it to the competition this year. The team consists of a core of four dedicated students with additional six or so part time members. With the guidance of team advisors and the determination of team members to learn how to build a plane, the team has embarked on what could be the greatest learning experience of the undergraduate experience. The following sections outline the team's adventure.

1.1 Conceptual Design Summary

Conceptual design centered on creative input, practical realizations and limitations put upon the team by supply availability. Early on in the design process the design was limited to using two Astro 25 engines. The team secured the two Astro 25's after failing to find any Astro 40's for sale. This procurement decision locked the design into a two-engine plane with limited power.

The team looked at single fuselage planes, but ruled them out based on the length required to carry a set number of softballs. The single fuselage idea was referred to as "The Bus." In the end, the rated aircraft cost along with the structural weight required to make it stiff made this design infeasible. Other ideas followed including separate fuselages. Given the aircraft cost rubric, having several smaller fuselages yielded a higher score than one longer fuselage. A symmetric puller aircraft was considered. Again, the necessity to "beef-up" the fuselage to carry the mass of the motor made this design less attractive. Later rule clarifications helped chose the final configuration: a plane with three fuselages: two payload fuselages with little structural strength and a main structural fuselage to hold the batteries and motors. Also, a pusher/puller configuration was chosen to allow both motors to be attached to the same pod. From this point, the next step was aircraft sizing and materials selection.

The predominant materials used for construction are composites such as graphite epoxy and Kevlar. Chosen for its high strength-weight ratio, composites were also the natural choice given their availability within the department. Kevlar in particular was available to the team free of charge.

During the conceptual design phase, the design of the payload carrying fuselages was also considered. Small mockups of fuselages were constructed and a final internal payload configuration was chosen at this stage.

1.2 Detailed Design Summary

Given the aircraft configuration, detailed design and sizing began. Using codes write by past years' teams to determine take off distance, MotoCalc to determine power available and Xfoil and AVL to determine a plan form, a final aircraft size was chosen. The final aircraft was sized to carry 16 balls, have a wingspan of 96 inches and a gross takeoff weight of 23lbs. Xfoil, an airfoil design program, was used to design the airfoil used. Ahyow completed this task under the direction of Prof. Drela. The use of AVL (Athena Vortex Lattice) allowed for the design of a near elliptical lift distribution. AVL is used to predict inviscid lift and drag for a 3-D plan form. The end result was a wing 96 inches in span composed of six tapered sections, three on either half with a root chord of 14.25 inches and a tip chord of 7 inches. The Plane Geometry program (an Excel script) was used to size the v-tail positioning and size given the wing design. Since motor selection was made by default, the battery was chosen to maximize voltage given weight and current limitations. The propeller was chosen after running extensive MotoCalc simulations of the takeoff and flight sections of the competition. MotoCalc takes information about the motor, batteries, and propeller, as well as the lift, drag and weight of the aircraft. With this information, MotoCalc can calculate the power left in the battery after each leg of the competition (i.e. takeoffs and laps with and without payload). The final propeller chosen maximized the flight score given the aircraft configuration by assuring the aircraft would have sufficient power to takeoff and maintain speed in the later laps of the competition run.

Spar sizing was completed using a combination of quick tips offered by the Charles River RC Club and advisor Prof. Drela. The spar was slightly oversized to accommodate the possibility of carrying more payload or even later changes to the wing. Tapered spar caps were used instead of tapering the spar itself. Because the entire wing was tapered about the quarter chord, the spar was also able to manufactured as a single straight beam. This simplified construction greatly.

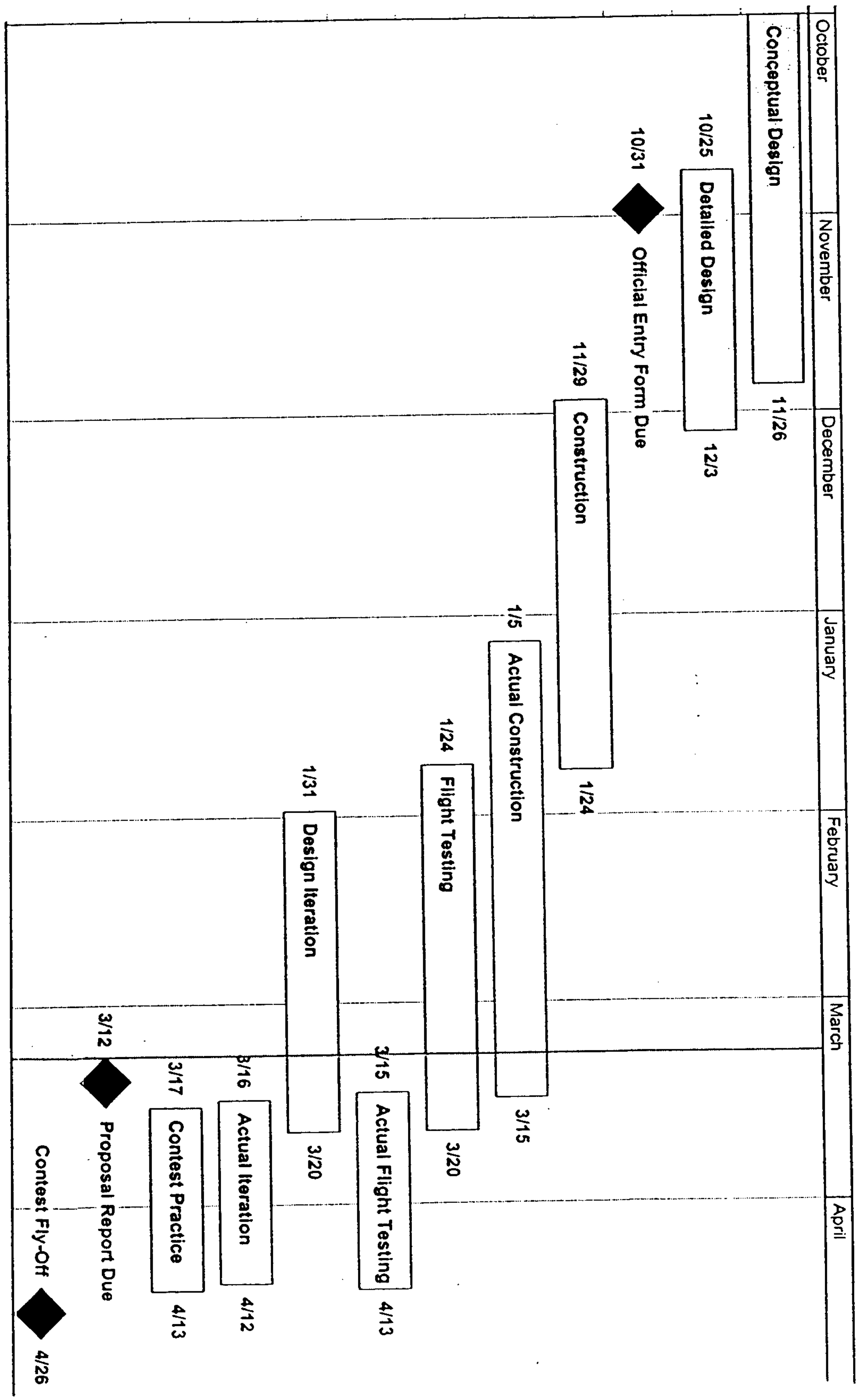


Figure 1.1 Project Schedule.

Management Summary

The "Better than Jack" team is operated as an extra-curricular activity. Participation is completely voluntary and all organization is organized through the AIAA-DBF chair, an elected position within the MIT chapter AIAA executive board.

2.1 Personnel and Tasking

MIT's "Better than Jack" team consists of four dedicated members. Three juniors, Richard Cross, Adam Diedrich, and Caroline Twomey, and one sophomore, Jack Willard. Bernard Ahyow and Larry Basket, both graduates were often referred to for advice, especially in the manufacture of the airplane—"Blue Light Special." The design team consisted of the aforementioned four undergraduates and a handful of other causal team members. Design was carried out in a casual manner, taking ideas solicited by member and then comparing relative designs using the team's figure of merit (FOM's) and the rated aircraft cost.

Diedrich brought to the team prior design build fly experience as a participant in the 2000 competition. Diedrich, with the help of Ahyow, performed the bulk of analytical calculations pertaining to the design of the aircraft. Twomey brought to the team organizational skills and some past experience in composite manufacturing. Cross added color and humor to the team in addition to being a quick learner, running manufacturing later in the project. As a sophomore, Willard is the team's best bet for a future leader. Willard joined the team with his enthusiasm and will to learn. He has been involved in every aspect of design and manufacture.

Diedrich headed up all preliminary design work and helped out extensively with the manufacturing of the airplane. Cross worked extensively on detailed design and manufacture of the plane. In addition, Cross acted as the team's treasurer, keeping the team on budget. Twomey worked on manufacture, procurement of supplies and workspace, as well as detailed design. Twomey also worked hard to keep the team on schedule, and later worked to help delegate tasks to get the team back on schedule. Willard worked especially hard on wing manufacture and has become quite proficient with regards to the wing.

Team advisors Col. Peter Young (A.F. Ret.) and Prof. Mark Drela provided practical advice and assistance during design and construction. Most notable, Col. Young helped the teamwork around supply distribution issues regarding the purchase and hipping of the wing spars. Thanks to his help, a shipping error was caught and the spars were eventually received.

2.2 Management Structure and Scheduling

Twomey and Diedrich managed team personnel and scheduling. The team set benchmarks for design and manufacturing and then worked towards them by keeping a task chart in the work area. Once a task was completed, the person who completed it checked it off and put down any new tasks. During January, the team offered a part time class, facilitating a daily team meeting and construction session. During this time, the bulk of manufacturing was completed. However, due to difficulties getting the wing spar, wing construction was delayed, further delaying the entire construction schedule.

Figure 1.1 outlines the original schedule, the new schedule and changes between the two. The greatest setback was the inability of the team to construct the wing during January term, an independent activities period during which regular classes are not offered. As of the writing of this report, final integration and testing are all that remain. The subsystems are all completed, save the landing gear, which are going through design finalization.

3 Conceptual Design

3.1 Systems Configurations Investigated

Alternate configurations investigated included "The Bus" and the symmetric puller. "The Bus" consisted of one main fuselage inside of which the payload, single motor and batteries were enclosed. "The Bus" had a total length of approximately 85 inches assuming a set payload of 16 balls. Its wingspan was also set at 96 inches, the span used for each configuration investigation. The second configuration investigated was the symmetric puller. The symmetric puller had two fuselages, each with a puller motor mounted on the front. The advantage of this configuration was the reduced length, approximately 65 inches. The symmetric puller was also sized to carry 16 balls and have a wingspan of 96 inches. The final configuration looked at was the tri-fuselage configuration. In this configuration, a small center fuselage contained two motors, a puller and a pusher, along with the batteries and radio receiver. Two fuselages set out on the wings contain the payload. The benefit of having only the balls in the payload fuselage is that the payload fuselage can be made non-structural. This means, that in addition the decreased length (by about 3 inches) the large payload fuselages can be made of lighter, thinner material.

As is shown above, for simplicity and comparison purposes, the design parameters of payload and wingspan were held constant across the designs configurations investigated. The design parameters varied were length, number of fuselages and number of motors/configuration of motors.

3.2 Analytic Methods Used

The analytical methods utilized included the use of MotoCalc, a MATLAB simulation, Xfoil, AVL, and the Plane Geometry Excel script. These analytical methods allowed for a number of trade studies to be conducted.

MotoCalc, a commercially available software package for the PC, was instrumental in designing the motor/propeller/battery system. MotoCalc is a tool used to provide estimates of electric-powered R/C aircraft performance including power usage, power remaining, and maximum speed to name a few. MotoCalc allows the user to vary components such as the batteries or propeller used to optimize the propulsion system for the given task—in this case three takeoff and landings and six laps, two of which are with payload.

Several propeller combinations were analyzed and the final decision was to use a 13" x 10" propeller. This propeller produced the optimum combination of top speed (necessary given the weight of the plane) and battery life.

Using the results of MotoCalc, the next piece of software used was a mission simulation developed in MATLAB. The simulation allows the user to simulate different legs of the competition including takeoff, climb, cruise, and landing. Most importantly, the MATLAB script is able to predict the takeoff distance of the plane—thus dictating the overall lift required. See Appendix A for the source code

corresponding to takeoff module of the simulation. Results of the takeoff code are discussed in more detail in the detailed design segment of this report.

XFoil and AVL were used at a higher level. The airfoil used was developed for us by Ahyow and Prof. Drela using XFoil and is optimized for low Reynolds number flight. Ahyow took the lift and takeoff distance requirements and designed a planform for the Blue Light Special.

The Plane Geometry Excel script was then used to size the tail surfaces. The Plane Geometry script takes information about the wing and fuselage placement and tail configuration and sizes the wing given its placement. Unfortunately, the Plane Geometry script is not able to accept model multiple fuselages. As such, the model inputted modeled the three fuselages as being in the same place.

The combination of the above analytical tools allowed for the comparison of the three proposed designs. The final decision was made based on supply availability and the FOM's as decided by the team and explained in the next section.

3.3 Figures of Merit (FOM)

The figures of merit chosen to evaluate the plane are as listed below:

Figure or Merit	Weight Assigned
1. Number of Balls Carried	10
2. Ease of Manufacture	9
3. Minimum Rated Aircraft Cost	10
4. Modularity of Design	6
5. Time to Build	7
6. Simplicity of Design	6
7. Aesthetics	3
8. Maneuverability	4
9. Cruise Speed	7
10. Originality of Design	3
11. Minimum Weight	7
12. Easy to Fly	8
13. Durability (Resists Hanger Rash)	6
14. Payload Access	7

Each FOM relates to a different aspect of the plane's design, construction, or performance. FOM's affecting design include the payload access, minimum rated aircraft cost, number of balls carried, simplicity and aesthetics. FOM's related to construction include ease of manufacturing, modularity, time to build, simplicity, and minimum weight. FOM's related to performance include maneuverability, cruise speed, durability, and ease of flight. Of course, ease of flight is closely related to maneuverability.

Maneuverability is meant to cover turns, takeoff, climb and landing, while ease of flight is targeted more at stability.

3.4 Rated Aircraft Cost (RAC)

The rated aircraft cost model given for the competition is a function of the number of engines, battery weight, empty airframe weight, a variety of length measurements, and the number of control surface's servos and propellers used. The advantage to "The Bus" is that is minimizes the number of engines and propellers. However, "The Bus" has the longest longitudinal length by nearly 50% of the three proposed designs. Also, the available engines do not have enough power to propel an aircraft capable of carrying 16 balls in a single engine configuration. The symmetric puller has a reduced longitudinal length—making it a significant improvement over "The Bus." However, the symmetric puller pays a penalty in RAC for its two engines and two propellers. Given the engine supply issue, the use of two engines is inevitable. Therefore, the benefit of reduced longitudinal length greatly outweighs the penalty of having two engines and twp propellers. The third design, the pusher, puller concept that has come to be called "Blue Light Special," has a slightly reduced longitudinal length as compared with the symmetric puller. In addition, Blue Light Special also has the potential for a reduced empty weight because the payload fuselages are no longer structural members and can therefore be made lighter.

3.5 Final Configuration Decision and Justification

The final configuration chosen is the Blue Light Special configuration. Given the assumptions that by running the middle of the line on all the performance metrics, the maximum score can be achieved, the Blue Light special is designed to carry a moderate number of balls, 16, fly at a moderate speed, and have moderate length and wing span. The key here is choosing something that is original but not so revolutionary that it is a risky proposition. Other assumptions made in the design process include the ability to accurately model the lift and drag of the wing using the inviscid AVL code, the ability to accurately model battery drain using MotoCalc, and the ability to accurately model takeoff distance using the MATLAB simulation. The justifications for these assumptions come from their proven validity in past DBF planes. The design process used for Blue Light Special closely follows that used on MIT's past DBF planes.

As can be seen in Figure 1 the Blue Light Special, there referred to as the Pusher/Puller, wins out over the other designs when evaluated using the FOM's chosen by the team. Each configuration was evaluated for each FOM on a scale from 1-5 with five being the best. The final totals were then divided by five for purposes of comparing smaller numbers.

Features incorporated into the final configuration include the use of three fuselages, two motors, a pusher/puller configuration, and a unique ball loader constructed of rails that channel the balls into the fuselage and allow for easy removal.

Figure 3.1 Figures of Merit Chart Comparing Three Designs

Figures of Merit

	FOM	The Bus	Twin Fuselage	Pusher / Puller
Number of balls carried	10	4	4	4
Ease of manufacture	9	4	3	3
Minimum RAC	10	2	3	3
Modular	6	2	3	4
Build time	7	4	3	3
Simplicity	6	5	3	3
Aesthetics	3	2	3	5
Maneuverability	4	2	3	3
Speed	7	3	4	4
Originality	3	1	3	5
Minimum Weight	7	3	3	3
Ease of piloting	8	4	3	4
Durability	6	3	3	4
Payload Access	7	2	3	5
Totals		57.8	59.2	68.4

4.0 Preliminary Design

The preliminary design stage built on the ideas of the conceptual stage and provided a structure with which to evaluate those ideas. The large number of resources open to the team helped differentiate feasible and infeasible configurations. The team used textbooks, the department library, professors, computer simulations and the team's compiled experience to sort through ideas and produce a working model within budget and on schedule. Although no design process can be fully complete, this design was complete to the final stage of production with minimal changes and produced a complete aircraft on cost and close to on time.

4.1 Major Parameters and Trade-offs

Combining the design of "The Bus" and the symmetric puller resulted in a final configuration of pusher/puller. The team's first design decision was to choose two fuselages to hold payload. The team choose "The Bus's" center mounted engine, but since the available engine did not provide the required power, a pusher/puller configuration was adapted. Minimizing weight for this configuration required careful selection of materials. "Space-age" composites often provided a huge benefit over conventional materials but were not used in every situation as some building applications required more conventional materials either because of their material properties of the ease with which they can be used. Standard blue insulating foam and balsa wood are as important to the aircraft structure as Kevlar and carbon fiber.

4.1.1 Number of Fuselages

The largest discrepancy between "The Bus" design and the symmetrical puller is the number of fuselages. In choosing between multiple and a single fuselage, the team considered number balls carried, RAC, and aerodynamics. Instrumental in choosing the multiple fuselages over the single fuselage was the Excel Plane Geometry script. It provided a clear view of structure and produced important parameters that allowed analytical comparison of the choices. Skin stress was also of major concern. The fuselage hollow body transmitted stresses through its skins. As diameter increased, skin thickness increased to restrictive levels. The team discovered the aerodynamic loadings would be better balanced by multiple structures. RAC forced overall length of into the considerations. Increased payload required a longer structure. This could be mitigated with multiple structures. Aerodynamic differences between one and multiple structures proved to be negligible as compared to these other factors.

4.1.2 Engine Configuration

The second largest design decision involved engine number and configuration. Since the desired engine was unavailable, the team espoused a secondary design. Using MATLAB scripts to simulate the race and MotoCalc, the team determined two engines were necessary. Mounting

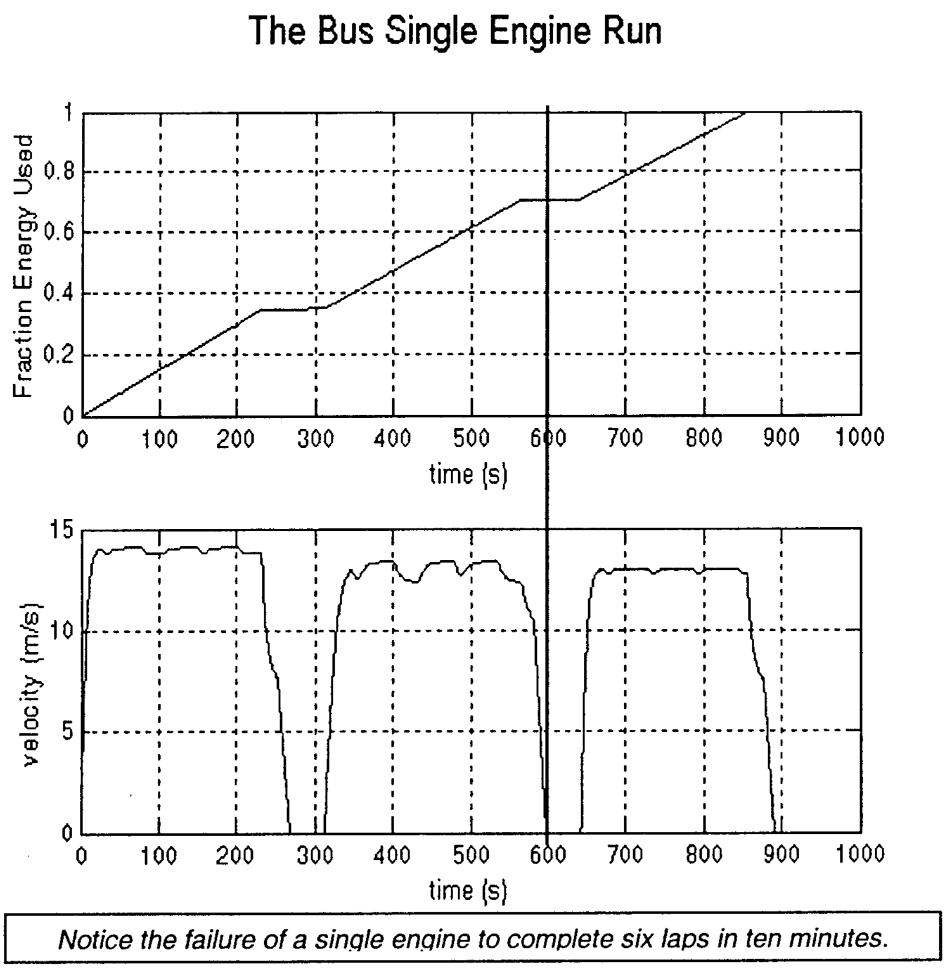


Figure 4.1 The Bus fails to finish the mission.

the engines to the front of the payload pods provided to difficult to build and vastly inferior to a center mounted engine. This problem elicited the most unique solution and set the tone for the rest of the design. The team opted for a pusher *and* a puller motor acting from the same center fuselage. Our team chose this to harness the power of two motors and complement the multiple fuselage design. Other solutions proved infeasible to build or insufficient in power.

4.1.3 Material Selection

Well-selected materials were key to design choices. Most of the team has experience building with composites. This knowledge helped the team use composites often and well. Wing strength was increased with a carbon fiber and balsa spar. Fiberglass skins increased stiffness and foam provides shape. Our fuselages are constructed of Kevlar and unidirectional carbon fiber. These materials are extremely useful in unidirectional and bi-directional loading such as experienced by fuselage skins. Two special pieces are the spar caps and the tail boons. The spar caps are linearly tapered to the tips and

provide huge tensile strength to the wing. The tail boons are spiral wrapped carbon fiber, extremely strong in torsion and bending.

4.2 Meeting The FOM's

The Figures of Merit arrived at in the conceptual design stage drove ideas and produced a unique, creative design that meets or exceeds all competition requirements and puts this team in a competitive position for the up coming contest. The FOM's helped clearly defined the upcoming problems. Once problems were defined, solutions were gleaned from the preliminary design that most closely lined up with FOM's. This also allowed the team to divide up the workload between sub-teams. Each sub-team presented primary, secondary, and tertiary designs for their sections to the team as a whole. Sub-teams were composed of groups of interested team members. Often one team member would serve on several sub-teams. These individual designs were compiled into a design that maximized benefit for the entire structure. By reducing independent variables to a few sections of three each, the team clarified a complex picture and simplified choices.

4.2.1 Number of Balls Carried

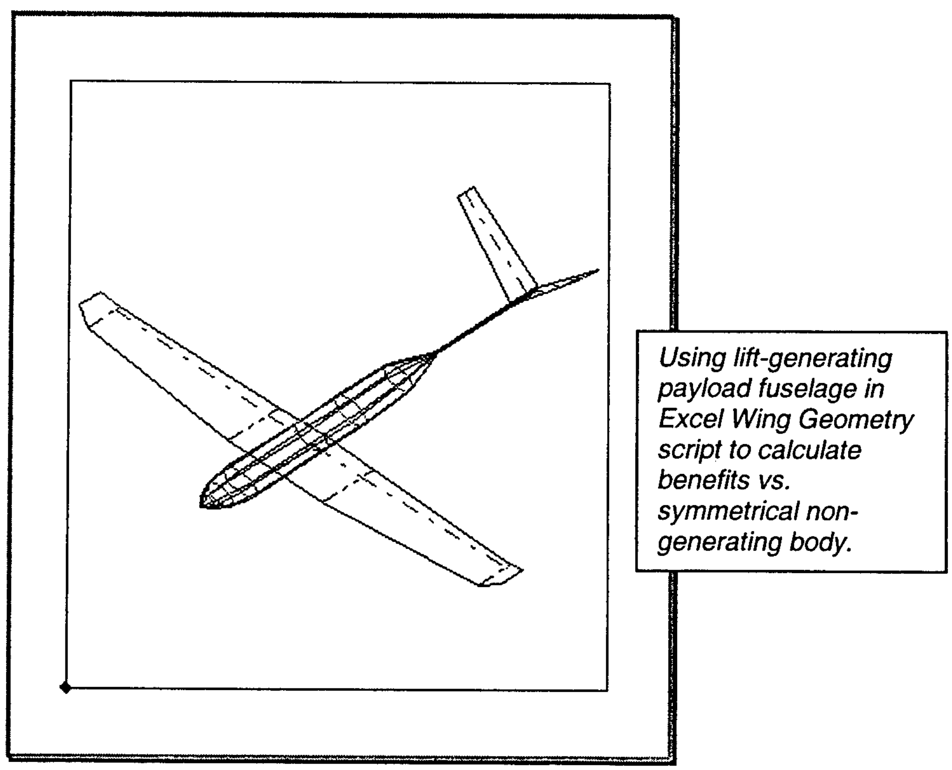


Figure 4.2 Single Fuselage Prototype.

A primary concern was exactly how many balls should be carried. The number of balls would determine the scaling of the plane. Initial advice from professors pushed the team to the middle range of possibilities. It was important to consider weight of balls and their aerodynamic cross-section. The team obviously needed a structure to carry the balls with the least amount of energy lost to weight and drag. This led to an interesting conclusion. The most efficient structure would be lift-generating bodies. Instead of symmetrical payload structures, the team designed two wing-mounted pods that would produce lift. This in turn required the development of a construction process for these bodies. Plaster molds were made from hand formed dies for the final lay-up.

4.2.2 Minimize Rated Aircraft Cost

Throughout the design process, the team kept the cost calculations near at hand. At least one major design idea came out of this. The team opted to use a V-tail empennage to reduce Manufacturing Man Hours. The cost of a conventional tail for the airplane would be 40 Man Hours (two vertical surfaces and two horizontal surfaces given the twin tail boom design) while the cost of a V-tail is only 15 Man Hours. With two less servos at 5 Man Hours per servo, the team saved 35 Man Hours or $700 according to the Aircraft Cost Model.

4.2.3 Ease of Pilot Control

From previous competitions, the team members have learned that the winds of Wichita, Kansas can demand superior piloting for the most stable craft. The team tried to compensate for the instability added by the V-tail with a more stable wing configuration. Therefore, the team added three degrees of dihedral to the outer wing sections and angled wingtips.

The team purchased tapered, wrapped carbon fiber tubes to extend from the payload bodies to the V-tail to minimize deflection. The team also used large ailerons to further increase airplane response.

The team consulted with Col. Young and Prof. Drela, both experienced pilots, and asked for suggestions. Many of their suggestions can be seen on the final airplane as small changes that make a big difference. The team purchased a large R/C airplane to begin training the pilot while the airplane was in production.

4.2.4 Speed

Of course, speed is a large concern. The aircraft, designed around a rather large payload and a limited power source, is a slow, heavy lifting aircraft. To gain speed, the team attempted to reduce battery consumption as much as possible. The most useful tool in designing an optimized aircraft was MotoCalc. Running race simulation scripts the team optimized for speed and battery drain at different loads to determine the theoretical optimal speed.

4.2.5 Minimum Weight

Weight was the easiest parameter to design around. Material properties and simple analysis gave sound insight into the stress around the airframe. The materials of choice were balsa, foam insulation, fiberglass, carbon fiber, and Kevlar. The best example of weight minimizing efforts is the main wing. Constructed of six different materials, the wing is light, yet strong enough to support the entire airplane with lifted at the wingtips. The backbone of the wing is a laminate spar at the quarter-chord. Constructed of tapered carbon fiber in shear and end-grain balsa in compression, the spar incorporates two entirely different materials in a beautiful harmony. Lightweight foam is used to provide an airfoil shape and fiberglass arranged at 0-90 and 45-45 holds the wing in shape. Paint, on Mylar slicks, is the final component, arranged so the pilot can easily determine the attitude of the aircraft. By researching good construction techniques and referencing experienced builders, the team was able to use advanced building techniques that saved a large percentage of weight.

4.2.6 Payload Accessibility

Payload was the word of the day for this competition. Not only did a lot of it have to be carried

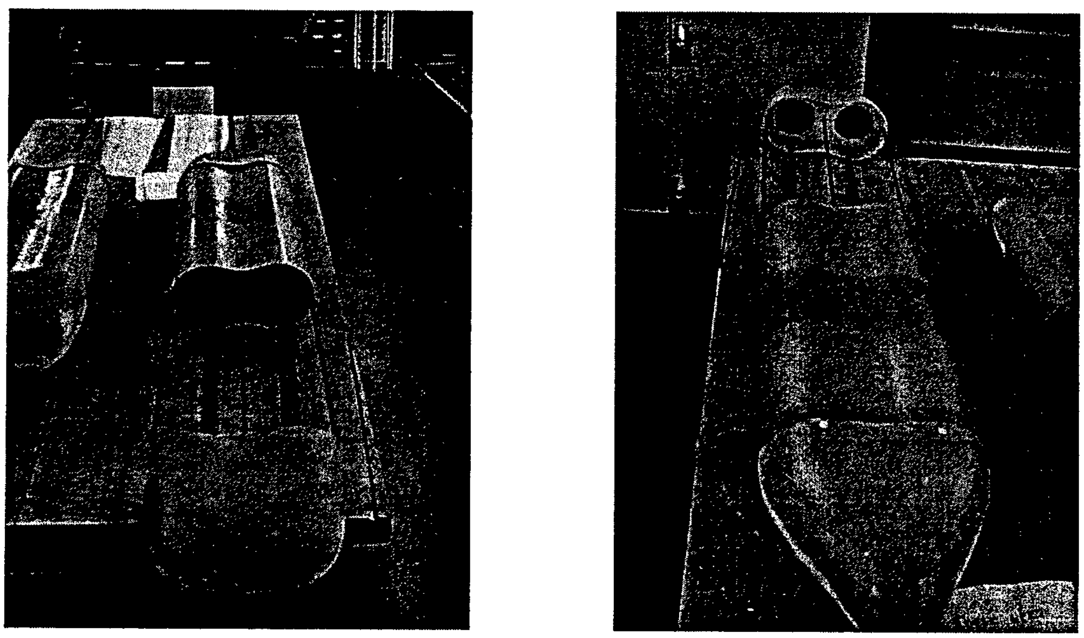


Figure 4.3 A rail loading system.

fast, but also it had to be loaded and unloaded fast as well. It's a well-known fact that engineers are not the most coordinate people and MIT engineers probably the least of all. The team needed a design that was strong, light and easy to load. The fuselages surrounding the balls were designed for torsional loads and could not be cut. The balls had to be accessible all at once so that team members could grab a lot

of them. The design chosen was a front-loading fuselage with the payload on rails. By putting the payload on rails, it could slide in and out of the casing all at once without affecting the structure of the airframe. The rails align the balls as they are loaded and allow for easy, quick removal.

4.2.7 Modularity, Simplicity, Durability, and Ease of Manufacture

Modularity is the key to simplicity, durability and ease of manufacture. By dividing the structure into pieces, the team was able to simplify design, replace broken sections quickly, and build multiple parts of the aircraft at once. Modularity allowed the team to break apart the structure and concentrate on a single part. This type of intense analysis has allowed rapid building and has enlarged the margin of error. Because the structure is connected with adjustable mounts, tolerances increase. The structure is more durable to impact. Max build time

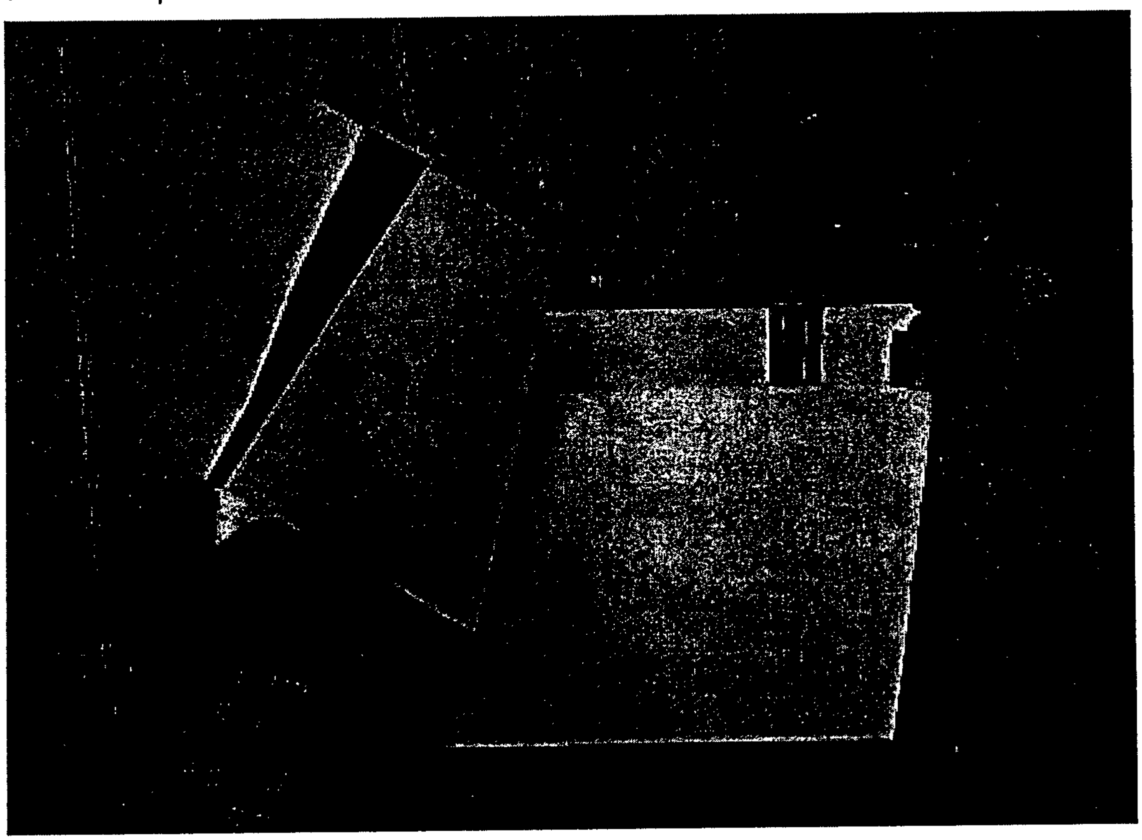


Figure 4.4 A delaminated wing tip showing blue foam, black carbon fiber, and white fiberglass. This piece can be produced in less than two man hours (plus overnight dry time). Modularity such as this made the plane easy to produce and easy to repair.

on any module is less than three days. That includes lay-up, drying time and fitting. Small parts that are known to suffer damage, such as the tail feathers or the wing tips, can be replaced in less than two man-hours (plus overnight drying time).

The biggest benefit of modularity is reduced building problems. Parts that do not meet specifications can still be used or, more importantly, easily replaced. For example, the first set of wing tips delaminated shortly after lay-up. This took only four man-hours to fix and the team reduced building time for the next set. These tips were reproduced without modification to any other part of the airplane. Modularity has been the approach to this complex problem because it produces the maximum benefit for dispersed work.

4.2.8 Originality and Aesthetics

Beaver Pride! As MIT students the team wanted to uphold the tradition of excellence at MIT as "Better Than Jack" displayed the airplane before other college students and engineers. The plane of 2000 was called the "Air Camel" and was about that ugly, too. This year the team has added some aesthetic touches and developed cleaner lines. The team has attempted to match the elegance of the solutions to the presented problems with the elegance of the aircraft in flight.

4.3 Summary

The key features of this airplane include pusher/puller engine configuration, multiple fuselages, separated V-tail, and low-mounted wing. Each of these design decisions can be clearly traced back to requirements, and produce a final configuration optimized around competition parameters. These decisions were reached after brainstorming and testing using skills typical to the field. Standard assumptions to simplify calculations were used when necessary. Additional assumptions about competition conditions influenced configurations. The team took a large amount of data and presented solutions that were appropriate and flexible. Testing supported and clarified these ideas. As major design decisions were reached, minor design parameters became clearer, greatly simplifying difficulties.

4.3.1 Wing and Power Loading

Computational Analysis has been instrumental in determining wing and power loading. Engine parameters input into MotoCalc gave the team an estimate power output. A simulated race run on Matlab scripts gave the team an accurate estimate of battery drain. The chosen airfoil was evaluated under Xfoil to give a clear idea of lift and drag.

4.3.2 Assumptions

The team drew on experience to form assumptions about race day conditions. The plains of Kansas are known to be windy and constants for formulations were compiled for that region. Experience also gave the team general assumptions about the configuration of the aircraft that are standard to the field, such as operation of flaps and control surfaces. These assumptions have been extremely helpful as a baseline. Computer simulations as described were assumed to be within a few percent.

4.3.4 Preliminary Phase Confirmation of Conceptual Phase

The ease of the preliminary phase is a tribute to the completeness of the conceptual phase. In the conceptual phase, testing and logistics narrowed down multiple possibilities to a single strong solution. By assuming linear solutions, the team was able to move towards better solutions, while not forgoing a stable design. Using medium numbers as a baseline and moving out from there is a conservative technique guaranteed to produce a feasible solution at a given time. The team used this model to insure that at the end of the design phase the team had a working model ready for construction.

5 Detail Design

The detail design section entails the creation of a full aircraft design from the high level concept proposed by the team during the conceptual design phase. This task is further subdivided into the sizing of the aircraft, propulsion design, internal layout, and aerodynamic design. The end result of these steps is the "Blue Light Special." This design is the evaluated using several metrics including weight, stability, and projected contest score.

5.1 Aircraft Sizing

The aircraft sizing decision was simplified by several constraints on the team. These included limitations put on the design by the contest rules and materials and parts available to the team.

The key rules constraining aircraft design were the motor specification of brushed Astro or Graupner motors, the 40 amp current limitation, and the 5lb.battery weight limitation. The rules concerning the mission profile were of course the driving factor in the aircraft design, but the above-mentioned limitations were the preliminary methods for constraining the size of the aircraft.

The availability of motors also turned out to be a significant factor in the aircraft design. The availability of Astro Cobalt and Graupner motors of sufficient size and power for this competition turned out to be very limited. The team found that two Astro Cobalt 25 motors was the maximum power with which the aircraft could be outfitted. This supply problem meant that the entire design was baselined around this motor configuration.

5.2 Propulsion Design

5.2.1 Battery Choice

The 40 amp current limitation, the 10-minute and/or 6 lap constraint, and the 5 lb limit were the major factors used to determine battery pack cell count and size. The current limitation meant that the team had to raise the cell number in order to achieve a total power output similar to previous years. The two motor configuration meant that each of two battery packs would independently drive one motor. The cell count of each pack was limited by both the motor's max cell limit and by cell capacity. For the given weight of 2.5 lb/pack, as the cell count rises, the capacity drops. A baseline cell count of 18/pack with capacity of 2400 mAh/cell was set to meet these demands.

5.2.2 Drivetrain Selection

Once the motor selection was fixed by the default supply situation and the battery pack configuration is set, the propulsion design became a mere selection of prop size/pitch and gear ratio selection. In order to make a selection that offered a good mix of efficiency, top speed, and static thrust, the team used Motocalc™ to model the propulsion system. Motocalc takes inputs that include the

motor(s) used, batteries, gear ratio, aircraft dimensions, and speed controller characteristics. It uses this data to predict the performance of the propulsion system at various throttle settings and flight speeds.

Several design points were important to the selection of a prop/gearbox combination. The selected system could not have a stalled prop condition at velocities above a few feet/sec. This would result in low efficiency and low static thrust. This metric is critical to takeoff, which is a limiting design parameter.

The selected system also could not exceed 300°F motor temperature for the uncooled case. The motors and battery packs in the selected conceptual design have internal flow cooling, which will bring the actual operating temperature down to a more reasonable level. Some drive combinations with high pitch props were ruled out on this condition.

Top speed of the aircraft is crucial for a high scoring sortie, since the sortie time appears in the denominator of the single flight score. Therefore, propulsion systems that could not achieve an estimated top speed of at least 65 mph were ruled out. The optimal configuration for these criteria was therefore a 13"x10" prop on a 3.1:1 gearbox.

5.2.3 Propulsion System Analysis

Using a Motocalc output of the propulsion characteristics as a function of velocity, the team used several Matlab scripts to predict aircraft performance. These scripts, which were used in previous years for the same purpose, were updated to reflect the new contest parameters. They perform a time-step through the entire sortie; calculating cumulative battery drain as the aircraft performs the takeoff, climb out, level flight, turns, etc. The scripts model all of these situations and the physics that they represent. The takeoff script is appended as Appendix A. The full mission script is appended as Appendix B.

Based upon an estimated payload mass fraction of 30%, the scripts helped to establish a limit on the number of softballs that the aircraft could carry, and a corresponding score. This data was used to further refine the aircraft sizing. This parameter sets the length of the two outer pods, since their length is set by cargo capacity. It also establishes an aircraft gross weight limit, and a battery life estimate. This battery life estimate was used to check the previous configuration the team had set. The baseline battery pack met this design constraint as well as the others previously applied.

The gross weight established here is applied to the aerodynamic design later in this detail design section. Based upon the estimate power consumption during the flight profile and a 20% safety factor, the matlab script prediction was a gross weight of approximately 23 pounds, for which the team estimated that the softball capacity would be 16 balls. These values are carried forward to the internal layout and aerodynamic design below.

5.3 Internal Layout

The conceptual design phase specified merely the general shape of the aircraft and locations for some of the critical components. Those specified included the push-pull configuration of the motors on

the central fuselage, the positioning of the softballs in the two outer pods, and the placement of nearly all of the electronics in the center pod. They also included a standard wing placement with twin tail booms with one tail surface each.

The push-pull configuration of the motors was chosen to simplify the outer pods. By mounting both motors on the same fuselage, the associated batteries and speed controllers could also be centralized in this fuselage. The center fuselage also contains the radio receiver, receiver battery, and the nose gear mount. The motor mounting plates are perforated with cooling channels to allow airflow into the interior of the center fuselage. The internal layout is set up so that flow obstructions are minimized. The motors, batteries, and speed controllers are all cooled by the internal airflow.

All of these electronic components are bulkhead mounted. The motor mounting bulkheads are carbon-balsa sandwiches. Each motor casing is further supported by another bulkhead. The battery packs are cradled by three separate bulkheads. Axial motion of the packs in the case of a hard landing is prevented by another bulkhead in front of the packs. The battery packs are spaced one inch laterally from each other to allow for wiring space, receiver and receiver battery mounting, and speed controller space.

The outer pods have almost no electronics in them. They are designed expressly for the purpose of transmitting loads and carrying softballs. The softballs are mounted on a sliding cage built of graphite spars. These spars are joined by bulkheads fore and aft. The forward bulkhead is mounted directly into the nosecone. The nosecone slides off of the outer pod, sliding the cage with it. This permits access to the softballs for loading and unloading. The cages are held into the fuselage by guide tubes anchored to the inner leading edge of the pod. When the nosecone is fully seated on the outer pod, it is locked into place, thereby holding the cage and the softballs in place.

5.4 Aerodynamic Design

5.4.1 Airfoil Design

The airfoil that the team selected would have to achieve a combination of low drag for the high speed portions of the sortie and high lift for the loaded laps. The team decided that, as in the past, the design would benefit from a wing designed to be "reflexed" in flight. That is, during the high speed cruise portion, the trailing edge that acts as the ailerons and the flap will be deflected upward. This reduces the lift generated, but also reduces drag. The reduction in lift is due to a reduction in the effective camber of the airfoil. The reduction in drag is due to the lessening of the adverse pressure gradient along the upper surface and the reduced vorticity generated by the wing as it generates less lift. The actual airfoil shape was based closely upon designs from previous years. These designs were also optimized for an aircraft that would need both lifting and sprinting capability.

The result, the DA11 airfoil, has a L/D ratio of 38 at a cruise C_l of 0.27 with the trailing edge reflexed. At the other extreme, it achieves a C_{lmax} of 1.32 with the trailing edge deflected. The airfoil

design was done using Xfoil, a 2-D viscous flow solver that iteratively couples a potential flow solver and a boundary layer solution. Xfoil allows airfoil design by geometry, surface pressure, or surface velocity. It also permits generation of basic aerodynamic forces such as lift, drag, and moment, in addition to polar plotting and boundary layer characterization.

5.4.2 Planform Design

Based upon the gross weight of 23 lbs set by the team's Matlab calculations and a desired wing loading of approximately 3 lb/ft^2 or .35 oz/in^2, the wing area was set at 1100 in^2. The desired wing loading is an empirical guideline that the team generated in conjunction with one of the team's advising professors, Prof. Drela. This wing loading level is consistent with model aircraft that have high power loading and moderate speed requirements.

The planform optimization was done using a vortex lattice method. The software selected for this was the Athena Vortex Lattice(AVL) program. AVL takes as input several measurements from the wing. The dimensions and orientation of each panel are entered, as is the airfoil used at the break between each panel. In the case of the Blue Light Special, all of the wing sections use the same DA11 airfoil. AVL allows induced drag calculation and Trefftz Plane analysis. The number of wing panels and the taper ratios between these panels were all refined in AVL.

5.4.3 Fuselage and Tail Design

The aerodynamic design of the fuselage and outer pods was limited to size and shape of the nose cones of the pods, since every other major measurement was defined by the contents of these components. The outer pod nose cones and tail boom adaptors were shaped to create lift over the surface of the pods. This decision was made to offset the large loss of wing area that results from the attachment of the three fuselage sections.

The tail design was done using stability data derived from Envision Design's "Plane Geometry MeasureV." This is an excel macro used for generation of stability derivatives and downwash angles. The aircraft geometry, mass, and center of gravity are used for these calculations. Based upon the derivatives generated for a given geometry, the aircraft can be modified, and the derivatives regenerated in real time. For the Blue Light Special, the twin boom configuration lent itself well to a split v-tail design, where one tail plane would be attached to each boom. In this v-tail configuration, based upon the data from the Plane Geometry macro, the tail surfaces were each set at 16 inches in length. They taper from a root chord of 6 inches to a tip chord of 4 inches. The tail planes use a nearly symmetrical HD801 airfoil.

5.5 Graphics

A set of three view drawings of Blue Light Special is attached. See Appendix C.

5.6 Design Evaluations

The performance data for Blue Light Special was calculated using Matlab, AVL, and the contest Specifications. These metrics include takeoff, handling qualities, g-loading, mission performance(score), weight and balance, and the Rated Aircraft Cost for the aircraft.

Rated Aircraft Cost:

(A*MEW + B*REP + C*MFHR)/1000 = **17.13**

Coef.	Description	Aircraft Value
A	Weight Multiplier	$100
B	Rated Engine Power Multiplier	$1500
C	Manufacturing Cost Multiplier	$20/hr
MEW	Manufacturers Empty Weight	17.5 lb
REP	Rated Engine Power	6.25
MFHR	Manuf. Man Hours	212.7 (see below)

Item	Cost($)	Units	#	Subtotal
Wing				
Wingspan	8	per foot	8	64
Max chord	8	per foot	1.21	9.67
control surface	3	per surface	3	9
Fuselage				
max length	10	per foot	5.5	55
Emp				
vertical suface	5	per surface	1	5
vertical w/cont	10	per surface	0	0
horiz. w/cont	10	per surface	1	10 (V-tail)
Controls				
servo/cont	5	per component	8	40
Prop				
Motor	5	per motor	2	10
Propeller	5	per prop	2	10

Total MFHR: 212.7

Figure 5.1 shows the output of the Matlab script simulating the mission. The energy graph shows the progression of battery usage. This is calculated using the Motocalc provided profile of current usage vs. throttle setting and the Matlab flight profile. The nearly even slope of the energy usage reflects the fact that current level is reletively flat during the flight. The only portions that are exceptions are the approach and pit stop sections. The flat portions are these portions of the flight. During these sections it is assumed for simplicity that the throttle is completely closed.

The velocity graph shows the quick accelleration of takeoff, climbout, and cruise. The perturbations in the cruise portion of the velocity profile are caused by speed loss in the turns. The time periods where velocity equals zero are the pit stops.

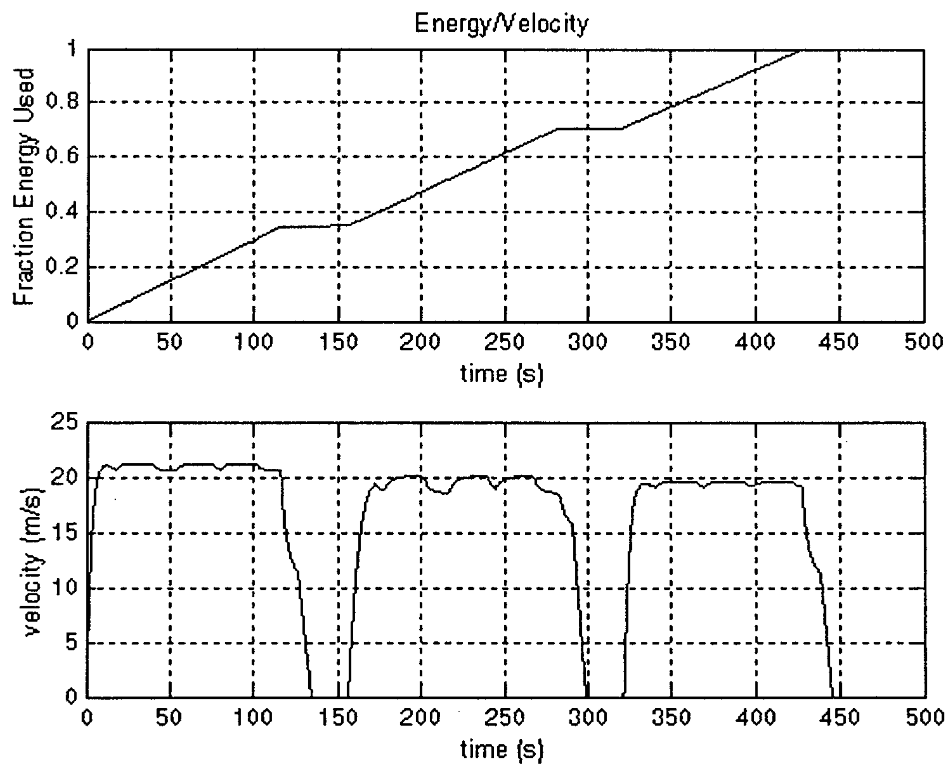


Figure 5.1 Mission performance.

Figure Figure 5.2 shows the takeoff performance of the aircraft for the first portion of the flight. For this flight, the aircraft is unloaded and fully charged. This is therefore the optimal takeoff condition. For the second takeoff, the aircraft will have somewhat drained batteries and a full payload. For the final takeoff, the payload will once again be removed, but the batteries will have been heavily taxed by the two loaded laps.

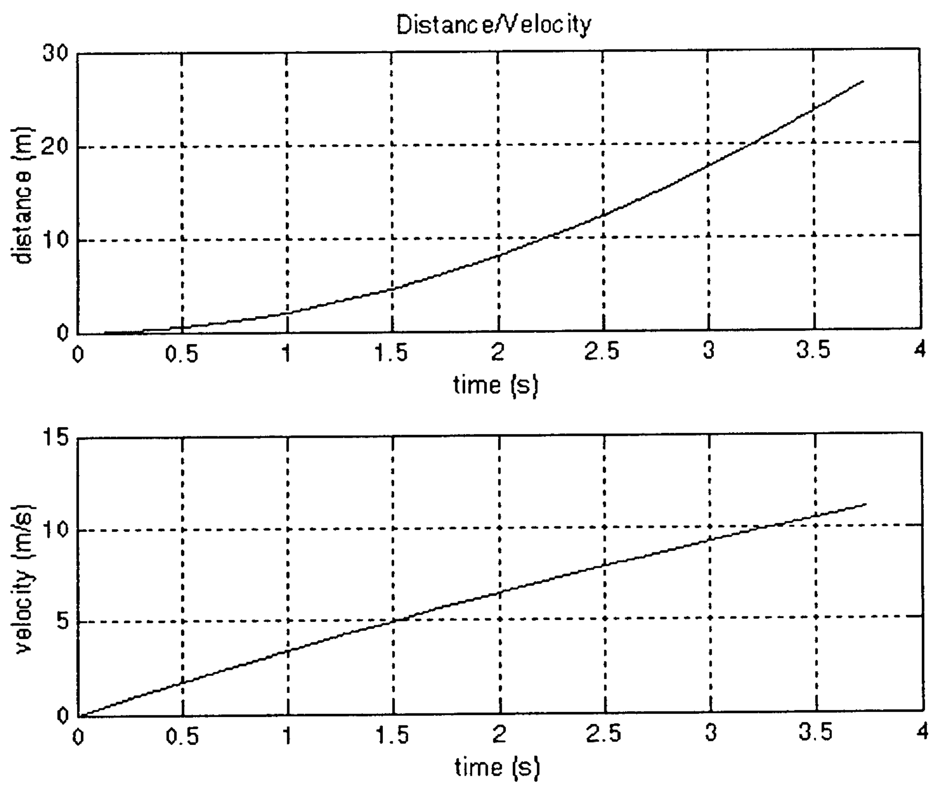


Figure 5.2 Takeoff Performance at full charge with no payload.

Table 5.3 Aircraft stability derivatives and downwash angles.

(7) STABILITY DATA	
K @ H=0	2.18
K @ H=0.2	1.69
Hh/b	0.06
K @ Hh	2.03
Effective H Stab area (sq in)	79.50
Sh eff / Sh	0.38
Y neutral point, wing-tail (in)	6.55
N.P. on Cmac, wing-tail (Cmac)	0.52
N.P. change for fuse + sweep (Cmac)	-0.0713
Y neutral point, aircraft (in)	5.74
N.P. on Cmac, aircraft (Cmac)	0.45
Measured Y of CG (in)	3.50
Measured CG on Cmac	0.25
Static Margin (Cmac)	0.20
Cl alpha wing (Cl/°)	0.0909
Cl alpha horizontal (Cl/°)	0.0659
Cl alpha vertical (Cl/°)	0.0307
d epsilon / d alpha	0.3469
Cm alpha wing (1/radian)	-0.0094
Cm alpha fuselage (1/radian)	0.3717
Cm alpha tail (1/radian)	-1.3882
Cm alpha total (1/radian)	-1.0259
Neutral Point, Cmac	0.4451
Cm alpha dot (1/°/sec)	-0.1035
Cm q horizontal (1/°/sec)	-0.2983
Cm alpha dot + Cm q horizontal (1/°/sec)	-0.4018
Cm delta elevator (1/°)	-0.0248
CL delta elevator (1/°)	0.0062
Elevator deflection per Cl (°)	8.3490
Cn r vertical (1/°)	-0.0012
Cn delta rudder (1/°)	-0.0012

(8) DOWNWASH ANGLE TABLE

relative to downwash at C/4 as a function of distance aft of C/4 in spans

Lh/b	K @H/b=0	Slope	K@ H/b =0.2	Slope
0.20	2.84	-4.20	1.97	-1.70
0.30	2.42	-2.10	1.80	-1.00
0.40	2.21	-1.20	1.70	-0.50
0.50	2.09	-0.60	1.65	-0.40
0.60	2.03	-0.50	1.61	-0.20
0.70	1.98	-0.50	1.59	-0.20
0.80	1.93	-0.50	1.57	-0.20
0.90	1.88	-0.30	1.55	-0.10
1.00	1.85	-0.30	1.54	-0.10
1.10	1.82	-0.20	1.53	-0.10
1.20	1.80	0.00	1.52	0.00

The weight and balance sheet, Table 5.4, shows the layout of the components of the aircraft. All axial locations are relative to the quarter chord of the wing. Positive axial locations are behind the wing, negative in front. The cg is therefore almost exactly at the quarter chord. Components such as the batteries can be moved over a limited range to adjust the Cg in the event that flight testing indicates it should be moved.

Table 5.4 Weight and Balance for Cg calculation.

Weight And Balance Sheet

Part	Number	Mass (g)	(from c/4) Axial Location	Moment
outer pod	2	266	0	0
center wing	1	888	0.025	217.56
outer wing	2	249	0.025	61.005
tail surface	2	40	1.25	490
tail boom/fairing	2	192	0.46	865.536
motor/spinner	2	435	-0.1	-426.3
center fuselage	1	230	-0.1	-225.4
Batteries	2	1100	-0.1	-1078
servos/controller	8	20	0.2	39.2
main gear	2	80	0.05	39.2
nose gear	1	75	-0.18	-132.3
wing tips	2	51	0.025	12.495
Softballs	16	188	0	0
misc fasteners	1	100	0	0

Mass:	9287	g
	20.46	lb
CG:	-0.0015	m
rel to c/4 pt	-0.059	inches

The aircraft scoring metric is dependent upon three separate numbers. The report score (RS) is multiplied by the total flight score (TFS). This product is then divided by the rated aircraft cost (RAC):

$$SCORE = \frac{RS \times TFS}{RAC}$$

Total Flight score is the sum of three separate single flight scores (SFS), where:

$$SFS = \frac{\#Laps + \#BallsCarried}{TotalTime}$$

Based upon Matlab simulation, 6 laps will be flown, with 16 balls on the loaded laps, with an estimated finishing time of 7.5 minutes. This yields an SFS of 2.93. The corresponding TFS is 8.8, assuming all single flight scores are equal. Finally, the total score, dividing out the report score, is 0.514.

Spar stress analysis/G load Capability

The maximum structural G-load of the vehicle is limited by the bending characteristics of the spar. This maximum structural load for the wing spar is set by a few criterion. The design maximum cap stress is 140ksi, which through previous spar failure testing is a known max stress for our pre-cured CF strips. The cap modulus is 20 msi, which is typical for the carbon used in the pre-cured pre-preg composite strips. While the spar could simply be designed not to break in strength, this would not preclude a wing that would bend excessively. Hence the wing spar was analyzed using a spreadsheet formula which outputted both cap stress load and estimated tip deflection.

Also helping out is the relatively span loaded weight distribution. As the outer fuselages hold a substantial portion of the aircraft's weight they help counteract the lifting force of the wing and hence help lower the bending moment. Hence a relatively high maximum g-load can be attained. For instance, a case run at 20 g's results in a tip deflection of 1.38". Deflection beyond this point could begin to have serious effects upon the controllability of the aircraft, so the team will consider this to be the design max G-load. This assumed the above CF properties and a proper load distribution from the center and outer fuselages as well as a spar height of 1.3".

While this spar strength may seem excessive, it is comforting to note that the spar is all but indestructible in the air. Also it will likely hold up to the rigors of initial testflights much better.

6 Manufacturing Plan

6.1 Relevent Figures of Merit

The figures of merit relevant to the manufacturing of the contest are listed below with their respective weightings:

Ease of Manufacture	9
Modular Construction	6
Minimize Time to Build	7
Minimize Weight	7
Durability	6
Aesthetics	3

The construction approaches detailed here were selected because they had worked for previous teams from the MIT Aero/Astro Department whose expertise could be consulted from time to time. The materials required for the chosen construction techniques (Kevlar, unidirectional carbon fiber, fiberglass) are expensive, but much of these were readily available as excess from previous projects, thus incurring no cost. Also, using these proven techniques greatly helped ease and time of manufacture for our novice team. As construction progressed, our skills in these techniques improved and expedited the process.

6.2 Wing construction:

Sub-assembly 1: Foam Cores

These were constructed using a computer numerically controlled hot wire foam cutter. The foam cutter is able to make sweep and taper. The foam cutter was also used to cut out gaps in the foam cores for the insertion of the wing spar.

Sub-assembly 2: Spar

The spar is basically a sandwich beam with carbon fiber caps on a beam of vertical grain balsa. Then, a fiberglass-epoxy layer laid on the 45° was a wrapped around the beam to relieve the shear on the balsa. The construction proceeded in three phases: shaping of the balsa layer, application of spar caps, and fiberglass wrap.

The balsa layer is to be constructed by cutting 1" by 3" balsa board stock into appropriately thick chunks then glue them end to end with wood glue. This could be done one of two ways. First the balsa could be cut to the exact contour of the wing for all 33 pieces required to make the span and minimize sanding. The other option is to cut the balsa pieces close to the required size and then sand to shape. The second option was chosen by applying the figures of merit, evaluation of the team's construction skill, and available tools. The second option is easier to implement and requires less time than the first. Also, with only a jigsaw available for cutting the balsa and the slight taper near the root, the task would quickly grow out of scope. For these reasons, the balsa was rough-cut first and then sanded to shape and compared manually with the foam cores, allowing extra room for the carbon spar caps.

The spar caps were attached to the balsa spar core with a thin layer of 15-minute epoxy. Since a long cure is unnecessary, this adhesive choice allows for expediting the spar construction. The spar caps were commercially ordered since they would be made at higher quality and far quicker than if done in-house.

The fiberglass layer was applied using 24-hour epoxy cure. The fiberglass was cut in a section that could wrap all the way around the spar. It was then wetted with epoxy and all excess was rolled out. The wetted fiberglass was then wrapped about the spar, and a long strip of polypropylene was wound about it to hold the fiberglass in place. Once prepared, it was placed in a vacuum bag over night to cure.

The final pieces to the spar were three vertical-grain balsa extensions placed along the spar for attachment of the three fuselages. These were attached using 15-minute epoxy since not much time was needed for the lay-up and the need to ensure good alignment. Next holes were drilled through each extension and the spar and carbon fiber rods were inserted and affixed with 15-minute epoxy.

Sub-assembly integration:

First, the foam cores were attached to the spar using 15-minute epoxy. This allowed for swift construction and also helped ensure proper alignment. If they were allowed to cure overnight unsupervised, it is likely that they may have fallen out of alignment. Any gaps were filled using epoxy with sanding filler.

The final process needed to finish the wing was the application of a fiberglass skin. Fiberglass was laid on at the 0°-90° and the 45°. In order to do this, pieces of mylar shaped like the top and bottom of the wing were cut out. The mylar was lightly waxed and then a coat of spray paint was applied. The decision to paint was made because of the aesthetics figure of merit and the performance boost from having a smooth painted finish.

Once the paint had dried on the mylars, the wing fiberglass was laid out on them and wetted out with 24-hour epoxy with all excess removed. In order to adequately cover the leading edge, and additional fiberglass strip was applied manually to the foam core and wetted out by hand. Once all the glass was wetted, the mylars with the wetted glass were placed around the foam core and spar, and the whole package was placed in the vacuum bag for overnight cure. Once cured, the wing was trimmed and control surfaces were cut using a razorblade and necessary parts were sanded to the desired shape.

6.3 Tail Surface Construction:

These were constructed in a similar manner to the wing. Foam cores were cut using the computer numerically controlled foam cutter. For a spar, a strip of unidirectional carbon fiber was applied on the top and bottom near the quarter chord. Mylars were cut to shape, lightly waxed and spray-painted. Two fiberglass sheets cut on the 0°-90° and the 45° were then placed on the mylars. The fiberglass was wetted with 24-hour epoxy and all excess was removed. A leading edge strip of fiberglass was also wetted directly onto the foam core. The mylars and fiberglass were then wrapped around the foam cores and the whole assembly was placed in a vacuum bag for overnight cure. Once cured, the excess was trimmed and control surfaces were cut into the fin with a razorblade.

6.4 Propulsion Fuselage Construction:

Mold Preparation:

In order to prepare the mold, a form was made out of foam based on the engineering drawings. It was rough cut using a hot wire and then sanded to the desired shape. In order to prevent the form from sticking to the plaster for the mold, it was wrapped in packing tape and lightly waxed. A cardboard shield was placed about the form so that the two halves of the mold could be made separately. Once the shield was affixed, a layer of Plaster of Paris was laid on it, followed by a layer of reinforcing drywall tape and another layer of plaster. Once this had dried, the other half of the mold was made using the same method.

Composite Lay-up:

The two halves were made of two layers of Kevlar laid on the 0°-90° and the 45°. Each piece of Kevlar was wetted with 24-hour epoxy and all excess was removed. The Kevlar was laid evenly in the mold with pieces of unidirectional carbon fiber where reinforcement was called for. The fuselage was allowed to cure overnight.

Internal Layout:

The internal structure consists of a series of bulkheads for the mounting of the motors and batteries. The motor face bulkheads were cut out of basswood faced with carbon fiber weave on the 45° and backed with fiberglass on the 0°-90° and the 45°. The backing and facing were affixed with 24-hour epoxy. The motor mounts were aligned by first aligning the entire fuselage to an external reference and then aligning the motors to the external reference. The mounts were attached using 24-hour epoxy with adhesive filler to increase the thickness of the bond. The boundary mounts were made of balsa faced and backed with fiberglass on the 0°-90° and the 45° laid up with 24-hour epoxy. The battery mount bulkheads were attached similarly using 24-hour epoxy and adhesive filler.

6.5 Payload Fuselage Construction:

Sub-assembly 1: Main Fuselage Body

The first task was to construct a form to lay the composite around. The form was made from two 4-inch diameter copper tubes that were attached together and wrapped with a foam layer to cover the gap between the tubes. To allow the finished composite piece to come off the form, it was wrapped in plastic and waxed. The lay-up consisted of two pieces of Kevlar, one on the 0°-90° and the 45° and two unidirectional carbon fiber strips cut to wrap once around the form. Each piece of fabric was wetted with 24-hour epoxy and with all excess thoroughly removed. Each piece was wrapped about the form, and then the whole lay-up was placed in a vacuum bag for overnight cure.

Sub-assembly 2: Nose Cone

A form for the nose cone was made out of foam to match the engineering drawings. It was rough cut with a hot wire and then sanded to the required shape. The foam was then wrapped in clear tape and waxed to prevent it from sticking to the plaster. A cardboard ring was put around the midsection of the piece to separate the top and bottom half of the mold. The mold was created by layering Plaster of Paris with reinforcing drywall tape.

The lay-up consisted of a single ply of Kevlar on the 45° wetted with 24-hour epoxy. One ply was put in each half of the mold and their ends were overlapped. The two halves of the mold were taped together and allowed to cure overnight.

Sub-assembly 3: Tail Cone

The tail cone was made in the same manner as the nose cone. A foam form was fabricated to make a plaster mold around. The tail cone was made three-ply with two layers of Kevlar on the 45° and one 0°-90°. Reinforcing carbon strips were included where the tail boom would mount.

Sub-assembly integration and Ball loading Mechanism:

The ball loading mechanism consists of carbon tubes attached to the nose cone and a back plate that form two ball-holding channels. The back plate is made of fiberglass reinforced balsa. The carbon tubes attach to the main fuselage by tubular sheaths glued to the main fuselage allowing the ball cradle to slide freely.

The composite tail booms are attached to the tail cones by a pair of bulkheads that the boom is inserted through. The two bulkheads are made of fiberglass-reinforced basswood. The bulkheads are attached to the tail cone by 24-hour epoxy with adhesive filler. The tail cones fit over the ends of main fuselage and are held in place by four nylon bolts.

6.6 Final Integration:

This is the plan for the final integration of the contest aircraft. This has yet to be done.

The fuselages are all carbon reinforced at their point of attachment. They will be bolted onto the wings at the points where the spar was extended. The center fuselage will be used to adjust the center of gravity. There is enough leeway for and aft that it can be used to adjust the center of gravity without altering the competition aircraft design.

The tail servos will be attached to the tail mounting bulkheads with the pushrods going through the composite tailbooms. The wing control surface servos will be sunk into the wing foam. The receiver will be located in the center fuselage attached to the top part to keep it separate from the batteries.

Table 6.1 Project Milestone Chart

Milestone	Date Scheduled	Date Achieved
Wing Complete	12/20/2001	2/25/2002
Complete Center Fuselage Shell	1/12/2002	1/24/2002
Center Fuselage Internal Layout	1/19/2002	2/12/2002
Payload fuselage shells	12/3/2001	1/12/2002
Nose Cones	12/6/2001	1/15/2002
Tail Cones	12/9/2001	1/16/2002
Payload Fuselage Assembly	12/13/2001	2/5/2002
Tail surfaces	1/10/2002	2/20/2002
Final Integration	1/24/2002	TBD

Appendix A

Matlab Integrated Vehicle Simulation

Takeoff Module

The takeoff module of the simulation consists of 3 Matlab scripts: takeoff.m, which is the main script; TO_deriv.m, which computes the net acceleration on the vehicle; and inputs.m, which contains all the input parameters for the simulation. All 3 files are presented below.

Takeoff.m

```
inputs;

[ t, s ] = ode23( 'TO_deriv', [ t0:0.05:tf ], s0 );

TO_index = find( s(:,1) > TO_vel );
TO_distance = s( TO_index(1), 2 )          % in m
disp( 'meters' );
TO_mah = s(TO_index(1), 4)/3600*1000       % in mA-hrs
TO_time = t( TO_index(1) )                 % in s
disp( 'seconds' );


% PLOTS
% -----

subplot(2,1,1)
plot( t(1:TO_index(1)), s(1:TO_index(1), 1) )
hold on;
plot( [t0, tf], [TO_vel TO_vel], '--' );
title('Takeoff Run');
xlabel('time (s)');
ylabel('velocity (m/s)');
axis( [ 0, t( TO_index(1) ) + 5, 0, 1.3*TO_vel ] );
grid;

subplot(2,1,2)
plot( t(1:TO_index(1)), s(1:TO_index(1), 2) )
hold on;
plot( [t0, tf], [TO_distance TO_distance],  '--' );
xlabel( 'time (s)');
ylabel('distance (m)' );
axis( [ 0, t( TO_index(1) ) + 5, 0, 2*TO_distance ] );
grid;
```

TO_deriv.m

```
function sdot = TO_deriv( t, s )

% define input variables/parameters
inputs;

% define expression for thrust:
index = round( s(1)+1 );
if (index > length(thrust))
   index = length(thrust);
end

T = thrust( index );

I = current( index );

% define expression for drag:
D = rho/2 * (s(1))^2 * ( S*Cd_o(1) + phi * S * (Cl_TO)^2 / ( pi * AR *
e ) + S_f*Cd_Af );

%define expression for ground resistance:
R = mu_r * ( m*g - rho/2 * (s(1))^2 * S * Cl_TO );

% define expression for sdot, the derivative of the state vector [s1 s2
s3 s4]
sdot = [ 1/m * ( T - D - R ); s(1); 0; I ];
```

Inputs.m

```
% inputs.m
% ========

% Environmental Parameters
% ------------------------

% freestream density:
% (kg/m^3)
rho = 1.225;

% coefficient of ground friction:
mu_r = 0.02;

% gravity:
g = 9.81;


% POWERPLANT Characteristics
% --------------------------

% NO LONGER USED:
% available power:
% (W)
%P = 300;

% propeller efficiency:
```

```
eta = 0.85;


% STRUCTURAL/PHYSICAL Characteristics
% -------------------------

% total mass = a/c mass + cargo mass:
% (kg)
m = 10.16;

% ground clearance of wing:
% (m)
h = 0.2032;


% AERODYNAMIC Characteristics
% --------------------------
% fuselage drag:
Cd_Af = 0.02;

% fuselage cross-sectional area
% (m^2)
S_f = 0.02;
% wing span:
% (m)
b = 2.134;

% wing area:
% (m^2)
S = 0.7097;

% Oswald's efficiency factor:
e = 0.9;

% coefficient of parasitic drag:
% (values at takeoff, climb, cruise, and landing configurations)
Cd_o = [ 0.012 0.012 0.012 0.2 ];

% coefficient of lift during takeoff roll:
% (based on incidence angle):
Cl_TO = 0.51;

% maximum coefficient of lift:
% (used for takeoff point--rotation)
Cl_max = 1.15;

% derived parameters:
% aspect ratio:
AR = b^2/S;
% ground effect correction:
phi = ((16*h/b)^2)/(1+(16*h/b)^2);
% takeoff speed:  (1.2 * stall speed)
TO_vel = 1.2 * sqrt( ( 2*m*g ) / ( rho*S*Cl_max ) );


% TAKEOFF Flight Parameters
% --------------------------------
```

```
% initial state:  (velocity, x-position, z-position, battery amp-hours
consumed)
s0 = [0 0 0 0];

% initial time:
t0 = 0;

% final time:
tf = 30;


% CLIMB Flight Parameters
% -----------------------

% flight path angle gamma
% (rad)
gma_climb = 20*pi/180;

% desired cruising altitude
% (m)
h_cruise = 17;


% LEVEL Flight Parameters
% -----------------------

% turn radius of aircraft
% (m)
turn_radius = 50 * 0.3048;

% ground track distance for 1 lap
% (m)
lap_distance = 2000*0.3048 + 2*pi*turn_radius;

% SORTIE SWITCH:  1 = loaded sortie;  2 = empty sortie
h2o_sortie = 1;


% LANDING Flight Parameters
% -------------------------

% desired landing glideslopes (2 phases)
% (rad)
gma_land1 = 20*pi/180;
gma_land2 = 10*pi/180;

% altitude where to switch from phase 1 to phase 2
% (m)
h_crit = 3;

% expected descent distance
descent_dist = ( h_crit/tan(gma_land2) ) + ( (h_cruise -
h_crit)/tan(gma_land1) );
```

```
% -----------------------------------------
% Thrust-Velocity and Power-Velocity data
load vel_thrust.txt -ascii;              % load data from Motocalc
thrust-velocity analysis
vel_data = vel_thrust(:,1)*0.44704;      % converting from MPH to
METERS/SECOND
thrust_data = vel_thrust(:,15)*0.278;    % converting from OZ to NEWTONS
current_data = vel_thrust(:,6);          % current is in AMPS

thrust = interp1( vel_data, thrust_data, [0:vel_data(length(vel_data))]
);
current = interp1( vel_data, current_data,
[0:vel_data(length(vel_data))] );
```

Appendix B

Matlab Integrated Vehicle Simulation

Full Mission Module

The mission module of the simulation calls all of the other modules. It runs them in a sequence to simulate the entire flight profile. By adding or deleting segments of the flight, the mission module can be easily updated to predict performance of varying missions. By editing constants module, the script can be modified as the aircraft is modified.

Mission.m

```
% mission.m
% AIAA D/B/F Mission Sortie Simulation
% Larry Baskett (updated by Adam Diedrich)
% December 2000/October 2001/March 2002
% -----------------------------------
constants;

% State vector initialization
% cumulative:
i = 1;     % state vector index
time = 0;
dist = 0;
Eused = 0;
lap = 1;
% current:
vel = 0;
alt = 0;
state(i,:) = [time,dist,Eused,lap,vel,alt];

   m = m_empty;
   Fcurve = emptyFcurve;
   Icurve = emptyIcurve;
   takeoff;
   climb;
   turn;
   level;
   turn;
```

```
turn;
turn;
level;
turn;
level;
turn;
turn;
turn;
land;
lap = lap + 2;
pitstop;
            m = m_loaded;
                Fcurve = loadeddrainedFcurve;
                Icurve = loadeddrainedIcurve;
                takeoff;
                climb;
                turn;
                level;
                turn;
                turn;
                turn;
  level;
  turn;
  level;
  turn;
  turn;
  turn;
  land;
  lap = lap + 2;
  pitstop;
m = m_empty;
 Fcurve = emptydrainedFcurve;
 Icurve = emptydrainedIcurve;
 takeoff;
 climb;
 turn;
 level;
```

```
    turn;
    level;
    turn;
    level;
    turn;
    land;
    lap = lap + 2;
    pitstop;

subplot(2,1,1)
plot( 3*state(1:i,1), state(1:i,3)/8000 )

%plot( [t0, tf], [TO_vel TO_vel], '--' );
title('Energy/Velocity');
xlabel('time (s)');
ylabel('Fraction Energy Used');
%axis( [ 0, t( TO_index(1) ) + 5, 0, 1.3*TO_vel ] );
grid on;

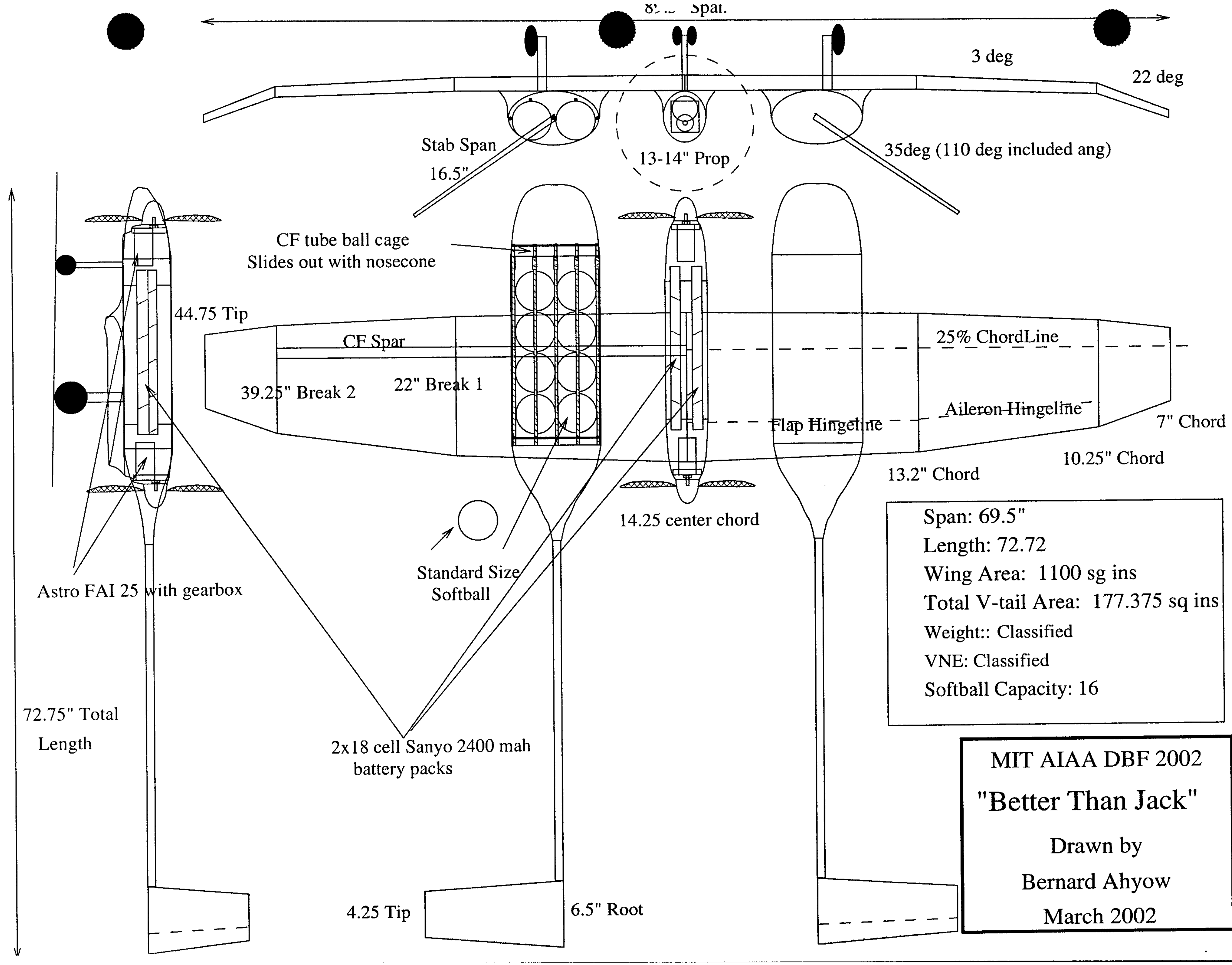
subplot(2,1,2)
plot( 3*state(1:i,1), state(1:i,5)/1.5 )

%plot( [t0, tf], [TO_distance TO_distance],  '--' );
xlabel( 'time (s)');
ylabel('velocity (m/s)' );
%axis( [ 0, t( TO_index(1) ) + 5, 0, 2*TO_distance ] );
grid on;

% O you who know what we suffer here, forget us not in your prayers.
```

Appendix C

Drawing Package


Spal.
3 deg
22 deg
Stab Span
16.5"
13-14" Prop
35deg (110 deg included ang)
CF tube ball cage
Slides out with nosecone
44.75 Tip
CF Spar
39.25" Break 2
22" Break 1
25% ChordLine
Aileron Hingeline
Flap Hingeline
7" Chord
10.25" Chord
13.2" Chord
14.25 center chord
Standard Size
Softball
Astro FAI 25 with gearbox
72.75" Total
Length
2x18 cell Sanyo 2400 mah
battery packs
4.25 Tip
6.5" Root
Span: 69.5"
Length: 72.72
Wing Area: 1100 sg ins
Total V-tail Area: 177.375 sq ins
Weight:: Classified
VNE: Classified
Softball Capacity: 16
MIT AIAA DBF 2002
"Better Than Jack"
Drawn by
Bernard Ahyow
March 2002

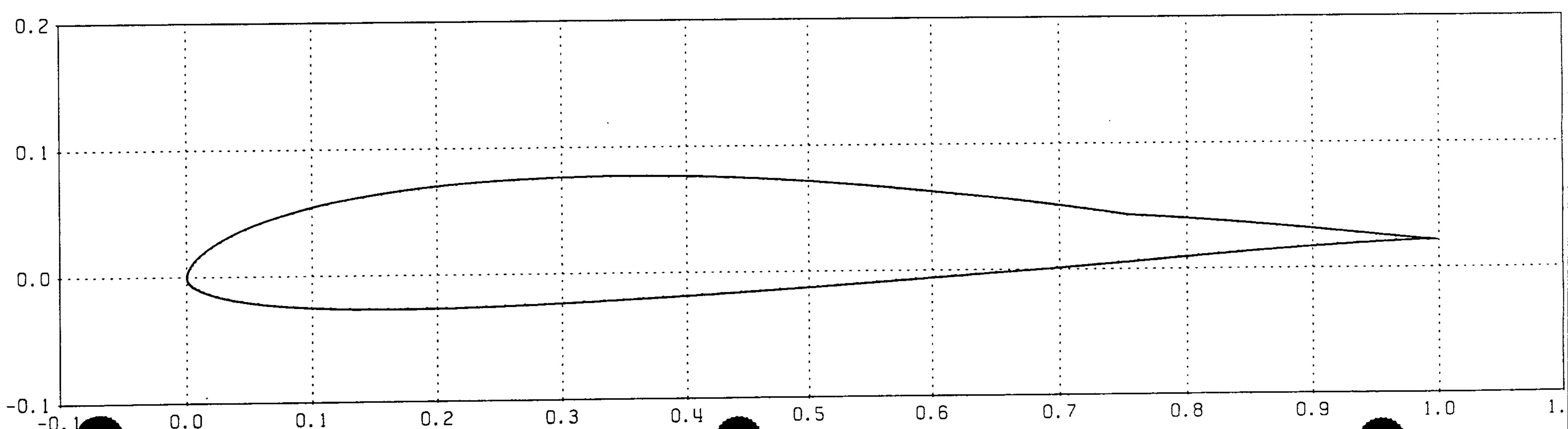
area = 0.06383

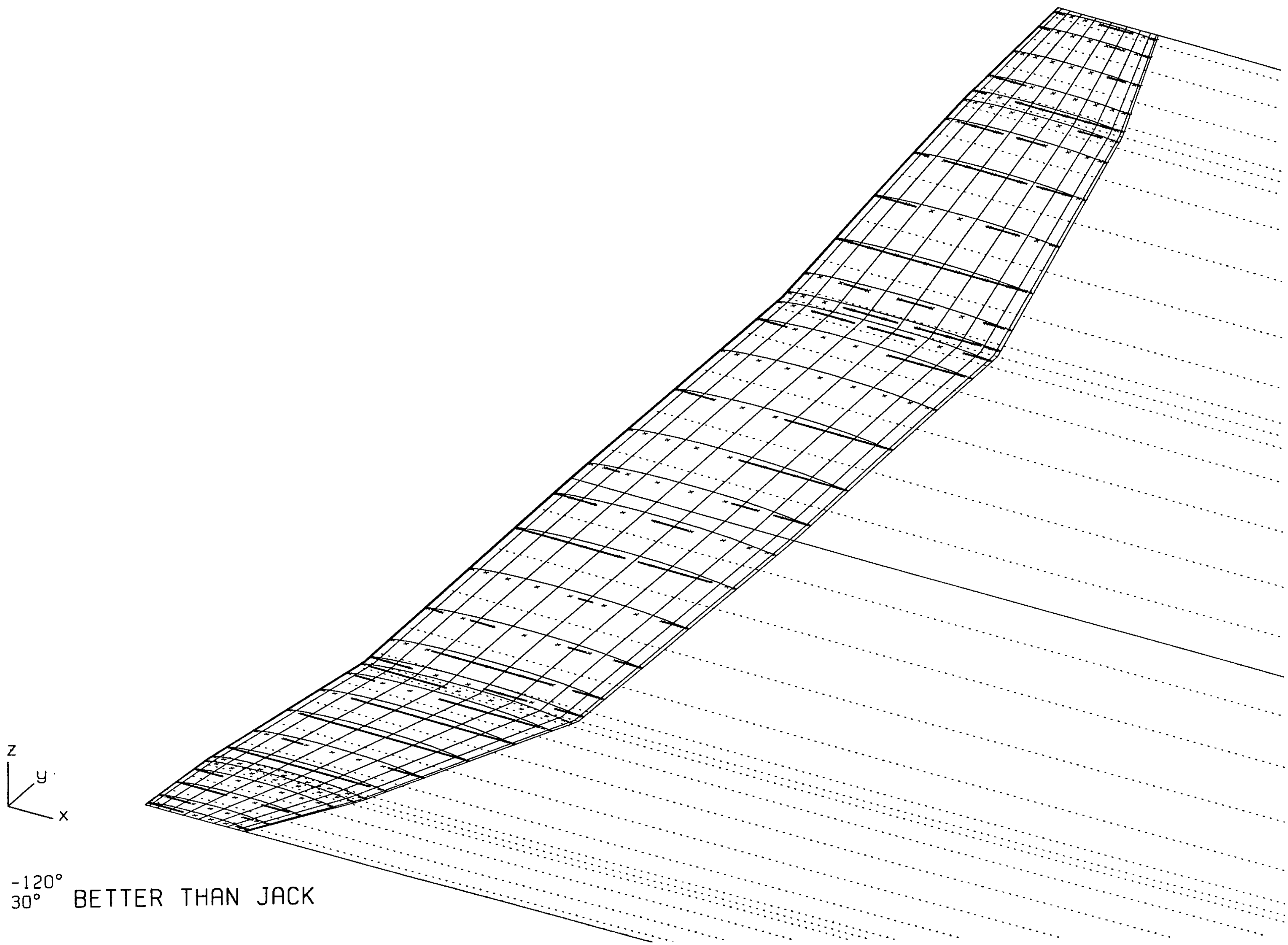
thick. = 0.10003

camber = 0.02068

r_{LE} = 0.00638

$\Delta\theta_{TE}$ = 2.31°



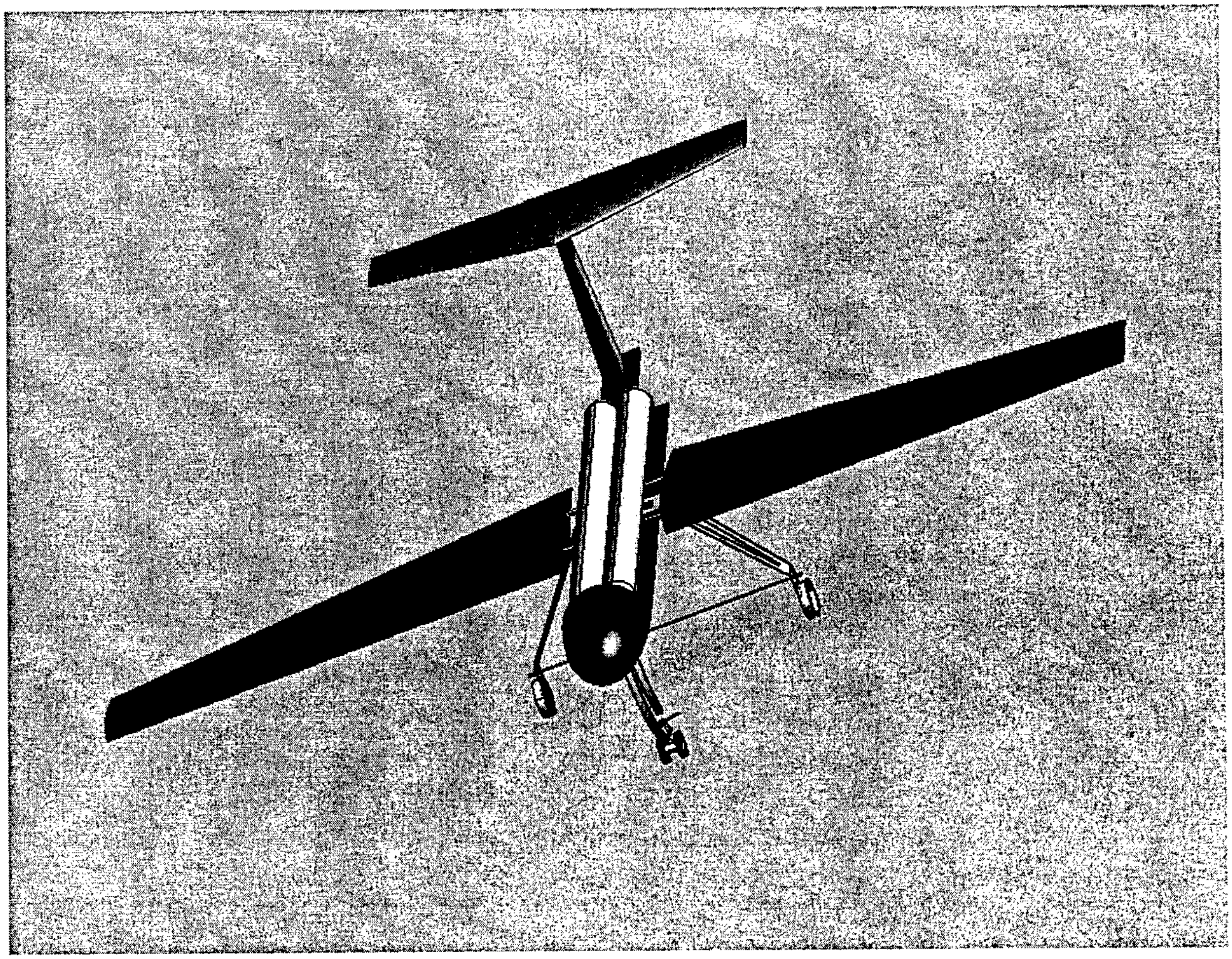


AZ = -120°
EL = 30° BETTER THAN JACK

AVL 2.71

2001/2002 AIAA FOUNDATION CESSNA/ONR STUDENT DESIGN BUILD FLY COMPETITION

Design Report
Proposal Phase



Team Darkhorse
United States Military Academy

March 2002

TABLE OF CONTENTS

1. Executive Summary

The United States Military Academy requires that all First Class Cadets, or seniors, be involved with a Capstone Project during the last one, or two, semesters at the Academy. As a result, eight members of the Class of 2002 chose to involve themselves with the Cessna ONR Student Design/Build/Fly Competition, a yearlong capstone that requires the design and fabrication of a remote-controlled, battery-powered aircraft. The aircraft must perform a set of missions, with or without a payload of softballs, within a ten-minute time limit. Seven of the seniors on the team are aerospace or mechanical engineering majors. The eighth senior is an electrical engineering major. In addition, there were four underclassmen, three juniors and a sophomore, who assisted on the project.

In order to successfully complete the project, the project was conducted in three phases: conceptual design, preliminary design, and detail design. For the greatest efficiency, the team was broken down into individual groups. Cadet Christopher Orlowski served as the Project Manager. Cadet Neal Erickson as the Executive Officer and was an assistant on the Fuselage. Cadet Kristine Conner served as the Budget Officer and an assistant on the Wing and Tail team. Cadet Jason Sienko was in charge of the Wing and Tail Team. Cadet Kevin Tobias was in charge of the Fuselage team. Cadet Bruce Skinner was in charge of the Landing Gear. Cadet Lucas Meyers was in charge of the Payload. Cadet Frank Hauben was in charge of all electrical connections: the servos, motor, control surface connections, etc. The responsibility for the design and the majority of the fabrication fell to each team. However, frequent meetings between individual teams and the project group, made the project run as smooth as possible. The key to the entire project was concurrent engineering. The underclassmen volunteers assisted in the fabrication process and were included as much as their schedules allowed in the design process.

1.1 Conceptual Design. As stated above, the design team was divided into five teams in order to facilitate design and increase efficiency. The first step in the process was developing a Gantt chart and planning the project from the beginning of the academic year to the competition in Wichita, KS. Afterwards, the group developed a QFD (Quality Function Deployment) in order to determine the customer requirements, resulting engineering requirements, and the relation between the customer and engineering requirements. The relationship, or weighting, between the requirements was very important, for it produced the most important requirements. By knowing the most important requirements, design

trade-offs were easier to evaluate and execute. Each individual team used tools learned in ME401: Introduction to Mechanical Design and taken from Robert Ullman's book *The Mechanical Design Process* to gain a scope of the problem and requirements for their respective part of the overall design. Design tools such as objective tree analysis, functional decomposition, and morphological charts narrowed the scope of the problem for each team and produce a set of concepts.

The fuselage team produced four main areas of focus: ability to carry payload, size, connections (with the wing, landing gear, electronics, etc.), and aerodynamics (reducing drag as much as possible). The resulting concepts produced all had the same general shape of a long, cylindrical tube. The shape was driven by the constraint of carrying the softballs at least two abreast. However, the difference in designs resulted in the front of the aircraft. The different nose configurations included a bullet-shaped design, pyramid, and the whole fuselage being one big oval (similar to a lifting body). The payload team generally concentrated on determining how the different payloads would affect the weight, and center of gravity, as well as the size need for the fuselage. The payload team considered egg-cartons and tubes as ideas to base the payload design on. The landing gear team developed two main constraints: supporting the aircraft's weight upon landing and before takeoff and enough stiffness for the propeller to not impact the ground upon landing. The team produced three main landing gear designs: a tricycle type gear, a tail dragger, and a three-wheeled gear with each piece being separate.

The wing and tail team brainstormed numerous different concepts in developing the design. The research of the team looked into possible sizes (chord and span) and airfoil types. The airfoil was based primarily on the Reynolds numbers for which the plane would be operating. Since we are operating below Mach 0.3, we considered the compressibility effects to be negligible. To increase the aerodynamic efficiency of the airplane, winglets and canards were among the concepts considered. In addition to the main wing, the tail concepts included a split tail, traditional tail, and a T-tail. Finally, the electronics team researched the power needed to produce the amount of thrust needed to make the airplane fly. The battery weight, size of motor, and propeller size were all considered during the conceptual phase. Calculations were done for all of the components using material properties, theories, equations and procedures learned in our undergraduate education.

1.2 Preliminary Design. The preliminary design involved the evaluation of the concepts and the research into necessary materials and the best materials that would work for the design. The tools used included Pugh's method (to evaluate concepts), MATHCAD, Microsoft Excel, Pro/Desktop, grunt work in the laboratories, and the wisdom of our adviser and laboratory technicians. At the end of the preliminary design process, the team had an idea of what it wanted to build, the materials it needed, and were ready to being building, make necessary modifications, and testing.

The final fuselage design is essentially a bullet-shaped design, able to hold all twenty-four softballs. The cross-section of the fuselage was minimized to a size that was just enough to hold the softballs and house connections for the wing, landing gear, batteries, servos, and motor. The minimization of the fuselage cross-section reduces drag, weight, and cost. The construction plan for the fuselage was to make a mold on a rapid-prototyping printer, wrap the shell in epoxy and carbon fiber, and then pull the shell out. Running the length of the fuselage would be a thin, rectangular aluminum spar, which would provide a place for the connections of the batteries, motor, wing, landing gear, and tail. The length of the spar (and essentially most of the aircraft) is approximately five feet. On the top of the fuselage, behind the wing and landing gear, the design called for a door to open and allow the placement of softballs in the fuselage. The CG of the aircraft was designed so that when the softballs were place in the aircraft, twelve would be on either side of the CG. The reasoning behind this design consideration was to not affect the CG too greatly whether flying with or without payload. Inside the fuselage, an inch-thick layer of foam would be laid to prevent the softballs from shifting in flight and disrupting the static stability of the aircraft. The placement of the CG was as forward as possible, in order to give the plane the greatest amount of longitudinal pitch stiffness.

The goal of the wing and tail team was to produce a wing design with the best lift to drag ratio for the operational Reynolds number. Numerous airfoils were researched, for both the tail and main wing. Calculations were made in MATHCAD and drag polar charts were researched to determine the most efficient airfoil for the design. The final decision was an SD7062 airfoil set on the aircraft at a four-degree angle of attack. The span of the wing is ten feet with a root chord of eighteen inches and a tip chord of nine inches. For the tail, a symmetric airfoil and a T-tail design were the best choices, especially for the vertical fin/stabilizer. The symmetric airfoil (NACA 0012) will produce the same amount of lift on either

side, which is ideal for the control of the airplane. The calculations by the wing and tail team were given to the electronics team in order to determine the power requirements for the airplane. The electronics researched the battery and motor choices available under the constraints of the competition rules. The motor and battery needed to be as light as possible, while still providing enough power and thrust to fly the plane for the total mission time of ten minutes. The motor needed to produce enough power to sustain the maximum rate of climb and cruising flight. Additionally, the battery must provide enough juice to maintain the motor power at all mission requirements. The final choice for the motor is an Astro Flight 691S with a SR2400 battery pack.

The landing gear concepts were developed and evaluated with extensive time spent by the landing gear team in the machine shops with the laboratory technicians. The design finally decided on was a tricycle style landing gear. The landing gear design needed to be a tall design, in order to ensure proper propeller clearance. The final choice was to use some sort of Aluminum (further research was required) for the landing gear with two legs with one wheel each in the back, and a single landing gear with two wheels in the front. An important consideration for the wheels was their ability to withstand a heavy load upon landing.

1.3 Detailed Design. The detailed design phase was largely experimentation with different building techniques and refinement of the previous design ideas to produce the most efficient airplane while working within our $3,000 budget. CG calculations were done in order to determine the plane's static margin and resulting amount of static stability. The end of the detailed design produced the following desired performance criteria:

Criteria	Performance
Maximum Rate of Climb (RC)	18.4 ft/s
Power at Maximum RC	1576 watts
Velocity at Maximum RC	74.6 ft/s
Maximum Take Off Distance (full payload)	199 ft
Total Lift Produced	45.7 lbs
Total Drag Produced	0.558 lbs
Aircraft Weight	35 lbs

Table 1.1 Performance Criteria

The detailed design phase produced the following changes to the overall aircraft design. The rapid prototype printer could not produce the sizes necessary for the fuselage mold. As a result, the fuselage will be made out of aspen wood frame with a wooden spar in the middle. The spar will be coated with

epoxy and fiberglass to ensure plenty of strength. The spars on the bottom of the fuselage were epoxied to the bottom of the main spar with two aluminum rods running through the spars. The aluminum rods provided two purposes in the fuselage. The rods provided a connection space for the landing gear, wing, tail, and motor mount. The rods extended from the front and rear of the main body of the fuselage for the motor mount and tail connections. The bottom of the fuselage was coated with aircraft plywood with epoxy and fiberglass fibers for added strength and resilience. The entire fuselage, from top to bottom, will be coated with Monokote to reduce drag and improve the aesthetics. The electrical connections required for the tail's control surfaces are to run down the center of the fuselage between the rows of softballs.

The wing will be constructed of Spyder foam with carbon fiber rods for added strength and stiffness. The rods will be placed in cut outs on the top of the wing, with expansive foam used to keep the rods in place. The wing was divided into four sections on either side, with each section being held together by epoxy. The wing will be coated with Monokote, and will provide space for the electrical connections and control surfaces. The control surfaces will be NACA 0012 airfoils attached in spaces on the wing, vertical stabilizer, and horizontal stabilizer. The tail will be attached behind the end of the door for the softballs. The tail, both horizontal and vertical stabilizers, will be made of NACA 0012 airfoils of varying chord lengths and taper ratios. Through all of the testing, we hope to decrease the weight of the aircraft and takeoff distance, while finishing the missions in the least amount of time possible.

2. Management Summary

The project group/design team, nicknamed Team Darkhorse, consisted of eight seniors, three juniors, and one sophomore. Six of the seniors are aerospace engineering majors, one is a mechanical engineering major, and the last senior is an electrical engineering major. The four underclassmen were all either mechanical or aerospace engineering majors. The team's mission was to beat the other competing team at the Academy in a Fly-Off and then go on to win the competition in Wichita, Kansas.

The design team broke down into five subsections: wing and tail, fuselage, payload, landing gear, and electrical components. Table 2.1 shows the individual team members and their assignments. The team met once a week during the first three months of the project, the conceptual design phase of the project. The team met as a whole for progress and discussion of the connections for each of the individual subsections. In addition to entire project group meetings, the individual subsections met together as much as necessary to hammer out problems and ensure the best possible design.

Name	Primary Assignment	Secondary Assignment
Christopher Orlowski	Project Manager	Misc. Construction
Neal Erickson	Executive Officer	Fuselage Team
Kristine Conner	Budget Officer	Wing and Tail Team
Jason Sienko	Wing and Tail Team	Misc. Construction
Kevin Tobias	Fuselage Team	Misc. Construction
Lucas Meyers	Payload Team	Fuselage Team
Frank Hauben	Electronics/Motor	N/A
Bruce Skinner	Landing Gear	Misc. Construction
Jared Koelling	Misc. Construction	N/A
Joseph Haury	Misc. Construction	N/A
Amber Raub	Misc. Construction	N/A
Alfredo Gonzales	Misc. Construction	N/A

Table 2.1 Team Assignments

During the preliminary and detail design phases, the individual teams met as much as possible in order to accomplish the goals and deadlines of the project. The Gantt chart below shows the intended plan for the entire project. Unfortunately, not everything went as planned. The final term briefing did not take place until December, the schedule of other academic requirements pushed it back. As a result, the majority of parts were not ordered until after the Christmas holiday. The second semester did not start off well either. The first two weeks were very busy, and not enough time was devoted to working on the project. Supply problems with ordering and delivering of materials, as well as some project management

problems resulted in the plane not being assembled by the 28th of February. The goal is to have testing complete by the 19th of April still and the finished plane on the 11th of March.

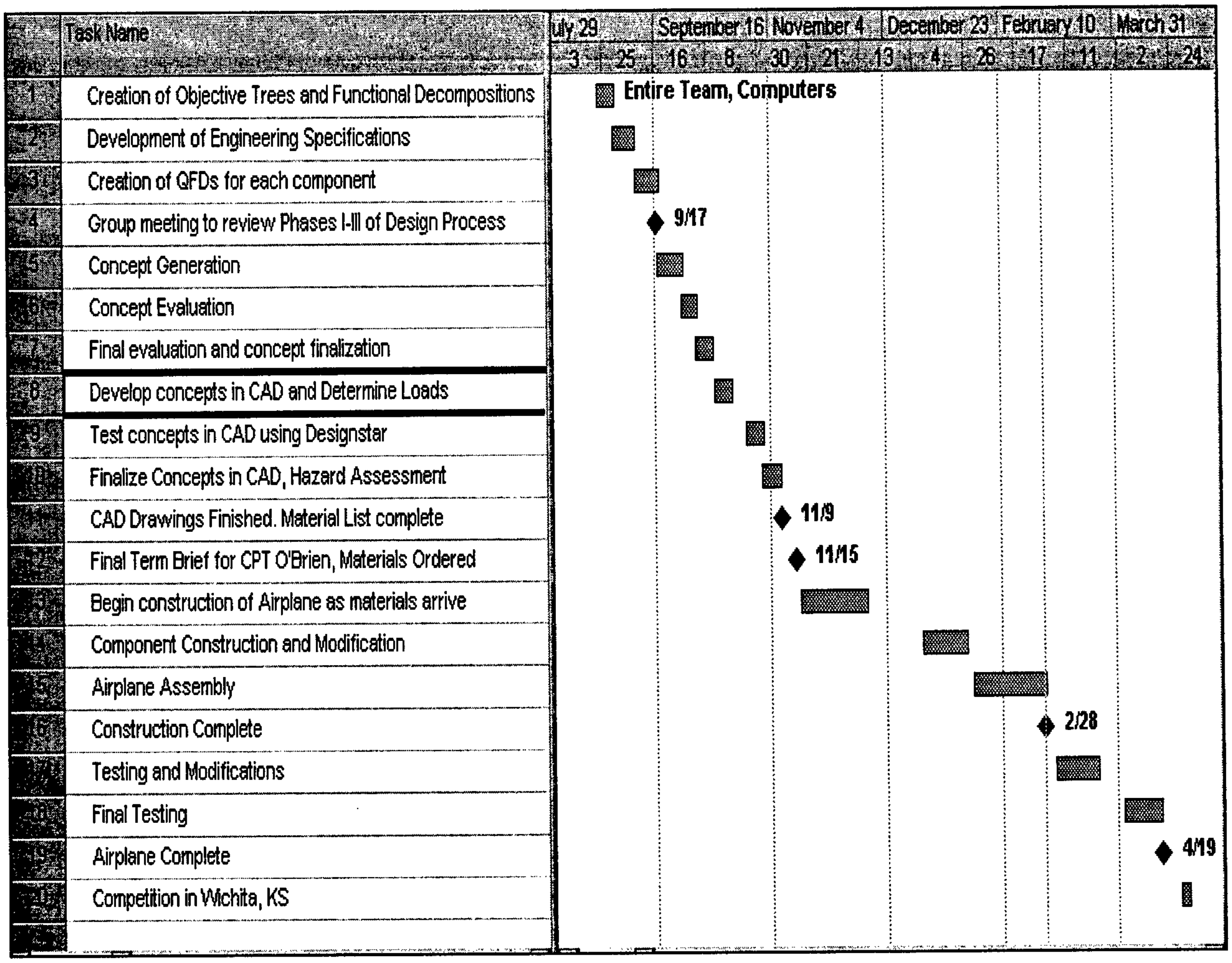


Figure 2.1 Project Plan (Gantt Chart)

After the project was planned, the scope of the project was narrowed using design tools such as functional decompositions, mind mapping, and objective tree analyses. The combination of these design tools, interviews, and past experiences combined to produce customer requirements. The customer requirements were weighted using pair wise comparison and the following results were produced:

Customer Requirements	Weight
Cost	10
Weight	12
Stability	13
Durability	12
Easy to Repair	7
Interaction Between Components	3
Balanced	13
Height on Ground	1
Wing Easily Attach to Fuselage	9
Sturdy/Strong	19
Survivable	16
Fuselage Aerodynamic	11
Minimize Space/Size	7
Fuselage Weight	12
Absorb Landing without Failure	17
Support Airplane	20
Strength(LG)	21
Capacity for Softballs	9
Efficient Loading Time	3
Payload Weight	8
Motor Size	5
Motor Weight	4

Table 2.2 Customer Requirements

The customer requirements were translated into engineering requirements, per subsection of the design team. Using the customer requirements and engineering requirements, the relative importance of each engineering requirement was calculated using a Quality Function Deployment, or QFD. Knowing the relative importance of each of the engineering requirements allowed us to make design decisions later on in the design phase. The planning and gaining a scope of the project was an important part of the management project for it provided a good head start and gave a springboard to being the conceptual design phase of the project.

3. Fuselage Design

3.1 Conceptual Design. Initially, there were several options to consider when building a fuselage. We knew from past years that the maximum load of 24 softballs (approximately eight pounds) was a healthy load and was often not attempted because of its size and its weight. Due to this fact, we considered a lifting body design, which would essentially be a large, hollow wing. Another option was to make the fuselage an oval, but, in the end, we determined that the same manufacturing difficulties would hinder production:

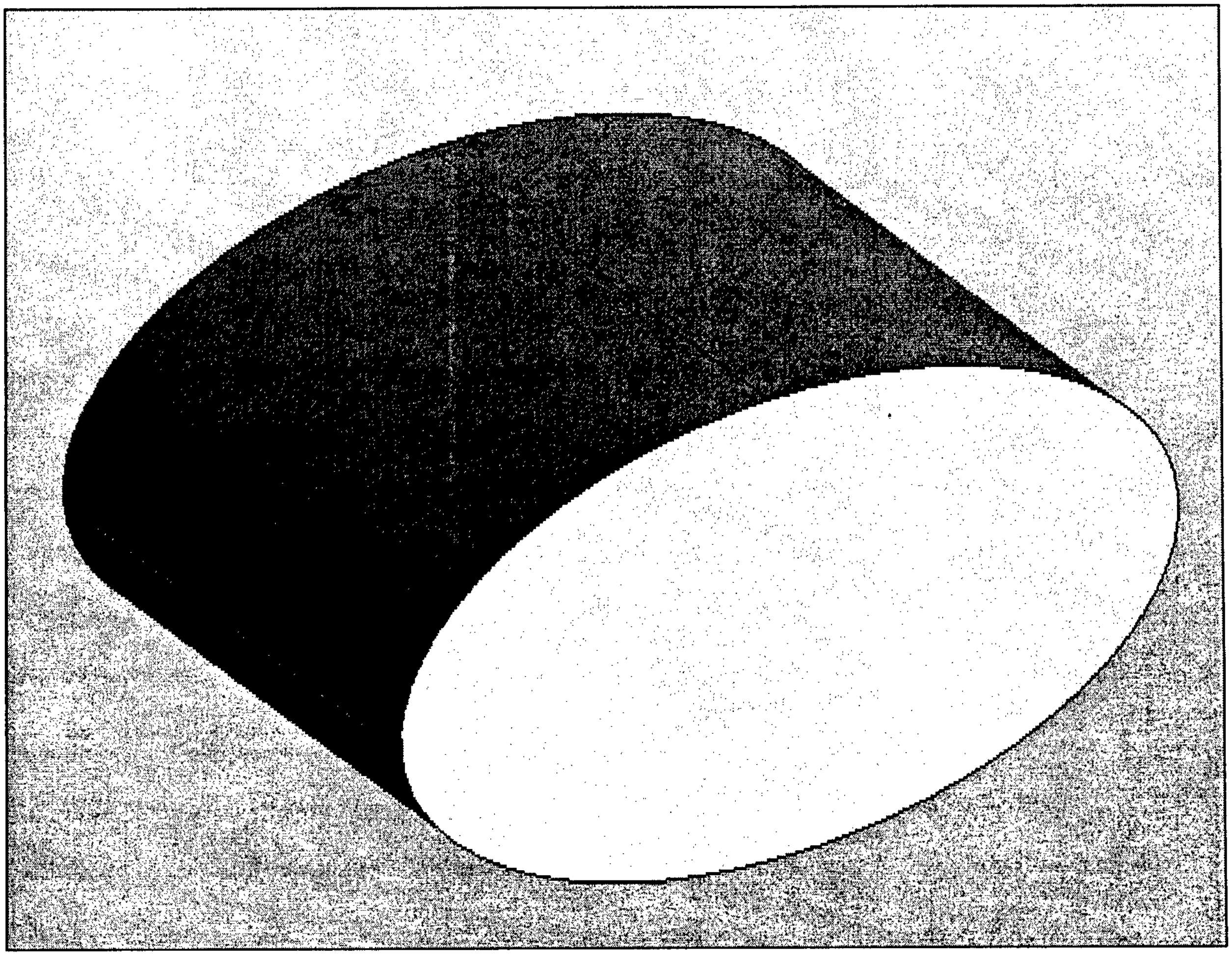


Figure 3.1 Fuselage Concept #1

However, considerations were made for the manufacturability of such a fuselage. On such a large scale, the airfoil shape would be difficult to create and determine how much lift and drag it would produce. This

would affect our rated aircraft cost in that the manufacturing hours would increase significantly. Other options were variations of a generalized cone shape are shown below:

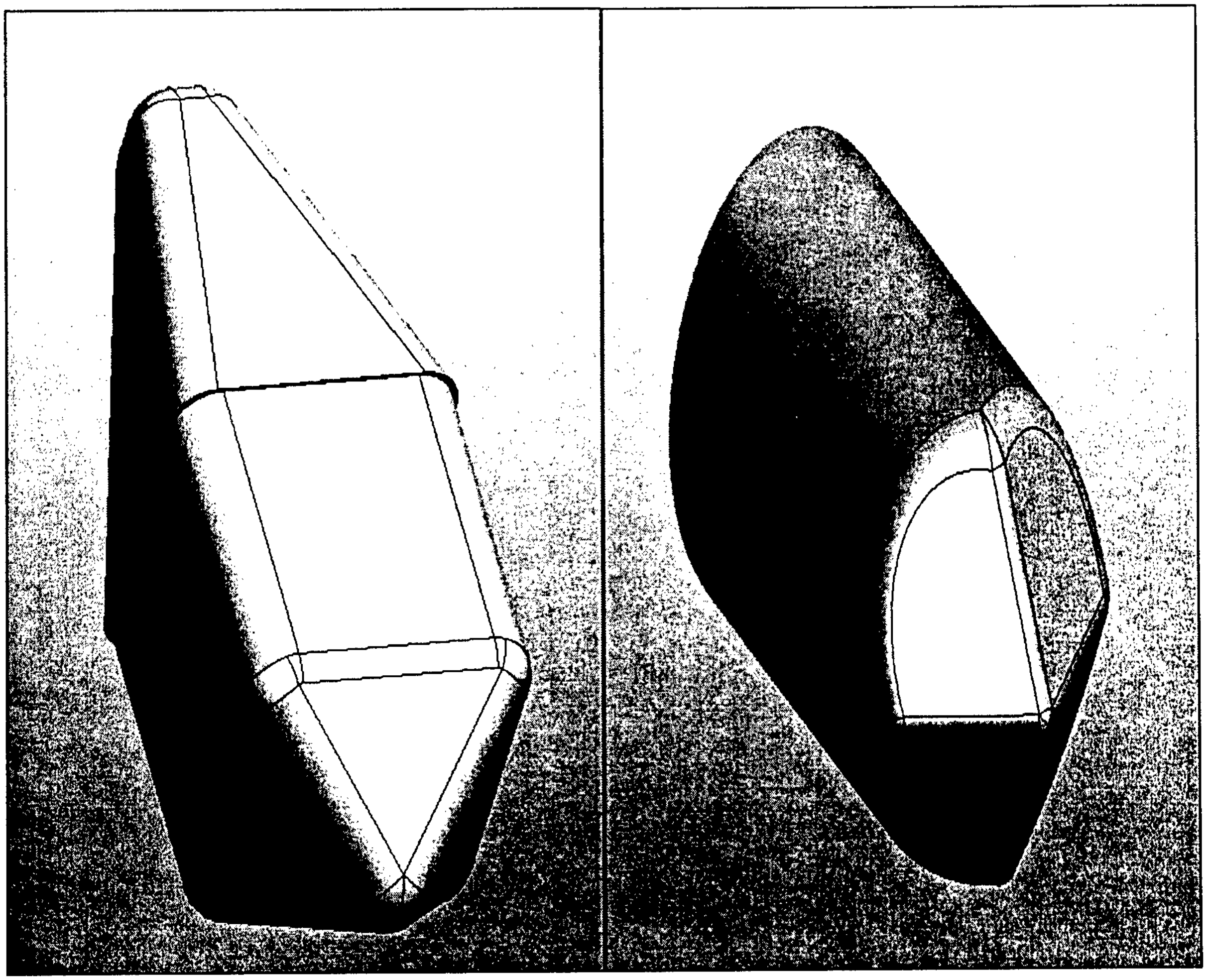


Figure 3.2 Fuselage Concepts #2 and #3

After these ideas were brainstormed, they were determined to be easier to construct. However, the reason we did not pick these was because of the internal connections and inefficient use of internal space. The shapes were boxy and would require fairly complex internal connections to both support the payload and also connect the battery pack. We felt that this could be done, but would require a decent amount of material, which would add weight. Connections between these internal devices would provide stress concentrations around the shell, which is inherently undesirable. The rated aircraft cost would most likely be very good for a shape such as this because they were not designed to be very long; however, the poorer performance, as discussed earlier, outweighed this consideration.

Further improving on this design would be a bullet-like shape. Inherently, this is the shape of choice; since bullets are generally have a high aerodynamic efficiency. Fuselages on passenger airplanes are shaped roughly like bullets. The shape of fuselages on commercial aircraft is inherently efficient with respect to drag due to nearly 100 years of design and redesign. The bullet design proved efficient for our needs. We modified it slightly from a circular cross section to two superimposed circles to save space.

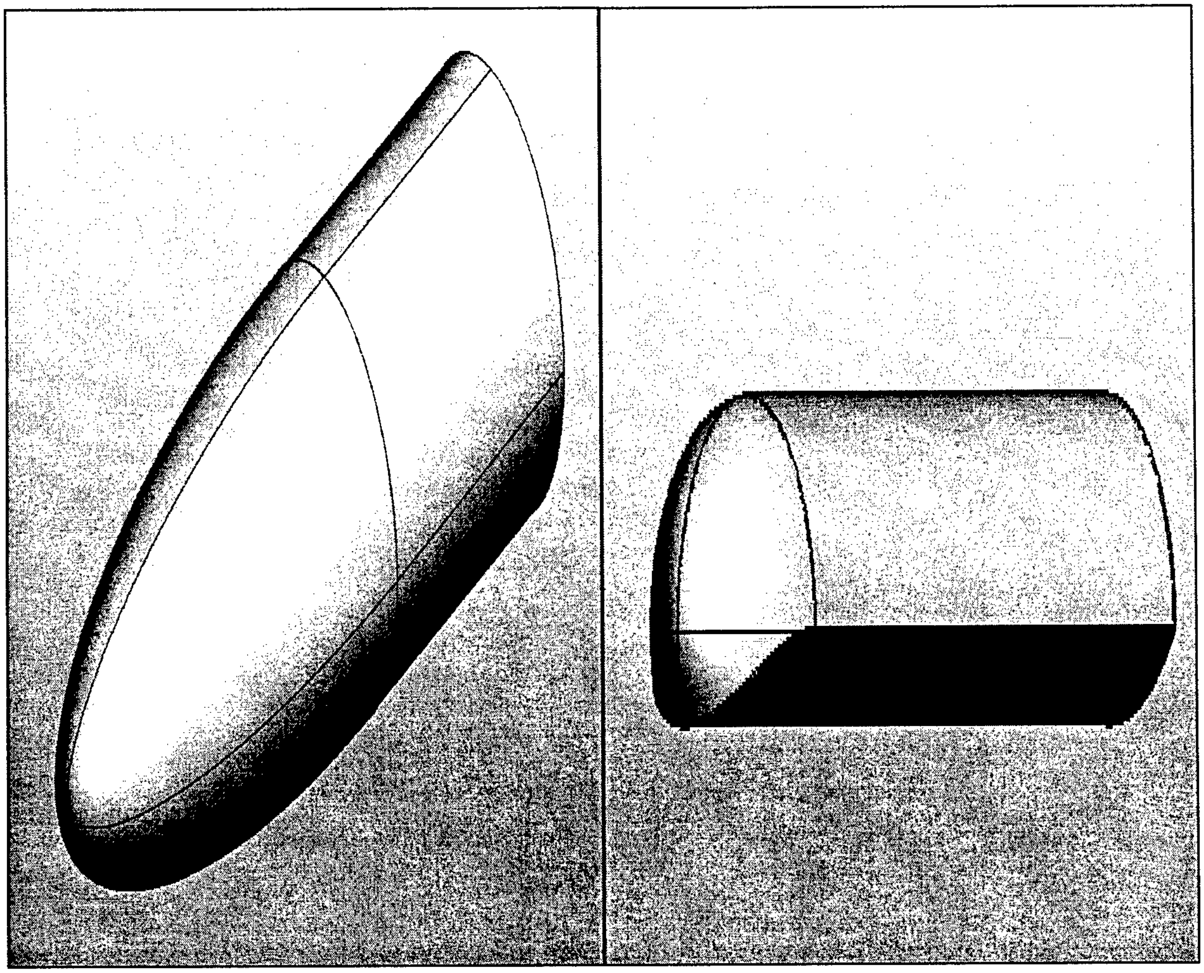


Figure 3.3 Fuselage Concepts #4 and #5

The design on the right is our final design and is a simple bullet shape consisting of a 7-inch radius circle superimposed on a 5-inch radius circle. The rated aircraft cost would increase slightly because of the length of the fuselage, but the compactness of used space and efficiency by which this design would operate would serve as great advantages when flying.

3.2 Preliminary Design. The design parameters that we investigated were the packing efficiency of the softball payload, placement of batteries, and the connections between the fuselage and all of the plane's

component parts. The decision to stack the softballs in a 2x12 fashion was done through trial and error. One long row of softballs stacked one behind one another would have made the fuselage nearly 10 feet long. This would create an enormous bending moment about the center of gravity simply due to the weight of the fuselage. We felt that any other arrangement (3x8 or smaller) would create more pressure drag and that the problems with a bending moment created by a 2x12 arrangement were outweighed by the pressure drag created by a fuselage with a larger cross-sectional area. The calculations for different battery configurations is shown in the following table:

Configuration for 10 (2x5):		Configuration for 12 (2x6):	
Width of Container=	7.6396	Width of Container=	7.6396
Length of Container=	19.099	Length of Container=	22.9188
Weight=	3.563 lbs	Weight=	4.275 lbs
Area=	145.9087	Area=	175.0905
Configuration for 14 (2x7):		*Configuration for 16 (2x8):*	
Width of Container=	*7.6396*	*Width of Container=*	*7.6396*
Length of Container=	*26.7386*	*Length of Container=*	*30.5584*
Weight=	*4.988 lbs*	*Weight=*	*5.7 lbs*
Area=	*204.2722*	Area=	*233.454*
Configuration for 18 (2x9):		*Configuration for 20 (2x10):*	
Width of Container=	*7.6396*	*Width of Container=*	*7.6396*
Length of Container=	*34.3782*	*Length of Container=*	*38.198*
Weight=	*6.413 lbs*	*Weight=*	*7.125 lbs*
Area=	*262.6357*	Area=	*291.8174*
Configuration for 22 (2x11):		*Configuration for 24 (2X12):*	
Width of Container=	*7.6396*	*Width of Container=*	*7.6396*
Length of Container=	*42.0178*	*Length of Container=*	*45.8376*
Weight=	*7.838 lbs*	*Weight=*	*8.55 lbs*
Area=	*320.9992*	Area=	*350.1809*
Configuration for 12 (3x4):		Configuration for 15 (3x5):	
Width of Container=	11.4594	Width of Container=	11.4594
Length of Container=	15.2792	Length of Container=	19.099
Weight=	4.275 lbs	Weight=	5.344 lbs
Area=	175.0905	Area=	218.8631
Configuration for 18 (3x6):		Configuration for 21 (3x7):	
Width of Container=	11.4594	Width of Container=	11.4594
Length of Container=	22.9188	Length of Container=	26.7386
Weight=	6.413 lbs	Weight=	7.481 lbs
Area=	262.6357	Area=	306.4083

Table 3.1 Payload Configurations

The chart shows the small increase in length per design from a 2x5 configuration up to a 2x12 configuration. Therefore, the 2x12 minimizes the form drag of the airplane, in lieu of a 3x or 4x configuration. The placement of the batteries and motor was fairly simple. They would be placed in the

nose of the aircraft and their weight would ensure that the center of gravity of the plane remained in front of the aerodynamic center (and hence, statically stable). The placement of center of gravity was important, we did not want it to shift backwards, or to far forward, when the softballs were added or removed. If the CG shifts backwards, the airplane could become statically unstable. Likewise, if the CG shifts too far forward, the airplane may be too stable and difficult to maneuver. As a result, the CG placement was designed so that it would shift less than two inches towards the rear when the softballs were removed, while still remaining statically stable.

The overall design of the fuselage was based around leaving as little empty space inside the shell as possible, making it very light and minimizing the drag that will be produce by its travel through the air. We initially had a simple, round bullet shape with the softballs occupying only half of the cross-sectional area. We realized that the design was very inefficient with respect to internal space, so the superimposed circles idea cut down on unused space. All other aspects of the fuselage were maintained and designed around the size of the motor and the battery pack (which could easily fit inside the area occupied by two softballs side-by-side).

We made a few, fairly valid assumptions when designing the fuselage. The first was that we could construct it in such a manner that produced a smooth surface so as to decrease skin friction drag to a point where it would not play a dominant role in overall drag. The second assumption was that, due to the placement of the wing and the center of gravity, the bending moment around the shell of the fuselage would be comprised mostly of its own weight. In equilibrium flight, the lift vector pointing straight up will have a very small moment arm (only a few inches) and, with the materials we chose, will not play a significant design role in constructing the fuselage. The third assumption we made was that, when statically stable, the maximum upward force the fuselage would see would be 50 pounds, the maximum set by the wing team. Using this assumption, instilled a factor of safety in the airplane design for the final weight should be less than 40 pounds. Other small assumptions were made in constructing the fuselage, but all ended up being over-designed rather than under-designed (steel hinge for the door, nuts and bolts, other items over-designed out of convenience rather than out of calculation for yielding).

Our goal with the fuselage focused heavily on maximizing the space available for internal components and the 24 softballs, while minimizing the size of the fuselage. The fuselage is designed to

adapt to any manufacturing problems that we will run into including an unexpected shift in the center of gravity of the aircraft that occurs during the construction. Basically, before we mount the wing and landing gear we will be able to determine the exact location of the center of gravity of the loaded and unloaded fuselage with the motor and batteries included. The batteries weigh enough at five pounds that by shifting the location of the batteries, we will be able to align the loaded and unloaded center of gravity. There will be some error, but it will be minimized by proper determination of the CG.

Figure 3.4 Center of Gravity Diagram

Our theoretical CG was calculated to be 19.46 inches behind the nose of the aircraft. We found this CG by finding the moment arm about the nose of the aircraft for each of the primary loads.

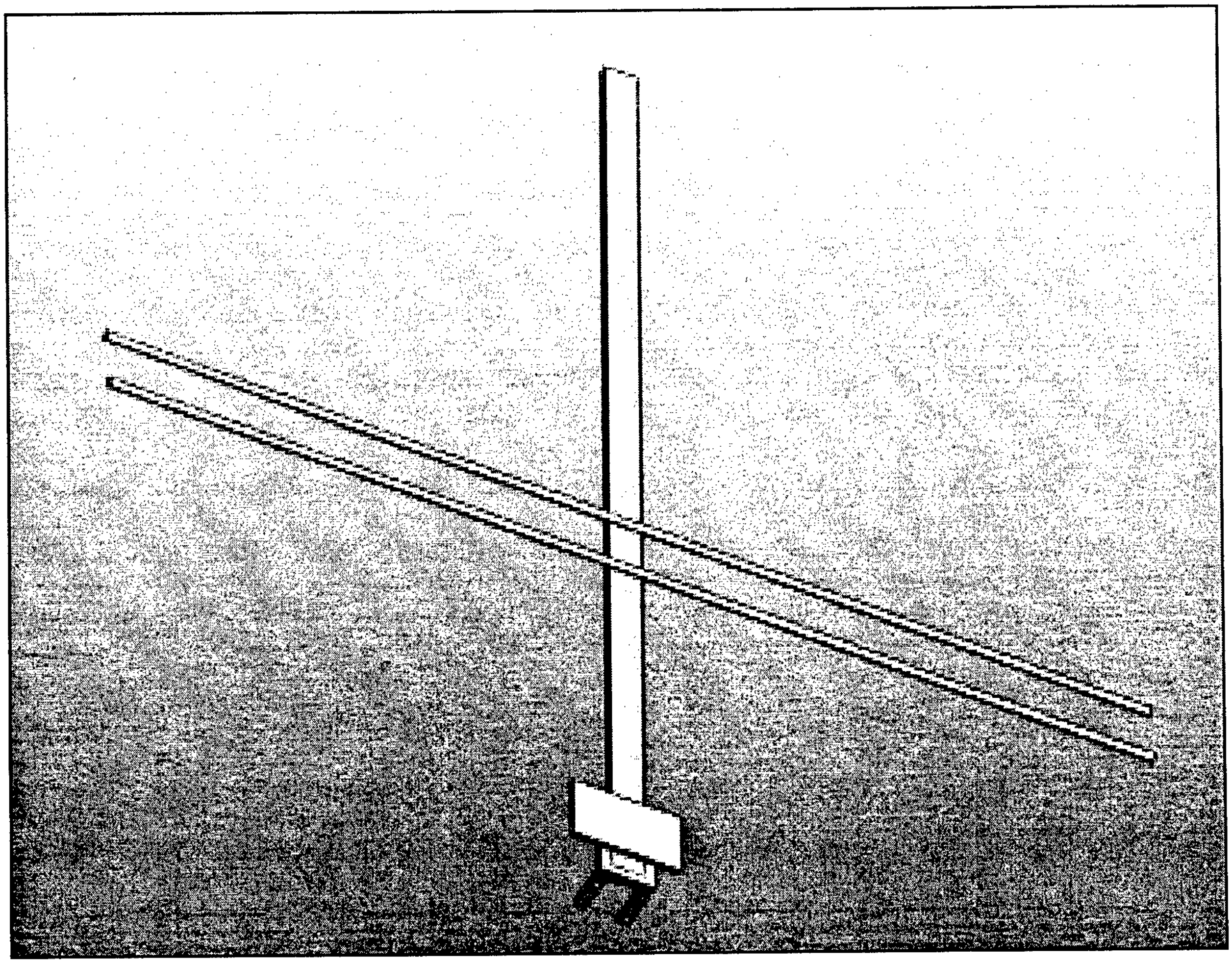


Figure 3.4 Internal Skeleton of Airplane

This layout shows the internal skeleton that we envisioned prior to construction and provided the basis for our construction plan. The wing location will move based upon our working model CG. The front shows the motor mount and the two rods are the carbon fiber rods in the wing.

3.3 Detail Design. The fuselage of the aircraft must be strong enough to withstand the forces created by the internal loads, as well as handle the rigors of flight and takeoff/ landing operations. To handle these forces, we are making our fuselage out of aspen to create an internal frame. Aspen is stronger than the model airplane standard of balsa wood, but has only a minimal increase in weight. The aspen is being used as the ribs of the aircraft and are placed every four inches along the length of the aircraft. The

spaces will be filled with low-density foam and will act as a buffer for the softballs that are also placed between the ribs. In order to make the underside of the fuselage even we are using 1/16-inch balsa wood panels to create the outer shell. The outer shell will then be covered in fiberglass and epoxy to give the fuselage the strength it needs to withstand external forces. We are also using a fiberglass-coated piece of wood as the center spar of the aircraft.

We considered many different options when determining how to build the fuselage. Possible designs included a wood frame with either Monokote or fiberglass outer shells, also an aluminum frame could be used, or a foam core with a fiberglass outer covering. Every design included either an aluminum center spar or the fiberglass coated wood spar. We went with the aspen wood frame, a fiberglass coated center spar and a fiberglass shell. The advantages of this design were that it was cost effective for the strength that we get from the fiberglass and it was easy to manufacture based on a truncated building schedule.

Through experimentation and concurrent engineering, the fuselage concept needed additions to improve the design and allow for connections to the airplane. Two aluminum rods were added to the bottom of the fuselage, and ran through each of the ribs. The aluminum rods extended out of the front and the back. The extended aluminum rods allowed for the connection of the motor mount in the nose and for the T-tail to the rear of the fuselage. A space in the bottom of the fuselage was provided for the connection of the carbon fiber rods and the landing gear directly to the aluminum rods. Once all of the connections were made, the entirety of the fuselage will be coated in Monokote to reduce the drag created by only having the rough exterior of fiberglass fabric and an epoxy matrix.

4. Wing and Tail Design

4.1 Conceptual Design. Essentially, we began with three initial concepts for the design of our wing for the AIAA design, build, and fly competition. The goal was to make the wing as aerodynamic and efficient in flight as possible. Using this as a driving force, we looked at ideas such as winglets and canards, and a basic wing with no additions. While logically and theoretically thinking through our concepts, we looked at factors such as manufacturability, feasibility, cost, and overall impact.

Our first concept involved the use of winglets, and our purpose behind this concept was that the winglets would ultimately reduce the induced drag our wing would see, enabling the wing to sustain flight while creating less drag and using less power and battery life. The graph below shows the advantage of using different configurations of winglets as compared to a wing without winglets.[1]

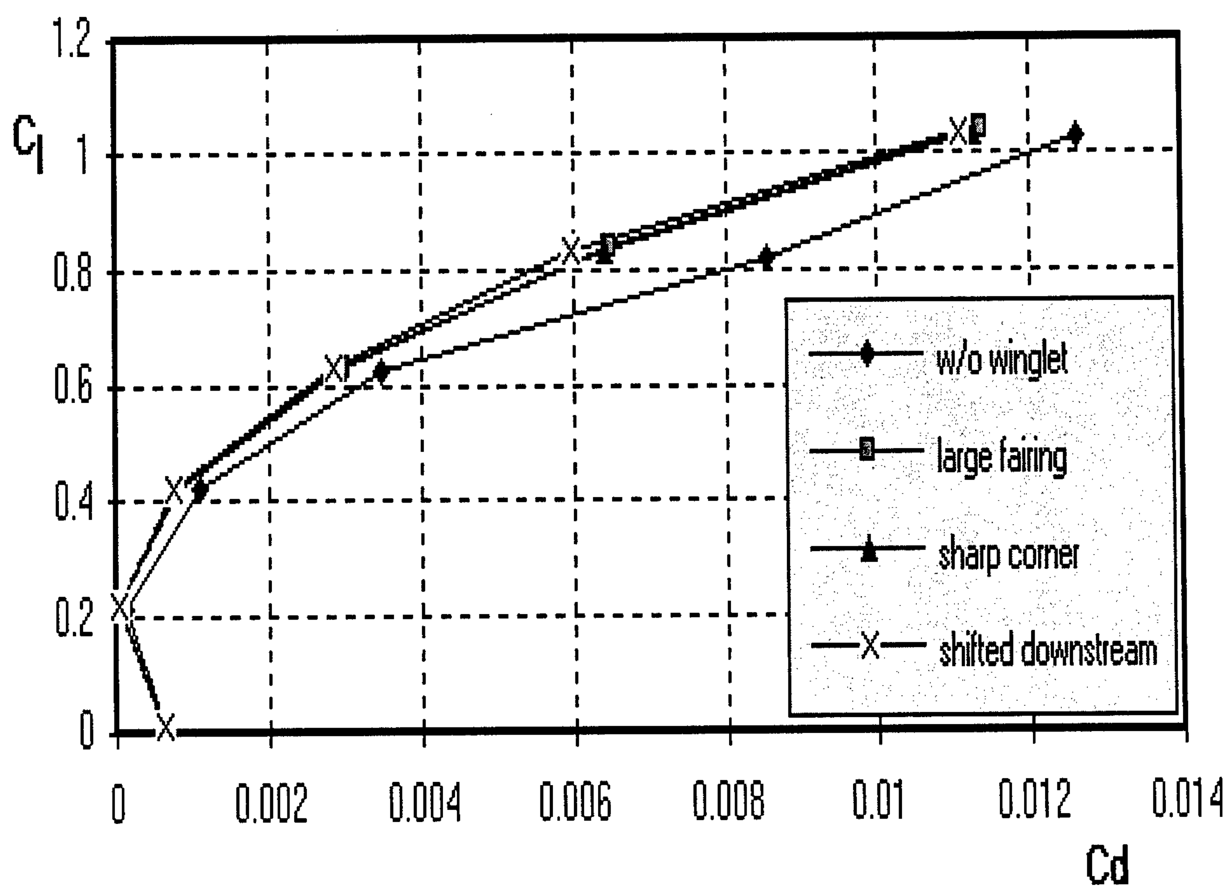


Figure 4.1 Advantages of Winglet Usage

[1] http://beadec1.ea.bs.dlr.de/Airfoils/winglt1.htm.

Our concept configuration was a rounded end winglet as can be seen by the pictures below:

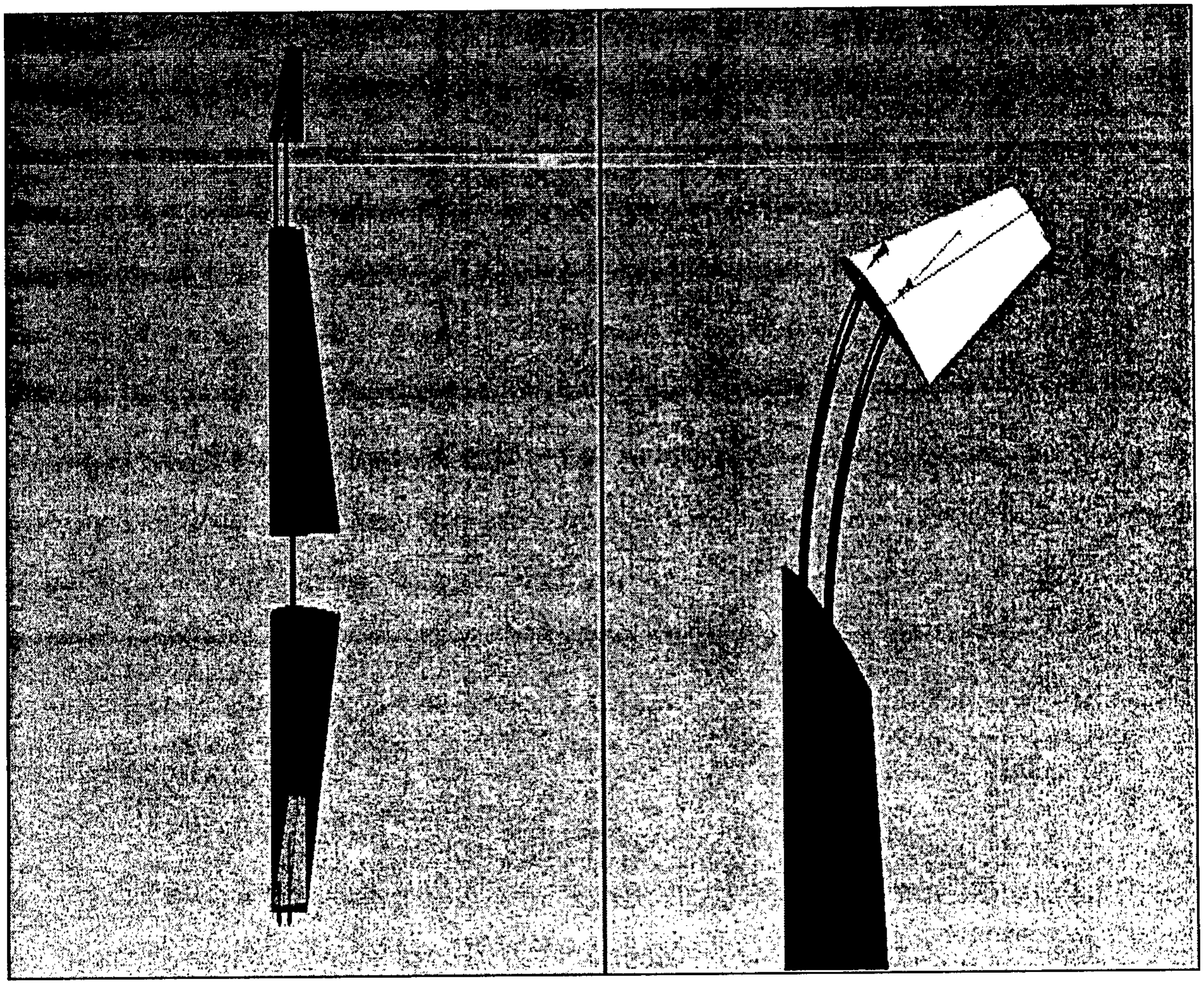


Figure 4.2 Winglet Concept (Full View) Figure 4.3 Winglet Concept (End View)

The overall impact of using winglets, as opposed to a basic wing without winglets, would have been more efficient, due to the decrease in induced drag. However, other factors led us to disregard this concept. In reality the manufacturability of placing winglets on the end of our wing would be very difficult, and given our time constraints and resource availability, we did not think this would be easily manufactured. In terms of cost, winglets would significantly add to the total rated aircraft cost.

Our second concept involved the use of canards on the front of the wing to help sustain flight. After some discussion, we discovered that the overall impact of this concept for our specific purpose would actually hold no benefit. The purpose of canards is to give the plane more horizontal stability at higher speeds, which it does by increasing the total wing surface areas of airplanes in order to improve their low-speed lift-to-drag ratios and trim characteristics. This doesn't really apply to our remote

controlled airplane, since we are not flying anywhere near supersonic speeds. The picture below depicts the purpose of canards and how they affect aircraft.

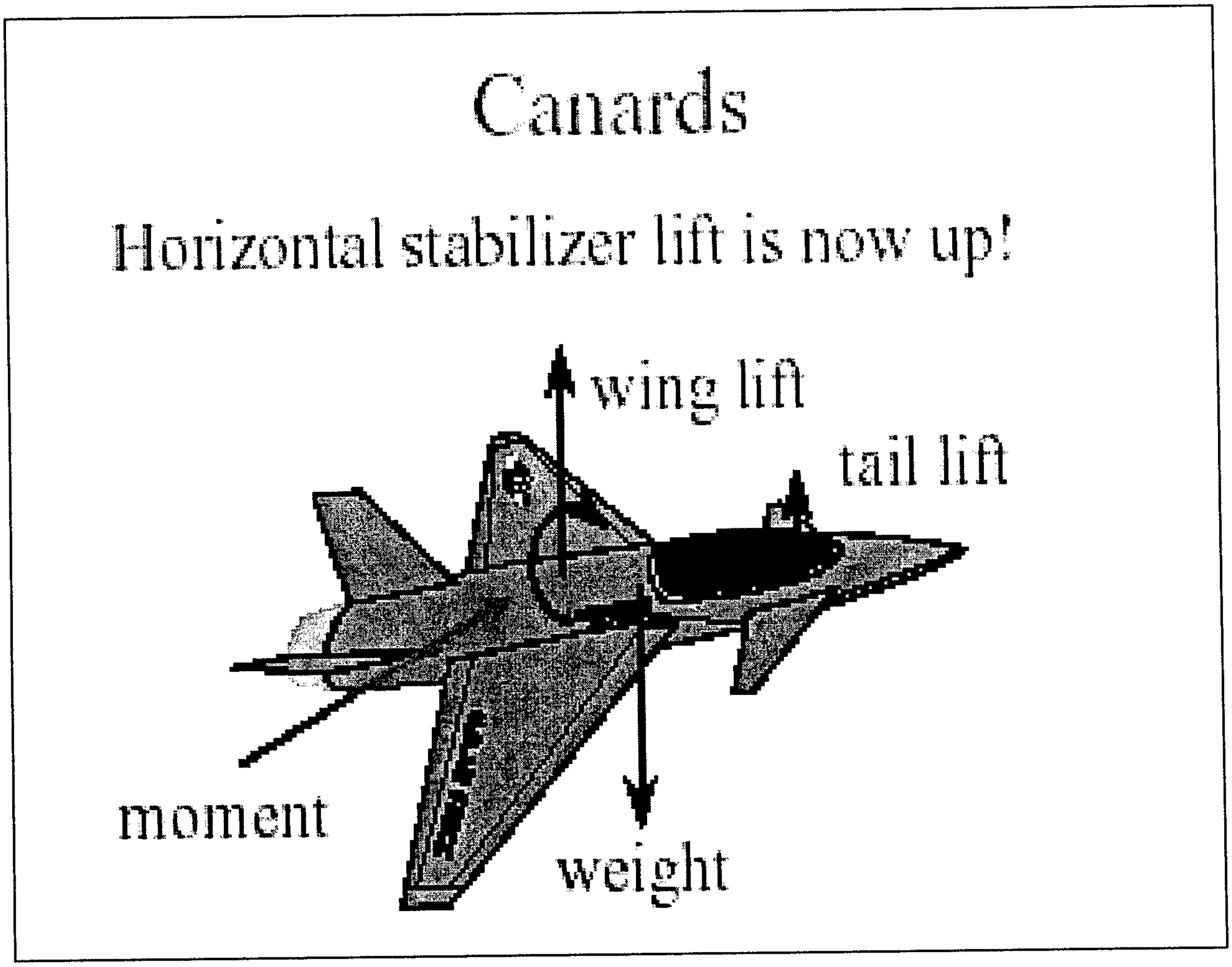


Figure 4.2 Canard Concept

Not only did we disregard this concept due to its impact not being overly beneficial, but also because of the difficulty in creating such a surface for the wing. Having little knowledge of how to construct canards also significantly impact our decision, as well as the added cost and weight it would place on our aircraft.

Our final concept was a basic tapered wing with no wingtips, canards or any similar additions. The basic wing concept proved to be the best plan for the airplane. The ultimate reason we chose this wing is because it had the most realistic and sensible benefits for our purpose. Given our time frame, resources, cost, and probability that we would have to construct more than one wing, we felt a basic wing

was our best choice. Below is a chart of how we ultimately chose our concept, through the use of Pugh's Method.

Customer Requirements	Weight	Concept 1	Concept 2	Concept 3
Easily attached to fuselage	9	D A	0	+2
Sturdy/Strong	19	T	-2	0
Survivable	16	U M	-1	-2
Total		0	-54	-14

Table 4.1 Design Matrix (Pugh's Method) for Wing Design

4.2 Preliminary Design. For the preliminary design of our wing, we first had to decide on an airfoil to suit our plane's needs, which involved making various assumptions. The assumptions we made inquired we would be flying at sea level, our plane weigh 40 lbs, it would travel at a velocity of 62 ft/s, and that we were traveling in straight and level flight for the majority of the competition. By making the airplane the most efficient at straight and level flight, we would save the most battery power since the airplane will be spending the majority of its mission time in, or close, to this condition. We also assumed taper ration of 0.5 for the main wing and a chord length of 18 inches, which resulted in a tip cord of 9 inches for the 10-foot wingspan. Using these assumptions allowed us to make the appropriate calculations and enable us to determine which airfoil would best meet our requirements.

4.3 Detail Design. We investigated numerous airfoils to determine the best choice for use on our airplane. Based on our assumption of aircraft weight and velocity, we were able to calculate the amount of lift, and therefore coefficient of lift, needed for the main wing. Using these assumptions and subsequent calculations, Based off our assumptions and calculations we determined from various drag polar plots that the SD7062, which has a coefficient of lift of .9 and a coefficient of drag of .011, was our best choice for an airfoil. Also, based on our assumptions and characteristics of our chosen airfoil, we determined our power required for straight and level flight, which we found to be 46.6 W. Compared to our conceptual designs, there were no differences in our basic assumptions that we would be flying in straight and level flight at sea level. Once we determined our airfoil shape and angle of incidence that would generate the amount of lift desired from the drag polar curve in Figure 4.3, we were able to develop the drag polar equation of our aircraft.

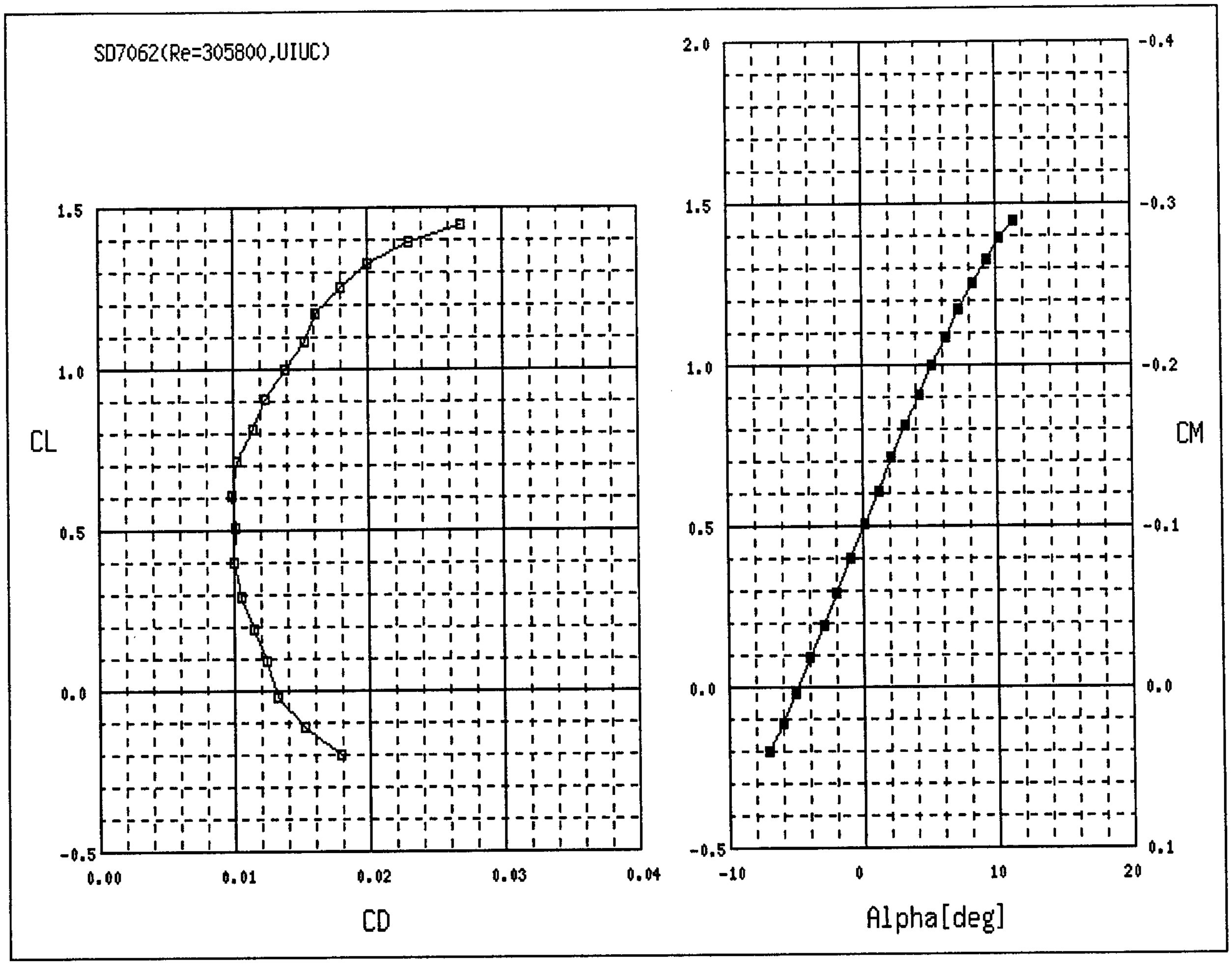


Figure 4.3 Drag Polar for SD7062 (Re:305800)

Utilizing the maximum values from Fig. 1, the drag polar equation is: $C_D = 3.705 \cdot 10^{-3} - .038 \cdot C_L^2$. This equation was used to calculate the rest of the aircraft performance properties listed in Table 4.2. To calculate these properties, we assumed straight and level flight at 61.6 ft/s (42 mph) and 40 lbs for the maximum payload. Our goal is to have a gross weight of 35 lbs, but to compensate for any errors and introduce a factor of safety; we decided to determine our properties for 40 lbs.

Once the maximum gross weight was determined, we wanted to design the wing to carry a payload of 45 lbs as an added safety measure. We also wanted to minimize the wing surface area while staying within the constraints of the take-off distance set by AIAA. Therefore, we iterated our performance in MATHCAD to achieve the most efficient design.

Aircraft Performance Properties			
Planform Area, S:	11.25	ft^2	
Root chord, c_r:	18	in	
Tip chord, c_t:	9	in	
Aspect Ratio, AR:	8.889		
Span, b:	120	in	
Reynolds Number, Re:	4.57E+05		
Thurst Required (T_R):	7.822	oz	(Straight and level flight)
Power Required (P_R):	46.631	oz	(Straight and level flight)
$(L/D)_{max}$:	42.308		
Max Rate of Climb :	18.390	ft/s	
Velocity, Max R_C:	74.207	ft/s	
Angle, Max R_C:	14.537	degrees	
Velocity, Stall:	42.572	ft/s	
Max Take-off Distance:	199.771	ft	
Minimum Turn Radius:	106.892	ft	

Table 4.2 Aircraft Performance Parameters

Using the assumptions of standard temperature and pressure, we calculated the lift coefficient needed for straight and level flight at 61.6 ft/s with a 40 lb gross weight. The lift coefficient is .788. At a zero angle of attack, most low-speed airfoils do not have a lift coefficient of 0.788 at our operational Reynolds number. Therefore, by placing the wing at an initial angle of attack, or incidence, we are able to gain the needed lift coefficient. Before the angle of incidence was applied, we looked for the airfoil with the greatest L/D. Specifically, we researched the following airfoils: SD7062, SD7032, E387, and CLARK-Y012. The SD7062 had the highest lift coefficient with 1.65, but CLARK-Y had the lowest drag coefficient. The other two had values in between. We chose SD7062 because it had a much higher lift coefficient than the CLARK-Y while sacrificing a negligible amount of drag. As a result, the SD7062 had the highest lift-to-drag ratio.

In choosing an angle of incidence, we wanted to minimize the amount of wing loading. We determined the RAC would be lowest with a wing loading value of between 1.1 and 1.2. Using this value in our calculations, the angle of attack (and hence coefficient of lift) that gave use the desired value was four degrees for the SD7062 airfoil. We could have chosen a higher angle of incidence, but it would decrease the rate of climb and the safety zone before the stall angle of attack.

The shape of the wing was chosen because it was a compromise between efficiency and manufacturability. It would be extremely difficult to manufacture a perfectly elliptical wing out of foam core. Additionally, it would be somewhat inefficient to make a perfectly rectangular wing. Therefore, we used a tapered wing with a taper ratio of 0.5. The taper ratio was chosen so that the wing had a good balance between being too tapered and not tapered at all. The span of the wing was chosen rather arbitrarily, 120 inches being the design constraint from last year's competition. Since we established the desired wing loading and span, we merely solved for the root chord and tip chord that would yield the planform area of a wing that created a force of lift of approximately 45 lbs. From there, our aspect ratio and Reynolds number could be calculated (See fig. 2)

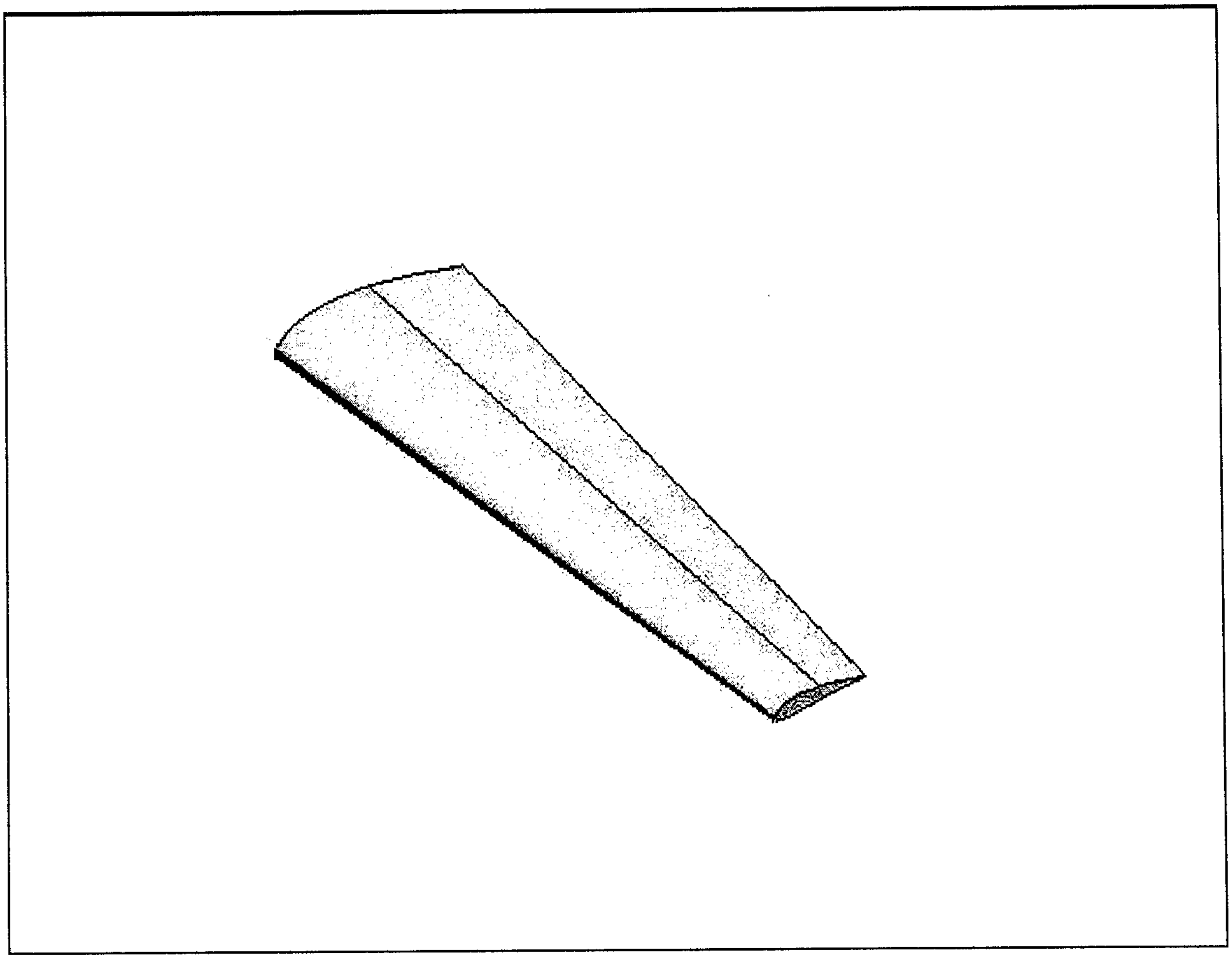


Figure 4.3 Final Wing Design

The tail of the aircraft was designed as a compromise between a RC sailplane and a pylon racer. Since the mission is to fly a course quickly and efficiently, we decided to combine the two. By combining

them, we decided on a value for the volume ratio in between a pylon racer and a sailplane. Since the configuration is a T tail, the vertical support can be made smaller due to the effect of the horizontal tail. The horizontal tail on top of the vertical tail acts as an endplate and reduces the induced drag and increases the lift slope. The vertical and horizontal tails are smaller due to the influence of the pylon racer, but the control surfaces will be larger than normal to provide good control (see Fig. 3).

The airfoils for each part of the tail are NACA0012. We chose these because we didn't want too much lift on the tail. It is a symmetric airfoil and does not need to generate a lot of force, as the moment arm of the tail is enough to control the aircraft. In addition to the horizontal and vertical tails being NACA0012, all of the control surfaces will be NACA0012 airfoils as well (i.e. the rudder, ailerons, and elevator). The performance parameters are in Table 4.3 and the assembled tail is in Figure 4.4.

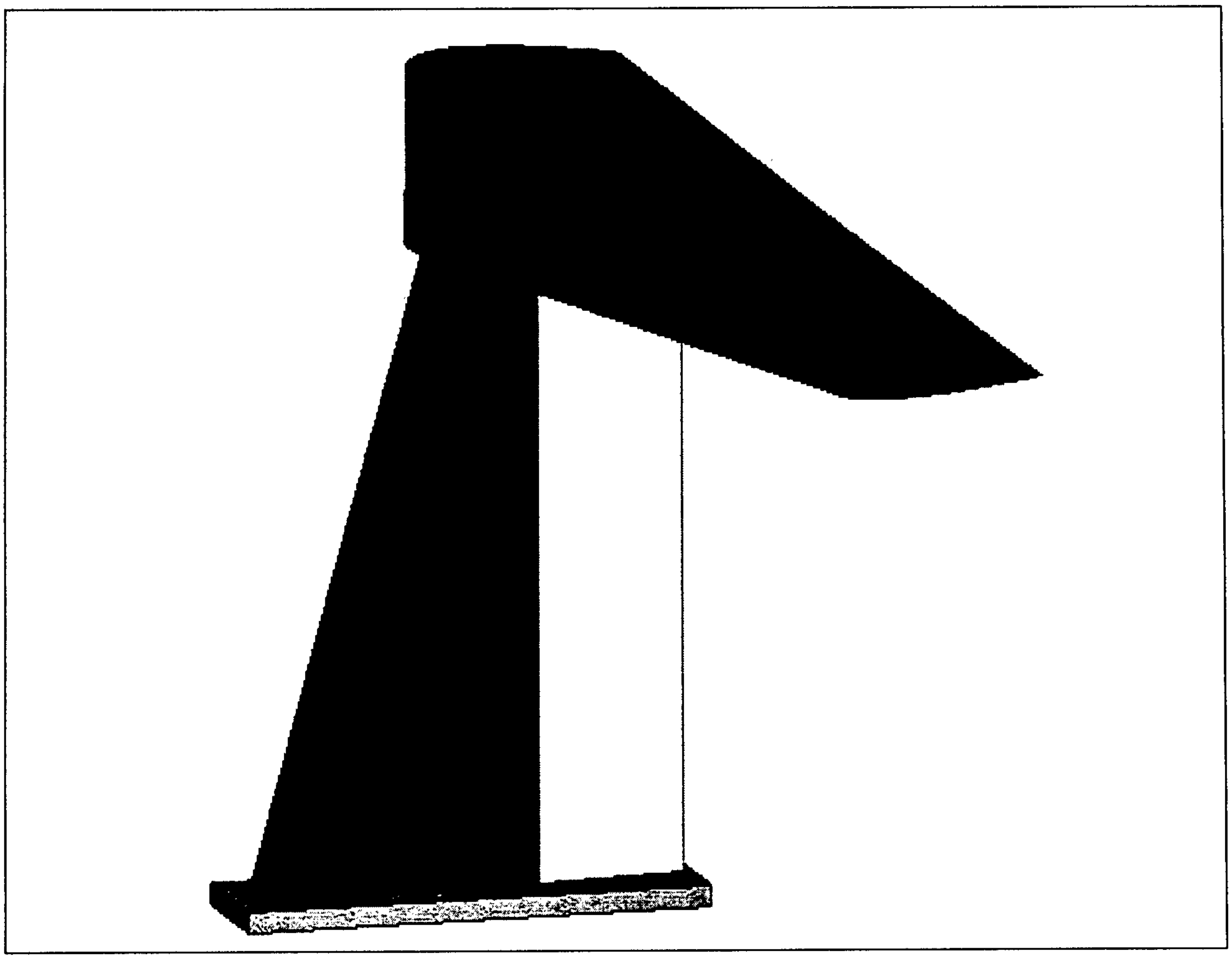


Figure 4.4 Final Tail Design

Tail Performance Properties			
	Horizontal		Vertical
Planform Area, S:	3.029	ft^2	1.125 ft^2
Span/Height, b/h:	3.692	ft	1.5 ft
Root chord:	0.984	ft	1 ft
Tip chord:	0.492	ft	0.5 ft
Taper Ratio:	0.5		

Table 4.3 Tail Performance Properties

The RAC of the wing and the tail together is as follows:

RAC Wing= 8*ft. wingspan+8*ft. root chord+3*# of control surfaces

RAC Tail = 10*vertical surface+10*horizontal surface

The overall cost is 80+12+6+10+10 = **118**. This is the cheapest the wing can cost while still being tapered, 10 ft. long, and maintaining the same planform area.

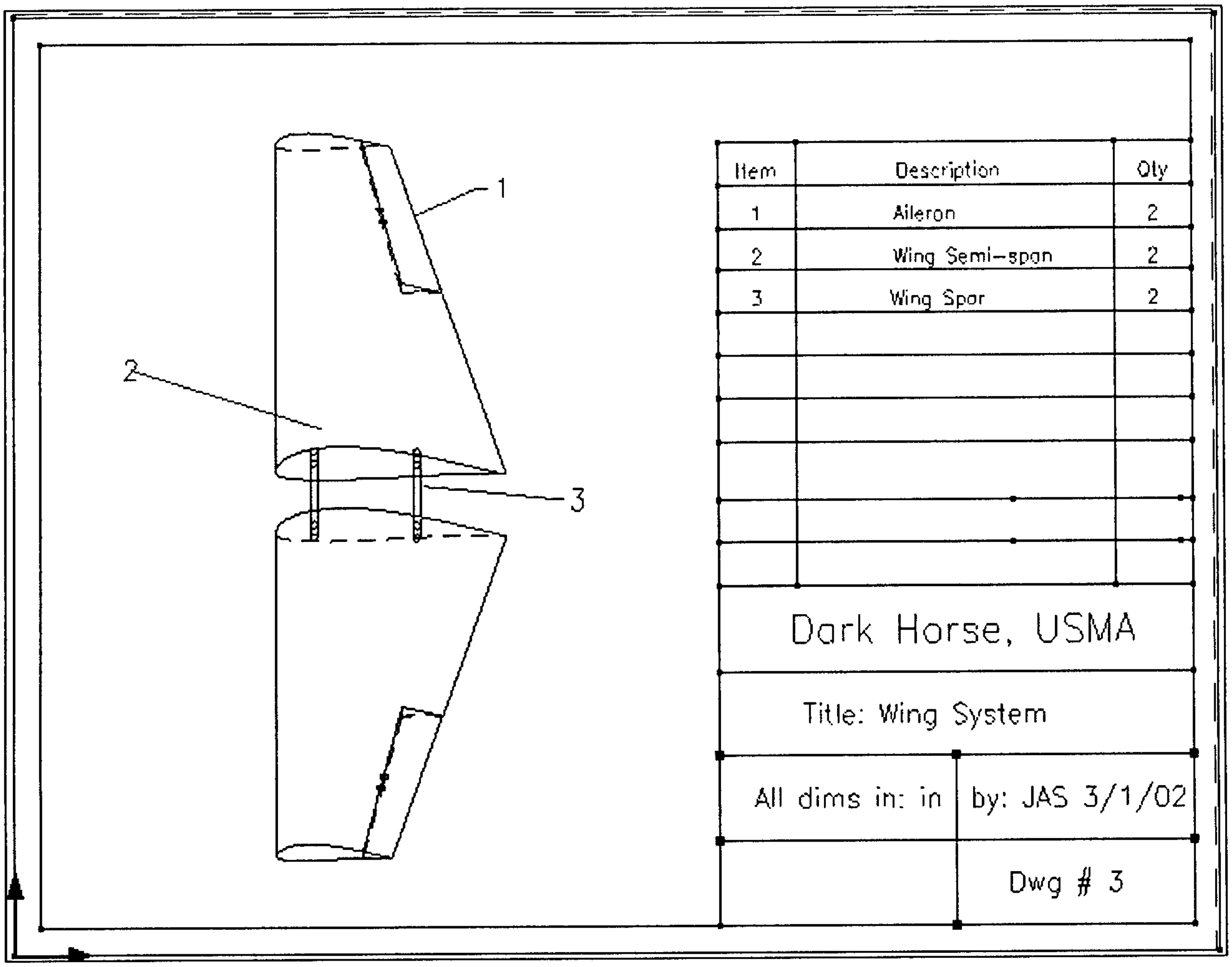
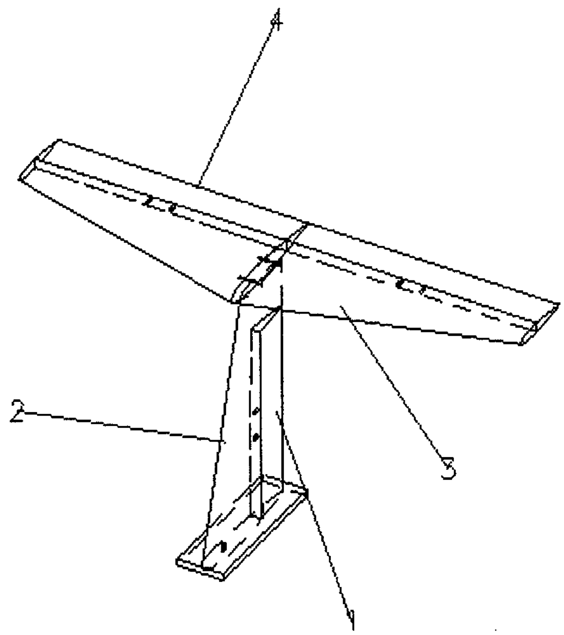


Figure 4.5 Wing Schematic



Item	Description	Qty
1	Rudder	1
2	Vertical Tail	1
3	Horozontal Tail	1
4	Elevator	1

Dark Horse, USMA

Title: Tail System

All dims in: in	by: JAS 3/1/02
	Dwg # 3

Figure 4.6 Tail Schematic

5. Landing Gear

5.1 Conceptual Design. The developing of ideas for the landing gear produced the following concepts: a tail dragger, and two types of tricycle gear. The tail dragger was eliminated immediately due to probability of poor handling and negative static stability on taxi. Since the center of gravity changes when the tail wheel leaves the ground, the chance of tip over increases. This type of gear also requires a great deal of effort to maintain a steady track on taxi since there is very little left-to-right stabilization after the tail lifts.

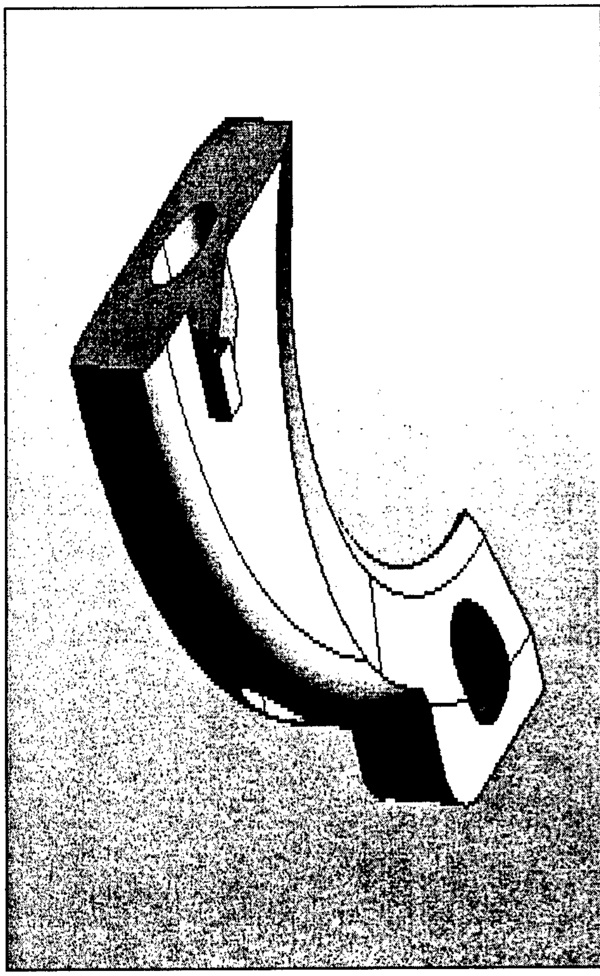


Figure 5.1 Front Landing Gear Concept

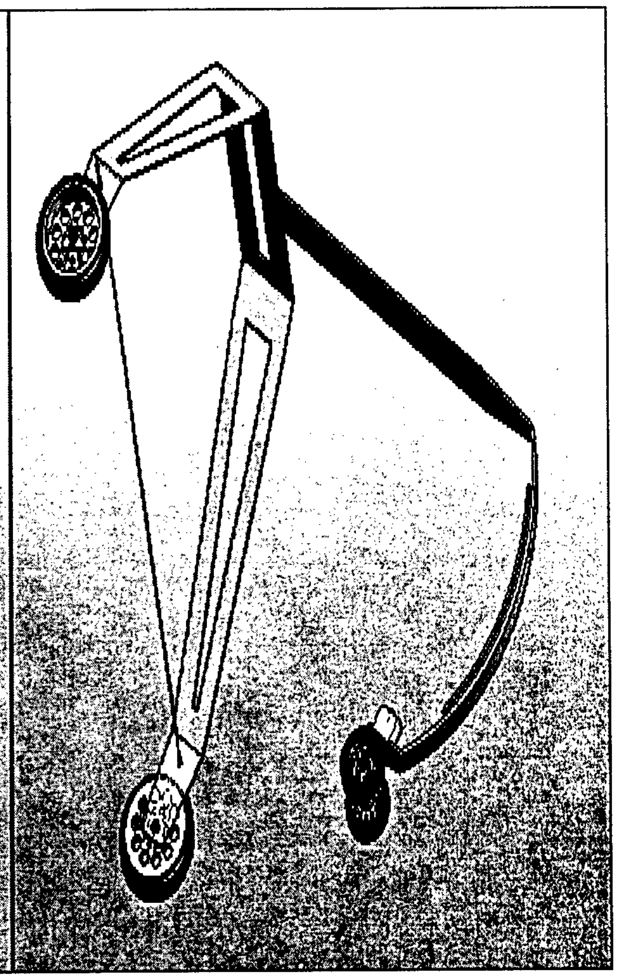


Figure 5.2 Total Landing Gear Concept #1

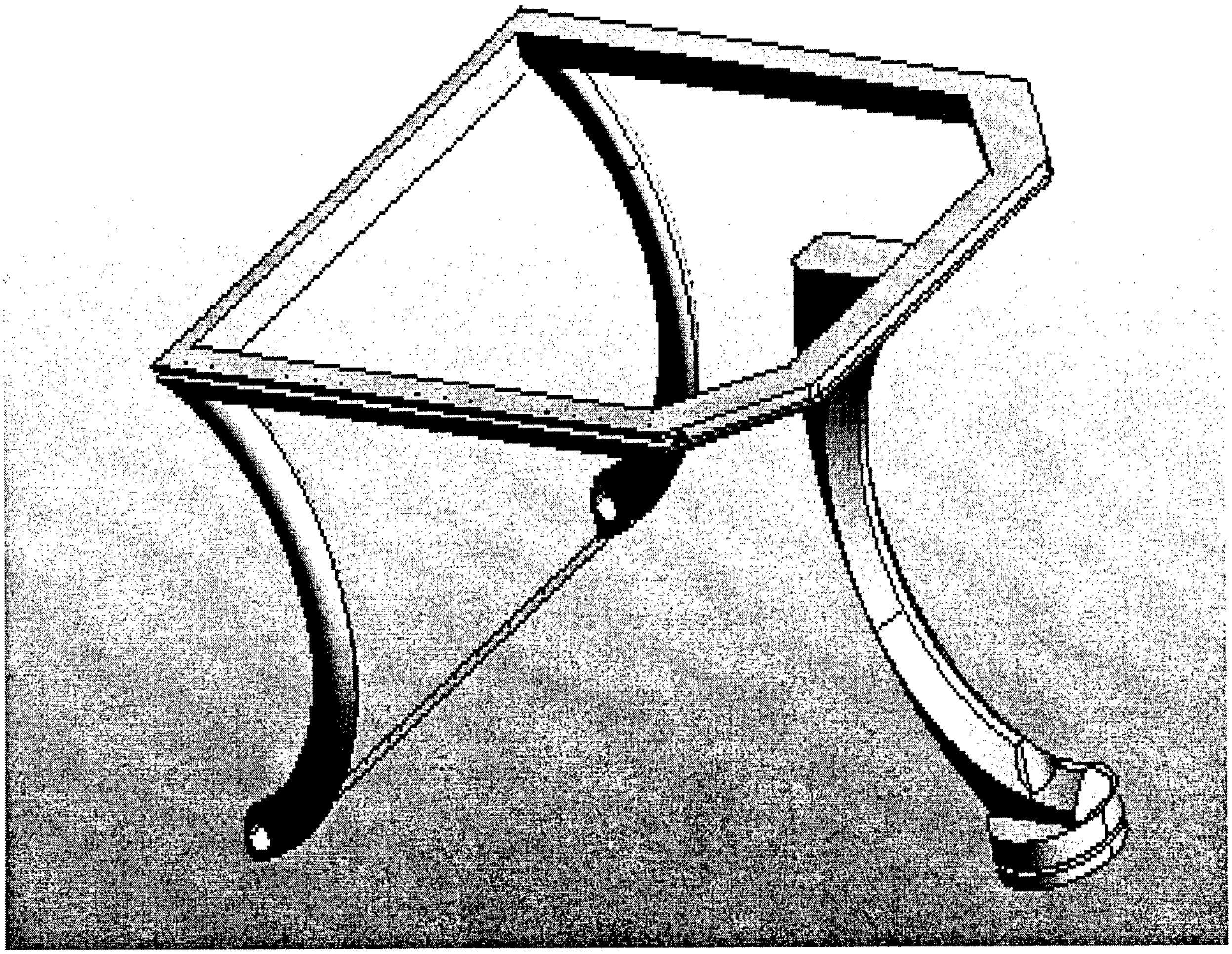


Figure 5.3 Landing Gear Concept #2

Design Parameters	Why important
Low frontal and side area	Reduces power losses
High memory	Could cause strut failure or prop strikage
Good shock redirection and dissipation	Prevents airframe failure
Low weight	Reduces power losses
Steady handling	Well controlled taxi for efficient take-offs/stability and recoverability on out-of-shape landings
Consistent Ride height	Prevents prop, fuselage, or wingtip strikage
Medium Tire hysteresis	Maintains ride height, and reduces power losses without significant loss of traction
High wheel strength and deflection	Prevents strikage on impact and maintains steady handling

Table 5.1 Landing Gear Design Parameters Importance

Design Parameters	Weight	Performance Criteria
Low frontal and side area	5	< 6in^2
High memory	1	No permanent deformation after five ft drop with 100lbs
Good shock redirection and dissipation	6	No deformation of test spine during drop test
Low weight	7	Less than 7lbs
Steady handling	8	Straight tracking on taxi test, turn radius less than 5ft with no tippage
Consistent Ride height	4	< 4in deflection
Medium Tire hysteresis	2	Minimal slippage during taxi test
High wheel strength and deflection	3	No deformation after drop test

Table 5.2 Landing Gear Design Parameters Performance Criteria

The assumptions made for the concepts shown above are: the majority of loading caused by the impact of touchdown will take place on the main gear, the aircraft will land under power, the pilot will be experienced and able, and the fuselage will absorb any shock transmitted to it through the landing gear. Additionally, the landing gear design is based on the fact that wing placement will coincide with the center of gravity to keep the aerodynamic center slightly rearward of the center of gravity, and main gear placement will be at quarter chord of the main wing. Due to the center of gravity being ahead of the aerodynamic center, the plane may have the tendency to nose in if velocity drops or stall occurs. This will occur if the aircraft must land without power. However, the center of gravity and wing placement will be adjusted to prevent nose-first attitude at low velocities and the pilot should be able to flare just before landing, if power returns. The landing gear will be connected to the fuselage via the main support member, and the main gear was designed with shock dissipation in mind.

Figures of Merit	Taildragger	Tricycle 1	Tricycle 2	Weight
Low frontal and side area	2	0	1	5
Good shock redirection and dissipation	1	1	1	6
Low weight	2	0	1	7
Steady handling	-2	1	1	8
Totals	14	14	26	

Table 5.3 Pugh's Method for Landing Gear Concepts

Since the material and tire choice can be similar for each of the concepts, only the four most differing parameters were compared. Of the three choices, the second tricycle gear was chosen for final production due to its well-rounded performance compared to the other two concepts.

5.2 Preliminary Design. The parameters that will be most important in the preliminary design phase are material selection and geometry. The size and material of each member will decide the ability of the gear to dissipate energy from the impact of touchdown. The geometry, along with tire and wheel selection, will also decide the handling characteristics of the aircraft on taxi.

Figures of Merit	One-piece Tricycle	Two-piece Tricycle
Material Choice	Limited to one type for nose and main gear	Freedom to make nose and main gear have differing properties
Energy Dissipation	More material will spread load and dissipate energy better	Different material choices for nose and main gear will allow different properties to dissipate energy
Cost	Money saved for one material; money spent for more material	Less material needed saves cost
Manufacturing	Difficult due to complexity of making entire piece at once and amount of raw material needed	Easier because of simplicity of design and smaller amount of raw material to work with

Table 6.4 Figures of Merit and Mission Objectives

5.2.1 Wing and Power Loading Requirements. One-piece gear would weigh more than necessary to meet mission standards, and would have greater drag, creating higher power and lift requirements for the aircraft. Two-piece landing gear allows the freedom of material choice to tailor properties for more shock dissipation in the main gear and more support in the nose gear in order to keep the prop from striking the ground on impact. This configuration will also cause the aircraft to right itself in the event that it lands nose first.

At this point, it was decided that the best type of landing gear for this aircraft is a two-piece tricycle gear. A tail dragger would reduce the likelihood of tipping in the event that the aircraft taxis through rough terrain, better than the tricycle gear would. However, we are assuming that the plane will be taxiing on a smooth surface and will not encounter trouble some rough areas. Since the plane will be taking off and landing on a paved runway, there is little chance that it will traverse very rough terrain on a regular basis. Therefore, the tricycle gear will be sufficient to prevent tipping during taxi. Since the two-piece gear could be made out of any material, it could weigh much more than a one-piece design.

However, careful material selection will prevent this. The assumptions made during the preliminary phase are narrower than those made in the conceptual phase. Taking the requirements of the landing gear one step further, it becomes clear that material choice and geometry will play a major role in its success.

5.3 Detail Design.

5.3.1 Performance data (takeoff, handling qualities, g-loads). Performance of the landing gear is based on its ability to absorb loads caused by landing, maneuvering the plane during taxi, and provide for efficient take-off and flight by reducing rolling resistance and drag. The geometry and material selection of the landing gear will protect the wings, propeller, and fuselage from striking the ground on impact by allowing the landing gear to deform by only a small amount. However, the gear must dissipate the energy without sending it straight into the fuselage, so the nose gear is shaped to bend in the direction of forces directed either directly upwards, or from front to rear. This will minimize the moment on the airframe by allowing the moment arm to bend in the direction of the load, transmitting the force in a normal direction. A set of spacers will further dissipate loads before they reach the airframe. The main gear is designed so that forces and loads are directed upwards, with the assumption that the airplane will land with a nose-high attitude. A nose high attitude will allow the gear to transmit a reduced normal force to the airframe, preventing the shear stress that could lead to connection failure. It will also minimize the moment on the airframe by directing the load through the gear, reducing the moment arm. A set of spacers will further diminish the load before it reaches the airframe.

Tire selection will play a role in handling on the runway, and will further dissipate forces on touchdown. It will also reduce vibration caused by surface irregularities on the runway. The track width will prevent rollover from occurring in the event of a landing on one wheel, and the wheelbase from nose to main gear will make turning easy, while preventing the airplane from tipping.

The gear set up will work very well, with proper placement on the aircraft. Further testing will be necessary to set the wheelbase, and the angle of attack on taxi will be taken into account when finalizing the ride height of the aircraft. If the angle of attack is too high, the plane may have a tendency to hop prior to takeoff and during landing. If the angle of attack is too negative, the wing will create down-force instead of lift, and the plane may not able to take off.

5.3.2 Component Selection and Systems Architecture. The material selected for the nose gear strut is 1075 spring steel that has been heat treated for added strength and memory. The spring steel allows for a very small area and weight with good returns in memory and elastic deformation. The rear gear is aluminum 5052, chosen for its ductility over aluminum 6061, which would break during manufacture. The 5052 alloy is ductile so a wire is needed to connect the legs for added rigidity, but the weight is very low, and the frontal area is minimal. The side area may be great compared to steel, but the weight savings make up the difference and drag is not drastically affected by it. Wheels are plastic with foam tires covered with rubber. They were chosen over hard plastic, pneumatic, and plain foam for their light weight, rigidity, and height.

The assumptions made at this point are that the material choices are the best for their jobs and that performance outweighed manufacturing costs. Since a skilled machinist and technician recommended the material for the nose gear, it was a safe choice. However, working the metal into its shape and then heat-treating it on premises may lead to some inconsistencies in its properties. The main gear is assumed not to reach plastic deformation due to the added strength of the wire connecting its legs, which is a safe assumption, since the wire can be custom fit to prevent yielding. A similar design has worked well in the past, even without the wire. The manufacturing costs are low, since there is a machine shop on premises, complete with experienced and professional technicians.

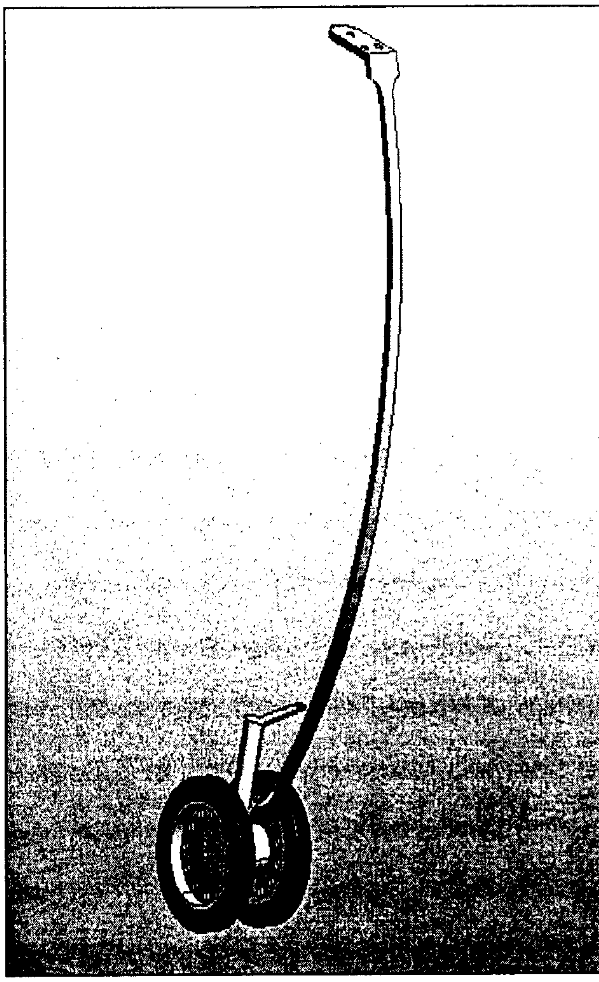


Figure 5.4 Final Concept Nose Gear

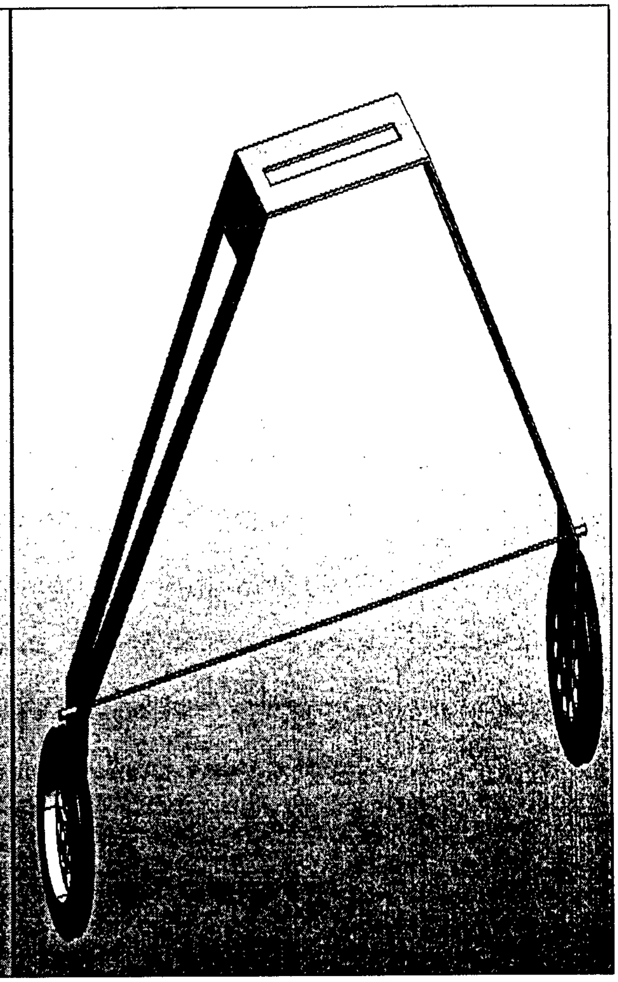


Figure 5.5 Final Concept Main Gear

6. Electronics

6.1 Conceptual Design. Key design constraints of the power and electronic components of the aircraft are as follows. The electric motor needs to be from the AstroFlight or Graupner family of brushed electric motors. A fuse connecting the positive battery terminal to the speed controller limits the current draw of the electric motor to 40 Amps. The batteries must be NiCad in type, and have a maximum weight of 5.0 lbs. Separate batteries must power the radio receiver and servos. The plane must also have a radio fail-safe mode that is automatically selected during a loss of signal. During fail safe, the aircraft receiver must select the following parameters: throttle closed, full up elevator, full right rudder, full right aileron.

The block diagram shown below in Figure 1 shows the layout of the electronic components on the aircraft. The main power source for the aircraft is the main battery pack. This battery pack powers the motor, which turns the propeller. A radio signal is sent from the transmitter on the ground to the receiver on the plane. The receiver sends a signal to the speed controller to control the current draw of the motor. It also sends a secondary signal to the servos, which manipulate the control surfaces to guide the plane in flight.

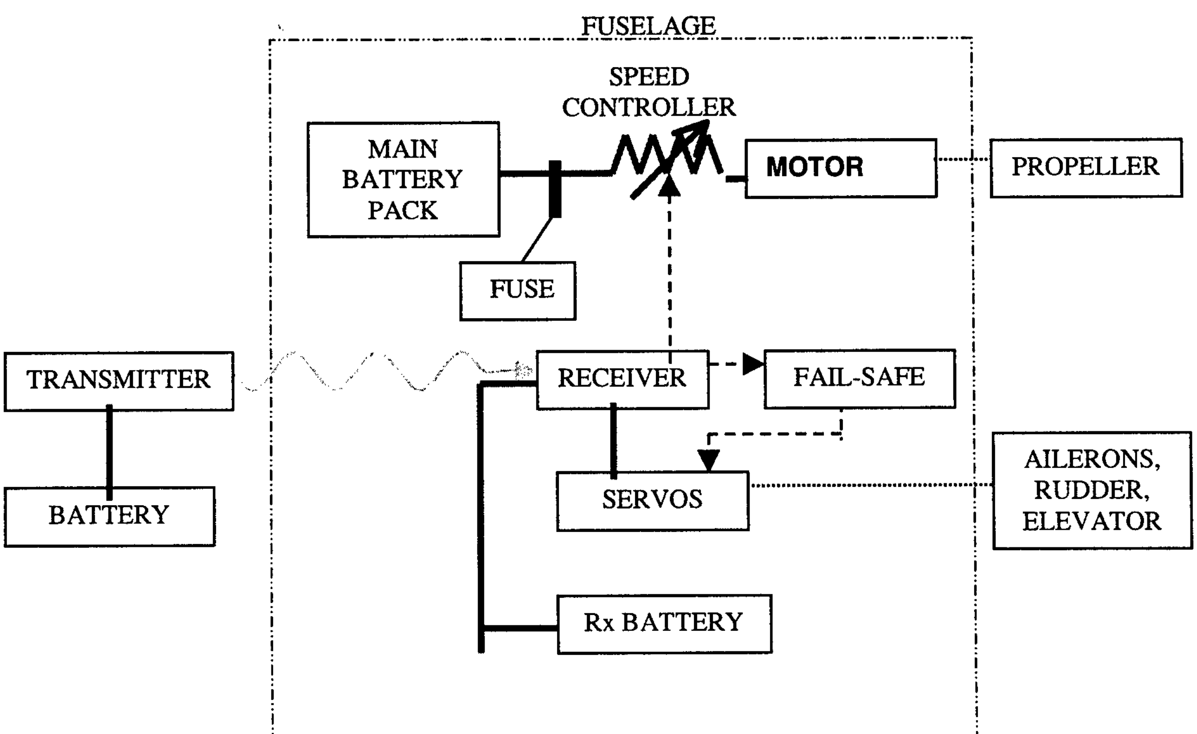


Figure 6.1 Electronic Block Diagram

The main concern in powering the aircraft is in providing enough thrust to lift the plane off of the ground within the given 200 ft. runway. We calculated the necessary power and thrust given our wing design, and we used these numbers as constraints for our motor selection. The wing design team then gave me their calculations of the power and thrust necessary to lift the plane off of the ground and the specifications for maintaining the plane in flight. Given a model weight of 55lbs., the aircraft would require a thrust of 200 oz. and a power of 1137W to liftoff from the runway in 196ft. The plane would then require a much smaller 21.6oz. thrust and a power of 156W to maintain straight and level flight. A second consideration for this project was whether to use more than one motor. We decided that the penalty in rated aircraft cost for using additional motors on the plane outweighed any benefit of having more than one motor.

6.2 Preliminary and Detail Design. To get an initial idea of what size motors we would need to use, we examined the aircraft from previous competitions. The AstroFlight website lists the characteristics of each AstroFlight motor in the company's inventory. The motors in the airplane category only differ in size and in whether they have gearboxes attached to them. AstroFlight produces motors with built in gearboxes which offer better efficiency and thrust when compared to a motor of the same size. We examined the AstroFlight motor specification data and selected the only motor which met our power and thrust requirements for liftoff. We then selected our battery cells based on the voltage and amperage required by our motor to output 1137W power and 200oz. thrust.

The NiCad battery pack must provide the necessary current and voltage to run the motor. The goal for this electronic component is to provide as much runtime as possible. At the very least, the battery pack must power the plane for 10 minutes of flight and at least 1 scoring sortie, consisting of 2 unloaded laps around the contest course. Ideally, the battery pack would power the plane for more than 10 minutes including 2 unloaded and 1 loaded sortie. The primary constraint on the battery pack is the 5.0 lb. weight limit. Consequently, the first battery characteristic that we examined was mAH/weight of the battery. We examined NiCad batteries ranging from 300mAH to 5000mAH in size. When we examined these batteries, we noticed that in general, larger batteries have a higher mAH to weight ratio. The mAH to weight ratio for Sanyo and SR batteries is shown in Table 1 below. The table also shows that under a capacity of 2000mAH, Sanyo batteries have a higher ratio of capacity per weight than their

SR battery counterparts. However, SR batteries with a capacity greater than 2000mAH have a greater mAH to weight ratio than Sanyo batteries.

Battery	MAH	Weight	Ratio	# within 5lbs.
SR 300	300	0.5	600.000	160
SR 600	600	0.91	659.341	88
SR 500	500	0.66	757.576	122
Sanyo KR-2000C	2000	2.64	757.576	31
Sanyo NT-40AAK	400	0.46	869.565	174
SR 1600	1600	1.8	888.889	45
SR 850	850	0.94	904.255	86
SR 1300	1300	1.4	928.571	58
SR 5000	5000	5.3	943.396	16
Sanyo KR-600AE	600	0.63	952.381	127
SR 1100	1100	1.1	1000.000	73
Sanyo N-3000R	3000	2.96	1013.514	28
Sanyo CP 1300SCR	1300	1.23	1056.911	66
SR 1400	1400	1.31	1068.702	62
SR 2000	2000	1.86	1075.269	44
Sanyo RC-2400SCR	2400	2.09	1148.325	39
SR 3000	3000	2.47	1214.575	33
SR 2400	2400	1.9	1263.158	43
Sanyo KR-1400AE	1400	1.09	1284.404	74
Sanyo KR-1100AAU	1100	0.83	1325.301	97
Sanyo KR-1700AU	1700	1.27	1338.583	63

Table 6.1 Battery Data

A second constraint for battery cell selection is the voltage requirement of the motor. These mAH to weight ratios must be combined with the voltage requirement to arrive at the best battery configuration. In a battery configuration, connecting battery cells in series increases the pack voltage, while connecting battery cells in parallel increases the endurance of the battery pack. These facts must be taken into consideration when choosing battery cells and configuration. Increasing the lines of batteries in parallel would increase the pack endurance, but it was not possible to place the batteries in parallel while satisfying the motor voltage and contest weight constraints.

The following shows the calculation of flight time.

$$\frac{\text{BatteryCapacity}(\text{A}\cdot\text{hr})}{\text{MotorCurrent}(\text{A})}\cdot\frac{60\ (\text{min})}{1\ (\text{hr})} := \text{FlightTime}(\text{min})$$

Equation 6.1: Flight Time

Assume 1.5 minutes at max throttle (5 sec max climb, plus takeoff time and compensating for turns)

$\text{MaxThrottleFlightTime} := 1.5$ minutes

$\text{BatteryTotal} := 2.4$ Amp·hrs

$\text{MaxCurrent} := 40$ Amps --Max current dictated by 40 amp fuse

$\text{LevelFlightMotorCurrent} := 4$ Amps --This current produces a power of 100W, more than double the 47W needed for straight and level flight.

Calculate Amp hours left in battery

$$\text{BatteryLeft} := \text{BatteryTotal} - \text{MaxThrottleFlightTime}\cdot\frac{1}{60}\cdot 40$$

$\text{BatteryLeft} = 1.4$ Amp·hrs

Calculate additional flight time at a lower current setting

$$\text{FlightTimeLeft} := \frac{\text{BatteryLeft}}{\text{LevelFlightMotorCurrent}}\cdot\frac{60}{1}$$

$\text{FlightTimeLeft} = 21$ minutes

$$\text{TotalFlightTime} := \text{MaxThrottleFlightTime} + \text{FlightTimeLeft}$$

$\text{TotalFlightTime} = 22.5$ minutes

Based on these characteristics, we selected the SR Battery 2400 size cell for use in our battery pack. We found out from SR Batteries that their 2400 size battery has a finished weight of 2.2 oz., so we calculated a maximum of 36 batteries in our pack. Since battery endurance is a major factor in aircraft performance, we decided to use as many battery cells as possible in the battery pack. Using the equation above, we determined the flight time of the plane at full throttle, or 40 Amps, to be 3.6 minutes. Although this runtime is much lower than the goal of 10 minutes, these calculations do not take into account the lower current used by the plane while it is in the air. This lower current draw will greatly lengthen flight time for the aircraft. We recalculated flight time using 1.5 minutes at full throttle and a current draw equally that needed for straight and level flight, and we arrived at a total runtime of 22.5

minutes. However, we realized that the battery would only be able to provide this long a runtime under ideal conditions. To test for actual runtime, we will need to run the motor at specific throttle settings which fit the given mission profile of 3 sorties at 2 laps each.

A final consideration for the battery pack is the physical placement of cells within the battery pack. We wanted to minimize the size of our battery pack to leave as much room as possible within the fuselage for the payload of softballs. To do this, we used a block configuration that placed the cells in a battery back 3 cells long, 3 cells wide, and 4 cells deep.

The last component of the electric power system is the speed controller. The speed controller controls the throttle of the aircraft by limiting the current drawn from the battery to the motor. The main requirement for the speed controller is that it should be able to handle a current draw of 40 Amps from the battery. Speed controllers have a maximum current which they can handle before the MOSFETs controlling the current burn out. The current from the battery is limited to 40 Amps by the 40 Amp fuse between the battery pack and the speed controller. If the speed controller cannot handle 40 Amps of current, however, then there is a chance that they speed controller will burn out, since the fuse will only trigger at currents above 40 Amps. Since we chose an AstroFlight motor to drive our plane, we chose an AstroFlight speed controller to go along with this motor. We realized that companies manufacture their equipment to perform with their own specific products. We reasoned that using an AstroFlight speed controller to control the throttle on an AstroFlight motor was a good idea. We knew that there would not be a compatibility problem when using these two parts together.

Servos are also used to fly the airplane. Four servos will be used on this aircraft to manipulate the control surfaces to guide the plane in flight. Two servos, one located on each wing, will control the wing control surfaces. A third servo, located on the fuselage, near the tail, will control the rudder. A fourth servo, also located at the rear of the fuselage, will control the elevators.

7. Manufacturing Plan

7.1 Introduction. The basis of the manufacturing was simplicity. Furthermore, the goal of the manufacture of the airplane was to construct the plane with best performance out of low cost materials. Craftsmanship and design were key, for the team could not afford all of the best and expensive materials. Each of the components was to be constructed separately. However, since one of the keys to the project is concurrent engineering, the individual teams needed to work together to ensure the proper connections.

7.2 Constraints.

7.2.1 Cost. The cost of materials was a great consideration during the construction of the airplane. We were limited to a budget of $3,000 given as a gift by the Academy's Association of Graduates. As a result, the materials available to construct the airplane limited the considerations given to building different aspects of the airplane. A running budget was kept, and all ordering decisions went through the Project Manager to ensure we did not go over budget.

7.2.2 Skill Level. The majority of the processes required to construct the airplane were within our skill level. We were able to construct everything ourselves, except for the landing gear. Constructing a wing out of foam, using epoxy and fiberglass, and wiring the connections were all simple tasks. The welding required for the landing gear was performed with the assistance of the laboratory technicians.

7.2.3 Time. Initially, time was not very constrained. We planned from the middle of January to the end of February for the construction of the airplane. However, supply problems and management problems pushed the time back to late February for the start of construction. Simpler construction designs/plans were necessitated in order to finish the plane in enough time to test and present to the faculty prior to the competition.

7.3 Overview of Manufacturing Plan

7.3.1 Fuselage. The initial idea for the fuselage was to make a mold on a rapid prototyping printer and then cover the mold with an epoxy matrix and carbon fiber fabric. However, the rapid prototyping printer could not handle the size of the mold necessary to construct the fuselage. Instead, the backup plan was put into effect. An aspen spar substituted for the aluminum spar. To strengthen the aspen, and increased its torsional and axial stiffness, the spar was coated with epoxy and fiberglass weave. The layup of the

fabric was set at forty-five degrees angle. As a result, only one layer was needed to improve the torsional stiffness in either direction. On top of the fabric at forty-five degrees angles, another layer was placed for longitudinal stiffness (the fibers ran perpendicular and parallel to the longitudinal axis of the spar). Both these layers were place on the top of the spar, due to the tensile strength needed on the top of the spar. On the bottom of the spar, a single layer, running perpendicular and parallel to the longitudinal axis, was added for additional strength and performance.

The mold of the fuselage was made out of balsa and aspen ribs. The sections were then covered with 1/32 inch plywood and covered with epoxy and fiberglass. The goal was to provide strength over the entire bottom of the fuselage to prevent failure upon landing. Monokote covered the fuselage to reduce drag and increased the aerodynamic efficiency. The construction took longer than expected due to the inexperience of the team, a couple of redesigns, and not seeking the guidance of the lab technicians and adviser sooner. Two aluminum rods were placed through the ribs to provide added support and a spot for the connections of the wing, landing gear, tail, and motor mount. The design should work well, for it leaves a good margin for error.

7.3.2 Wing and Tail. The wing and tail were constructed out of Spyder foam. The initial idea was to use pink insulation foam available at Home Depot. However, a change was made to blue Spyder foam. The Spyder foam is twice the weight, but over four times the strength. The benefits of the Spyder foam were much greater than the costs. The program Compufoil produced airfoil templates that included placement of the carbon fiber rods that were added to increase the strength and stiffness of the wing. The cutouts in the main body of the wing were filled with expansive foam to keep the rods in place. The rods extended through half of the wing span and the different sections were put together using epoxy.

When we decided to use foam wings, we recognized the problem of structural discontinuity due to constraints of the wire cutters used for cutting. In lieu of this fact, we explored epoxying two one foot pieces of blue foam together and compared its breaking strength to one two foot section. The epoxied material has a lower breaking strength, therefore we need something that will transfer the load between sections, which is where the carbon fiber rods come in to play. The carbon fiber rods act as mechanisms to transfer the load between the sections of the foam cut wing.

To finish the wing, Monokote and heat were used to seal the foam and reduce drag. Cutouts were left for the placement of servos and other vital electronics and control surfaces. The cost of the wing was a little expensive- the carbon fiber rods were about thirty dollars apiece. However, the wing is one of the vital components of the plan and no expense should be spared in building it. The carbon fiber rods were chosen for their axial strength as well as their ability to flex. The rods withstand the 2G-load test by themselves, and therefore the wing will be even better with the foam and Monokote. Additionally, the flexion of the rods will provide in-flight dihedral (similar to a C-5 Galaxy). This decision reduced the manufacturing time for the wing and eliminated the problems inherent with a foam wing with a dihedral. The assumption is justified because both are big planes designed to carry very heavy loads.

7.3.3 Landing Gear. The manufacturing plan includes hand-making the metal parts of the gear, along with the spacers and the axle assemblies for the rear wheels. The wheels were chosen from strollers, since they are very lightweight and capable of withstanding moderate loading. The nose gear will incorporate a hub and two wheels from an umbrella stroller, and the main gear will use two-four large stroller wheels. All gear and wheels will be faired.

Materials are readily available and the machine shop and technicians are at our disposal, so the structural members are easily hand made. The wheels and hubs are more difficult to make, since they are plastic so they will be purchased and bolstered with axles and bearings stout enough for this application. Wheel assemblies from industrial and hobby applications were investigated, and found to be less desirable than the ones that we chose.

Manufacturing Milestones	Planned	Actual
Component construction complete	25-Jan	4-Mar
Construction and assembly complete	28-Feb	11-Mar
First Flight Test	4-Mar	13-Mar
Testing Complete	19-Apr	3-Apr

Table 7.1 Manufacturing Milestones

8. Rated Aircraft Cost

Rated Aircraft Cost

$A := 100$

$B := 1500$

$C := 20\cdot\frac{1}{hr}$

$MEW := 35 - 8.55$

$MEW = 26.45$ lbs

$BatteryWeight := 5$

$REP := [\,1 + 0.25(1 - 1)\,]\cdot BatteryWeight$

$REP = 5$

$WBS_{Wing} := 8\cdot\frac{hr}{ft}\cdot(10ft) + 8\cdot\frac{hr}{ft}\cdot(1.5ft) + 3hr\cdot 2$ two control surfaces

$WBS_{Fuselage} := 10\cdot\frac{hr}{ft}\cdot(6ft)$

$WBS_{Empenage} := 10hr\cdot 1 + 10hr\cdot 2$ one vertical stabilizer and one horizontal stabilizer

$WBS_{FlightControl} := 5hr\cdot 4 + 5hr\cdot 1$ four servos and one motor controller

$WBS_{Propulsion} := 5hr + 5hr$ one propeller and one motor

$MFHR := WBS_{Wing} + WBS_{Fuselage} + WBS_{Empenage} + WBS_{FlightControl} + WBS_{Propulsion}$

$MFHR = 223hr$

$$RAC := \frac{(A\cdot MEW + B\cdot REP + C\cdot MFHR)}{1000}$$

$RAC = 14.605$

RAC=14.605 or $14,605

9. Resources

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